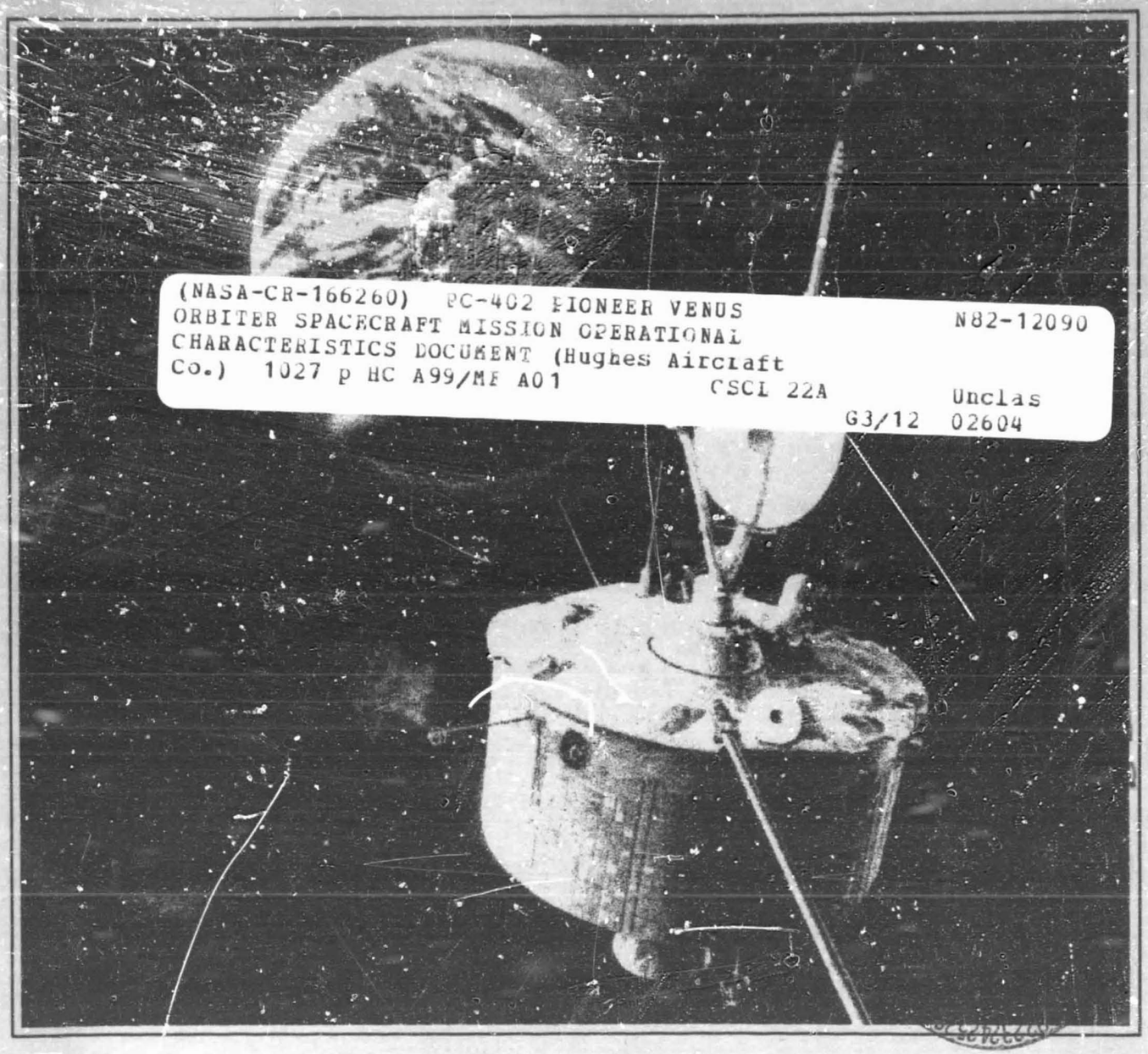
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- F.C. Barker
- L.W. Butterworth
- R.E. Daniel
- R.J. Drean
- K.A. Filetti
- J.N. Fisher
- L.A. Nowak
- J. Porzucki
- J.O. Salvatore
- G.A. Tadler

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1.0 INTRODUCTION

The purpose and organization of this document, applicable documents; and the Orbiter Mission Objectives and basic description, are presented below.

1.1 DOCUMENT PURPOSE

The Pioneer-Venus Orbiter Operational Characteristics document was generated by Hughes Aircraft Company between September 1976 and June 1977 for the National Aeronautics and Space Administration, Ames Research Center, under Contract Number NAS 2-9366. The purpose of this document is to describe the operational characteristics of the Orbiter Spacecraft and its subsystems in extensive detail to the NASA/ARC personnel having the responsibility for conducting the Pioneer-Venus Orbiter Flight Mission operations.

1.2 DOCUMENT ORGANIZATION

Section 2 is a summary description of the Orbiter spacecraft at the system level. It includes descriptions of the nominal phases, system interfaces, and the capabilities and limitations of system level performance.

Section 3 presents Orbiter Spacecraft functional and operational descriptions at the subsystem and unit level. The subtleties of nominal operation as well as detailed capabilities and limitations beyond nominal performance are discussed. A command and Telemetry Logic flow diagram for each subsystem is included. Each diagram identifies in symbolic logic all signal conditioning encountered along each command signal path into, and each telemetry signal path out of the subsystem.

Section 4 describes how the Orbiter Spacecraft performs in normal operating modes that correspond to the performance of specific functions at the time of specific events in the mission. Principal backup means of performing the normal Orbiter operating modes are included.

Appendix A is a listing of all telemetry grouped by subsystem, and essential pertinent operational

characteristics, including source derivation and power source. Source derivation is the generic signal source, prior to any signal conditioning, of the regulated or semi-regulated voltage applied to the generic signal source, starting with the Power Bus Source and tracing the power path to the generic signal source.

Appendix B is a listing of all telemetry in alphabetical order for the mnemonics. It excludes power source and source derivation, repeats some information from Appendix A, and includes some additional information such as Format assignment and TM Word (and BIT) assignment.

Appendix C is the collection of all foldout drawings and diagrams referenced throughout the document.

1.3 ORBITER MISSION OBJECTIVES

The basic spacecraft objectives of the Orbiter Spacecraft are: (1) to establish, during the 1978 opportunity, an orbit about Venus with an appropriate 24-hour period, periapsis altitude between 150 and 260 kilometers, and inclination of 105°, and a periapsis location in the northern hemisphere; and (2) to maintain orbital conditions for a minimum of 243 earth days while measurements via use of an on-board S-band transmitter; an X-band transmitter (occultation data only), and twelve on-board scientific instruments are made.

These transmitters and instruments will provide data that will be used to satisfy the scientific objectives of the mission, that are: (1) Determine the detailed structure of the upper atmosphere and ionosphere by in situ measurements, (2) observe interaction of the solar wind with the Venus ionosphere and with Venus' small magnetic field, (3) observe the characterisitcs of the atmosphere and of the solid surface on a planetary scale by remote sensing, (4) measure the gravitational field harmonics by effects on the spacecraft's orbit, and (5) detect gamma ray bursts for a solar system interferometer network.

Twelve scientific experiments are to be conducted by the twelve scientific instruments, and five radio-

science experiments will derive data from analyses of the radio signals from the spacecraft.

1.4 MISSION SYSTEMS DESCRIPTION

The top-level elements of the Orbiter Mission consist of the following systems: (1) The Orbiter Spacecraft that will journey to Venus' atmosphere to obtain scientific data for a minimum of 243 earth days in orbit about Venus, (2) the Atlas SLV-3D/Centaur D-1AR Launch Vehicle that will place the Orbiter Spacecraft on the desired Type II interplanetary trajectory, (3) The Deep-Space Network (DSN) of Earth Stations that will provide the command and data links between Mission Control and the Orbiter Spacecraft. 64-Meter ground stations will be used for maneuvers, orbit insertion, and occultations. 26-meter ground stations will be used on a one-station-pass-per-day basis, otherwise. (4) The Pioneer Mission Operations Center (PMOC), located at NASA-Ames Research Center, that includes real time data processing and display equipment, off-line processing equipment for attitude determination and prediction, and the mission operations team. (5) The Mission Flight Sequence Plan and all ancillary reference documents and procedures. The Plan is segmented into nine Mission time phases for the purpose of detailed communication ahead in this document. The nine Mission time phases, in chronological order are: (1) Prelaunch through despin of HGA, (2) Attitude determination, star survey, and science checkout, (3) Jet Calibration and TCM (Trajectory Correction Maneuver) #1, (4) TCM #2 and cruise, (5) Transmitter power and bit rate changes, (6) TCM #3 and cruise, (7) Precess to VOI (Venus Orbit Insertion) attitude through VOI, (8) Configure spacecraft and trim orbit, (9) In-orbit operations.

These nine Mission Phases correlate in time to the NASA-Ames' defined set of Mission Phases that are listed in Table 1.4-1 and are illustrated in Figure 1.4-1. This information was obtained from Reference: Paragraph 1.5.33.

TABLE 1.4-1

NOMINAL ORBITER MISSION PHASE/PERIOD DEFINITIONS (NASA-AMES)

NOMINAL PHASE/PERIOD		TIME INTERVAL
1.	Test Phase	(Composed of discrete, separate periods)
1.1	Orbiter Pre-Launch Test Period	(Start GDS Testing to Start of countdown)
1.2	Multiprobe Pre-Launch Test Period	(End of Orbiter TCM2 to start of MP countdown)
1.3	Post Launch Test Period*	(End of MP TCM2 to start of Orbiter TCM3)
1.4	Approach Test Period*	(End of Bus Maneuver to VOI 48-hr)
2.	Launch Phase	(Start of countdown activities to launch +36 hr)
2.1	Countdown Period	(Start of countdown activities to liftoff)
2.2	Post Launch Period	(Liftoff to Launch +36 hr)
3.	Interplanetary Phase	(Launch +36 hr to VOI -48 hr)
3.1	Orbiter TCM1 Period	(Launch +5 days)
3.2	Orbiter TCM2 Period	(Launch +20 days)
3.3	Orbiter TCM3 Period	(Encounter -30 days)
3.4	Instrument Checkout Periods	(As required)
3.5	Orbiter S/C Calibration Periods	(As required)
4.	Orbit Insertion Phase	(VOI -48 hr to VOI +14 hr)
5.	In Orbit Phase	(VOI +14 hr to End of Mission)
5.1	Trim Maneuver Periods	(As required)
5.2	Orbiter S/C Calibration Periods	(As required)
5.3	Orbital Sequence (OSEQ) Periods	(As required)

*Carefully integrated with specific mission events in the Interplanetary Phase.

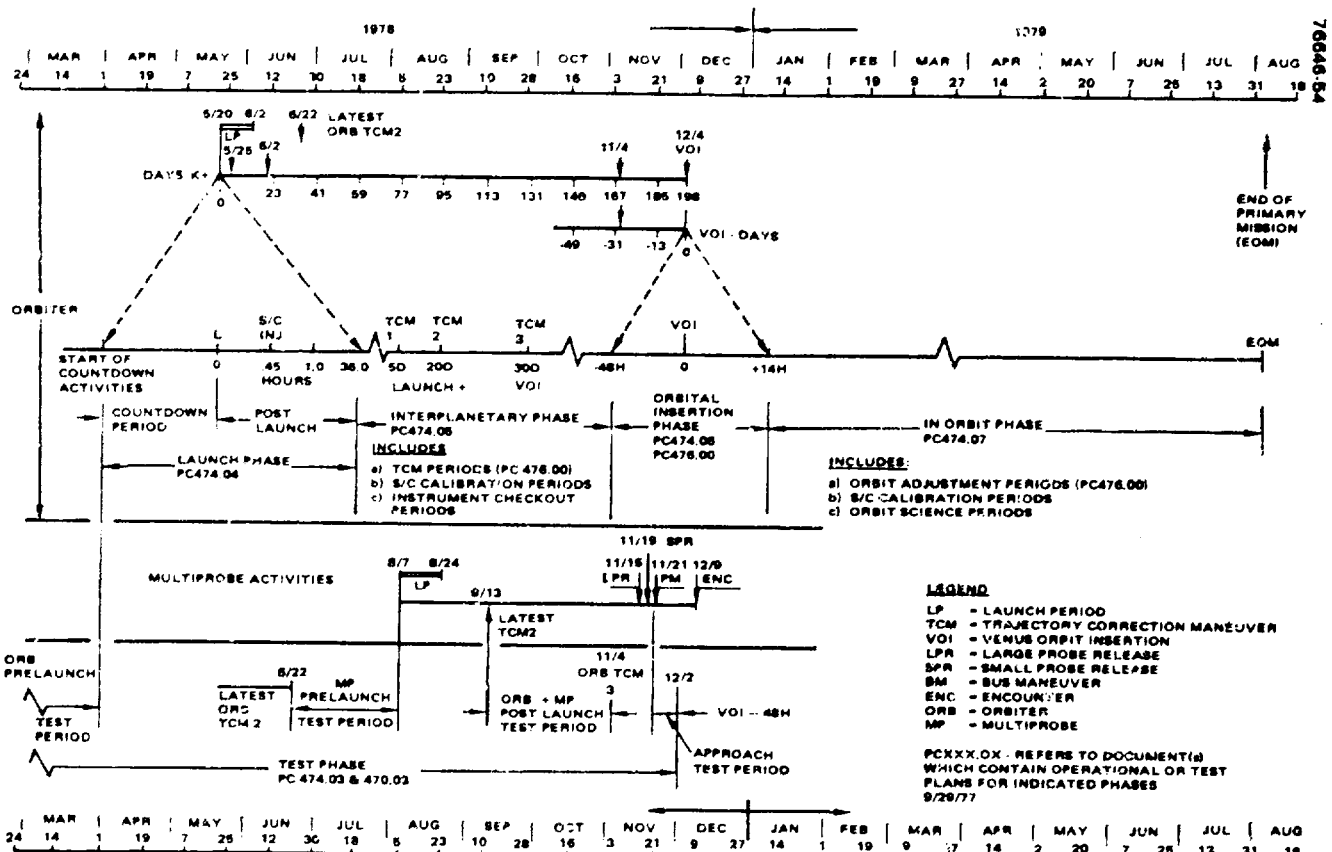


FIGURE 1.4-1. ORBITER OPERATIONAL MISSION PHASES AND PERIODS (NASA-AMES)

1.5 APPLICABLE DOCUMENTS

1.5.1 PC455 - Pioneer Venus Project Command and Control Data Processing System Software Specification, (CMDS).

1.5.2 PC454 - Pioneer Venus Project Real-Time Data Processing System Software Specification (TM).

1.5.3 Multiprobe/Orbiter Data Book - Hughes Document #HS507-5163, June 1976.

1.5.4 PC410 - Multiprobe and Orbiter Spacecraft and Related Characteristics, Revision 9, August 1976.

1.5.5 System Final Design Review - Vu-Graph Data Package - June 1976, Hughes Documents #HS507-5224 through #HS507-5228 (5 Volumes).

1.5.6 PT-402 - Pioneer Venus Orbiter Mission Trajectory Characteristics, September 1976.

1.5.7 PC433 - Orbiter Spacecraft/Scientific Instruments Interface, Revision 12, October 1976.

1.5.8 Pioneer Venus Command Assignments List - Hughes Specification #SS3639-008.

1.5.9 Centaur D-1A Systems Summary - Report No. GDCA-BNZ72-026, September 1972, General Dynamics.

1.5.10 Atlas SLV-3D/Centaur D-1A Configuration, Performance and Weight Status Report, GDC 63-0495-92, dated June 1976.

1.5.11 Centaur D-1A System Orientation, 21 April 1976, GD, Convair.

1.5.12 Rate, Clock and Status Signals Required by Pioneer Venus Subsystems and Scientific Instruments, Hughes IDC No. HS507-3738, Revision A, dated 18 November 1975.

1.5.13 Pioneer Venus Quarterly Review - February 8, 1977.

1.5.14 Mass Properties Report - Hughes Document No. HS507-2002-35, dated 31 January 1978.

1.5.15 Final Report on the Magnetic Testing of the Pioneer-Venus Orbiter Spacecraft - Hughes Memo No. HS507-7445, dated 16 January 1978.

1.5.16 Pioneer-Venus Orbiter Telemetry Data Conversion Handbook - Hughes No. SS31639-900, dated 14 July 1977.

1.5.17 Preliminary Mission Constraints and Spacecraft Signature Logs - Hughes Memo No. HS507-7044 dated 26 August 1977.

1.5.18 Pioneer Venus Criteria for "Hot RF" Switching - Hughes Memo No. HS507-6774, dated 30 June 1977.

1.5.19 PV Bus, Orbiter and Probes Electrical Loads Status Report, 11 July 1977 - Revision D - Hughes Memo No. HS507-4725.

1.5.20 Change in Initial Conditions for Telemetry Processor - Hughes AVO from G. A. Tadler dated 26 July 1977.

1.5.21 Star Sensor Flight Unit No. 2 Test Report, BBRC S/N 04, Hughes S/N 3, Ball Brothers Report dated 28 June 1977.

1.5.22 Orbiter Star Sensor (Flight Unit #2) Post-Vibration Alignment Data - from Hughes Test Procedure TP31639-610, dated 11 November 1977.

1.5.23 P-V Sun Sensor Large Sun Test Report - Hughes Memo No. HS507-4865, dated 17 March 1976.

1.5.24 Pioneer Venus Telemetry Assignments List - Hughes Specification #SS31639-009.

1.5.25 PV Sun Sensor Output Pulse Delay - Hughes Memo No. HS507-4161, dated 12 November 1975.

1.5.26 High Gain Antenna Pointing Control Error Analysis - Hughes Memo No. HS507-5782, dated 13 December 1976.

Section No. 1.5.27
Doc. No. PC-402
Orig. Issue Date 2/28/78
Revision No. _____

Revision

1.5.27 High Gain Antenna Pointing Control Error Analysis - Hughes Memo No. HS507-3339 - Original plus Revisions A through D, dated 31 July 1975 through 17 June 1976.

1.5.28 HGA In-Flight Calibration - Hughes Memo No. HS507-6755, dated 28 June 1977.

1.5.29 Pioneer Venus Spacecraft Time Code - Hughes Memo No. HS507-7155, dated 21 September 1977.

1.5.30 Pioneer Venus Command Timing - Hughes Memo No. 4141.12/42, dated 14 October 1977.

1.5.31 Delay Time in Starting the Command Memory from the 4096 Signal - NASA-Ames Memo No. SVS-6-81: 244-8, dated 24 August 1976.

1.5.32 Data Rate Capabilities of the Pioneer-Venus Orbiter Spacecraft - NASA-Ames Memo No. SVS-5-116:244-8, dated 26 August 1974.

1.5.33 Pioneer Venus Operational Mission Phase and Period Definition - NASA-Ames Memo No. SAM-7-72: 244-8, dated 30 September 1977.

1.5.34 Pioneer Venus Disturbance Torque update (final) - Hughes Memo No. HS507-5910, dated 28 December 1976.

1.5.35 Pioneer-Venus STV Test Report - Thermal Control - Hughes Memo No. HS507-7484, dated 16 February 1978

2.0 SYSTEM DESCRIPTION

System level (overview) descriptions of the nominal mission phases, Orbiter Spacecraft interfaces, and the capabilities and limitations of performance of system functions, are presented below.

2.1 GENERAL DESCRIPTION

Each top-level element of the mission (Orbiter, Launch Vehicle, Ground System, and nominal Mission Plan) is described on a system level below.

2.1.1 Spacecraft Description. The Orbiter System (Figures 2.1.1-1 and 2.1.1-2) consists of the Orbiter Spacecraft and a payload of 12 scientific instruments.

The Orbiter Spacecraft travels intact, as shown in Figure 2.1.1-1, from launch until about 2 days before Orbit Insertion Motor Burn (= VOI), a nominal cruise period of 183 to 202 days. The predominate orientation during the cruise period consists of the spacecraft spin axis (+Z axis) pointing normal to the ecliptic plane and in a northerly direction. Once in orbit, the predominate orientation consists of the spin axis pointing normal to the ecliptic plane and in a southerly direction.

The spacecraft consists of the following subsystems and functions: Mechanical Function (including the Spacecraft Structure), Thermal Function (accomplished by the Structure/Harness Subsystem), Controls Subsystem, Propulsion Subsystem, Data Handling Subsystem, Command Subsystem, Communications Subsystem and Power Subsystem.

The Mechanical features of the Spacecraft can be described by six basic assemblies, as seen in Figure 2.1.1-2: The despun antennas assembly, the BAPTA (Bearing and Power Transfer Assembly), the BAPTA support structure, equipment shelf, substrate (solar array), orbit insertion motor and it's case, and thrust tube. Shape and equipment layout conform to the basic mechanical

requirements of a spin stabilized vehicle. The solar cells on the cylindrical solar panel, antennas orientations, and thrust vector orientations provide efficient power, communications, and maneuverability while the Orbiter is spinning in its cruise attitude.

The thermal design is based on isolating the equipment from the external solar extremes experienced during the mission. (Solar Intensity increases by a factor of 1.98 from Earth to Venus.) Commandable heaters are provided to maintain the orbit insertion motor and safe and arm device within their specified temperatures ranges, to prevent possible freezing of hydrazine monopropellant, and to make up heat balance should there occur an inadvertent trip of non-essential spacecraft loads. Fifteen thermostatically-controlled thermal louvers are mounted on the aft side of the equipment shelf beneath units having high dissipations.

The controls subsystem provides the sensing logic and actuators to accomplish the following stabilization, control and reference functions:

(a) Spin axis attitude determination (via use of slit field-of-view type sun sensors and star sensors), science roll reference signals generation and spin period measurement.
(b) Control of thrusters for spin axis attitude maneuvers, spin speed control, and spacecraft velocity maneuvers.
(c) High gain antenna azimuth despin control and elevation positioning to a desired Earth line-of-sight pointing; additionally, antenna slew control for open-loop tracking of the earth line-of-sight
(d) Magnetometer deployment.
(e) Nutational damping, via use of a partially filled tube of liquid freon E3.

The propulsion subsystem provides the hydrazine monopropellant storage, pressurization,

distribution lines, isolation valves, filtering and thruster assemblies used to accomplish orbiter maneuvers throughout the mission.

The Data Handling Subsystem conditions and integrates into command-selectable (choice of thirteen fixed and one programmable) formats all analog and digital telemetry data (248 assigned channels) originating in the subsystems and science instruments. The selected format of the all-digitized data modulates a 16,384 Hz subcarrier at a command selectable (choice of thirteen rates between 8 and 4096 bps) bit rate. The resulting information is routed to the Communications Subsystem for modulation of the downlink S-band carrier.

The Data Handling Subsystem includes a data memory consisting of two Data Storage Units (DSU), that is intended primarily for use during any occultation. Data is stored or read out at the commanded bit rate. Each DSU has a capacity of 524,288 bits (equivalent to 1024 telemetry minor frames).

The Command Subsystem decodes all commands received via the Communications Subsystem at the fixed rate of 4 bps, and either stores the command for later execution or routes the command in real-time to the addressed destination.

Each of the 381 assigned commands is either completely decoded (discrete-type command) by the Command Subsystem and the execution command generated, or is partially decoded (quantitative-type command) by the Command Subsystem and the command is routed to the addressed destination for final decoding. ("QUANTITATIVE" IS USED THROUGHOUT THIS DOCUMENT TO DESCRIBE BOTH "MULTIFUNCTION" TYPE AND PURELY "QUANTITATIVE" TYPE COMMANDS).

The Communications Subsystem provides radiation reception and transmission capabilities for the command and telemetry information that modulate S-band frequency carriers (assigned in the 2.115 and 2.295 GHz bands).

There are two redundant reception channels; each includes a hemispherically omnidirectional antenna (aft or fwd) that spatially supplements the other to produce total spatial coverage. Optionally by command, the fwd antenna is replaceable by a High Gain antenna or a High Gain Backup antenna.

The S-band downlink is assignable by command to any one of the aft or forward omnidirectional antennas, or to the High Gain or High Gain Backup (directional) antennas. Its frequency may be slaved to the uplink frequency (phase-locked); or, according to circumstances, it may be a multiple of a crystal oscillator located in the channel receiver. The downlink may also be transmitted via any one of, or some pairs of four 10 watt power amplifiers.

There is an additional transmitter in the X-band range (11/3 of the S-band downlink frequency) that is intended for use in occultation measurements.

The power subsystem provides semi-regulated 28 volts ±10 percent to all spacecraft loads (including science instruments).

The primary source of power is the main solar array. When the solar panel output cannot provide adequate power for all spacecraft loads (at low sun angles and during eclipses), the two batteries (each rated at 7.5 amperes full capacity) come on line automatically through the discharge regulators. Battery energy is replenished through a small boost charge array. The power interface unit provides power switching for the propulsion heaters and OIM heaters. It also contains fuses for these circuits and the science instruments input power lines.

Power is distributed on four separate power buses. If a spacecraft over-current condition or under-voltage on either battery occurs, loads are removed to protect the spacecraft from potential catastrophic failure by tripping off buses in the following sequence: Science, switched loads and

transmitter. This leaves only those loads that are absolutely essential to spacecraft survival in a continuously powered ON mode.

The RF transmitters and exciters are on the transmitter bus. Controls and data handling units are on the switched loads bus. Instruments are on the science bus. Command units, OIM heaters and propulsion heaters, power conditioning units, and spacecraft receivers are on the essential bus.

Excitation for the pyro bus is derived from a battery tap located 16 cells (of a total of 24) from the ground reference level.

The bus voltage is limited to 30.0 volts by seven shunt limiters that dissipate all excess solar panel capacity in load resistors mounted on the solar panel substrate and equipment shelves.

2.1.2 Launch Vehicle Description. The space booster vehicle for the Pioneer-Venus mission is the two-stage Atlas SLV-3D/Centaur D-1AR. As shown in Figure 2.1.2-1, the vehicle is comprised of the Atlas booster that includes the interstage adapter, the restartable Centaur, and a nose fairing which encloses both Centaur and the Pioneer-Venus spacecraft. The interstage adapter supports the Centaur atop Atlas and despins with Atlas at separation. The spacecraft is mounted on a payload adapter that mates with the Centaur.

2.1.2.1 Atlas Booster. The Atlas stage-and-a-half concept requires that all five of its engines be ignited before liftoff, while it is held at the launcher, as a reliability measure. This permits an assessment of proper propulsion system operation before commitment to actual flight.

The rated thrust (431,040 lb. total) at sea level consists of 185,000 lb. for each of the two booster engines, 60,000 lb. for the sustainer engine, and 520 lb. for each of the two vernier engines. All five engines use in common the sustainer section tankage that provides RP-1

(liquid aydrocarbon) fuel and liquid oxygen as the propellant components.

From liftoff until separation from Centaur, Atlas is steereu by commands from Centaur. SLV-3D electronics are extensively integrated with the Centaur D-1AR astrionics hardware and software systems. The electronics aspects of the Atlas guidance, flight control, sequencing, and telemetry systems are all provided by Centaur.

The engine angle commands are generated by tne Centaur Digital Computer Unit based upon the guidance equations, derived roll rates, and Atlas Rate Gyro Unit pitch rate and yaw rate inputs. The commands are sent to tne Centaur Servo Inverter Unit and tnen to the Atlas Servo Inverter Unit.

These steering commands are executed by gimballing tne Atlas engines. The booster engines gimbal for pitch, yaw, and roll control. After booster jettison, the sustainer engine gimbals for pitch and yaw control, and the vernier engines swivel for roll control.

The booster section is composed of two nearly identical thrust chambers and a power package consisting of two dual turbo pumps that deliver tne propellants from the sustainer section tankage. After developing a rated I_{sp} of 257 seconds (sea level), the booster engines are cut off (BECO - at 5.7 g), approximately 150 seconds after the vehicle was released from the launch pad. Three seconds later, the booster section is jettisoned. Tne booster section is attached to a thrust ring near the aft end of the tank assembly by 10 pneumatically operated separation latches. Separation is activated by a command from the Centaur guidance system via the Centaur flight control subsystem, and backed up by a staging accelerometer.

The sustainer section houses, within an average 10 foot diameter stainless steel frame, the propellants and tankage. Tne sustainer engine is gimbal mounted at the aft end of the fuel tank.

The vernier engines are gimbal mounted on opposite sides of the tank structure. Liquid oxygen feed lines, tank pressurization lines, and electrical cable fairings are attached to the outside of the tank structure. Equipment pods containing electrical and electronic units are also attached to the outside of the tank structure.

The engines burn from liftoff until propellant depletion occurs, approximately 250 seconds after liftoff. The sustainer engine delivers a rated I_{sp} of 312 seconds (vacuum). The Centaur insulation panels are jettisoned 45 seconds into the sustainer phase (period between booster engine cutoff and vernier engine cutoff [VECO]), followed by nose fairing jettison.

About two seconds after sustainer engine cutoff (SECO), the pyrotechnic system releases the Centaur from the interstage adapter. Separation is achieved by firing the retrorockets mounted on the aft end of the Atlas sustainer tank.

2.1.2.2 Centaur. The Centaur D-1AR Vehicle has some basic redundancy features that the previous D-1A vehicle class did not have.

The main feature of Centaur is the ability to deliver two or more continuous firings of its engines separated by automatically-controlled coast periods.

The Centaur vehicle has a nominal diameter of 10 feet and is 41 feet in length.

This lightweight structure, combined with the high specific impulse (I_{sp} = 439 seconds minimum) of the liquid hydrogen/liquid oxygen main engines, makes it a very efficient vehicle for high energy missions.

Primary thrust is provided by two Pratt and Whitney RL10A-3-3 engines that develop 30,000 lb. total related thrust.

Two identical and separate hydraulic power supply systems provide the force to gimbal the Centaur main engines, one system for each engine.

During coast, separation, and retromaneuvers, attitude control and propellant settling are provided by a hydrogen peroxide-fueled thrust system. This system consists of small engines (6 to 24 lb. thrust) attached to the aft bulkhead.

The Centaur D-1AR astrionics system integrates many former hardware functions into the airborne computer software. Digital autopilot, maneuvering attitude control, sequencing, telemetry formatting, propellant management, plus guidance and navigation are all within the flexible system software scope.

Telemetry data from information sensors is converted to digital words for transmission to the ground station via an S-band transmitter.

The C-band tracking system provides data to determine position and velocity information for use by range safety at the eastern test range.

The Centaur airborne transponder returns an amplified radio-frequency signal when it detects a tracking radar's interrogation.

2.1.2.3 Atlas/Centaur Flight Sequence. The sequence of events described in Sections 2.1.2.1 and 2.1.2.2 are illustrated in Figure 2.1.2.3-1 and summarized chronologically in Table 2.1.2.3-1. The times for the listed events are estimated as of June 1976, taken from Reference 1.5.10. The actual times for most of the events after launch will vary slightly from those shown, based on Atlas/Centaur weight, engines performance, propellant loading, payload weight, launch day, launch time of day and launch azimuth.

TABLE 2.1.2.3-1

PRELIMINARY ATLAS/CENTAUR D-1AR SEQUENCE OF EVENTS
(AS OF JUNE 1976)

EVENT	BASIS	APPROX. TIME FROM LIFTOFF (SECONDS)
Liftoff	2-Inch Vertical motion of Atlas/Centaur	0
Roll Program	Liftoff + 2 seconds	2-15
Booster Engines Cutoff (BECO)	5.7 g	139.2
Booster Package Jettison	BECO + 3.1 seconds	142.3
Jettison Insulation Panels	BECO + 45 seconds	184.2
Jettison Nose Fairing	---	224.2
Sustainer Engine Cutoff (SECO)	Prop. Depletion	252.4
Vernier Engines Cutoff (VECO)		
Atlas/Centaur Separation	SECO + 1.9 sec	254.4
1st Main Engine Start (MES 1) - Centaur	SECO + 11.5 sec	263.9
1st Main Engine Cutoff (MECO 1) - Centaur	Parking Orbit (Guidance)	511.2
2nd Main Engine Start (MES 2) - Centaur	Guidance	1819.3 (Note 1)
2nd Main Engine Cutoff (MECO 2) - Centaur	Hohmann Apogee	1947.4
Centaur/Spacecraft Separation	MECO 2 + Δt (varies)	2107.4
Reorient Centaur to Retro Vector		2132.4
Start Centaur Blowdown		2447.4
End Centaur Blowdown		2697.4

NOTE 1: Coast time is variable between 12 and 18 minutes, depending on launch day, launch time of day, and launch azimuth.

2.1.3 Ground System Description. The Ground System used during the mission for Command, Telemetry and Tracking is described ahead.

2.1.3.1 General. The Pioneer Ground Data System (PGDS) is used to receive, process, record and display spacecraft telemetry and tracking information; and to transmit commands to the spacecraft. Its basic elements are shown in the overview diagram of Figure 2.1.3-1. The PGDS is comprised of the NASA Deep Space Network whose stations are located around the world (refer to Table 2.1.3-1), the Network Control System (NCS) located at JPL-Pasadena, and the Pioneer Mission Operations Control (PMOC) located at the NASA-Ames Research Center - Moffett Field, California.

The Deep Space Stations (DSS) are responsible for the RF Communications link with the Pioneer Spacecraft.

Command and telemetry data are block formatted via standard DSS ground transmission formatting and routed directly between the DSS and the PMOC via Network Control at JPL. Communications is accomplished through the Ground Communications Facility using the High Speed Data lines (HSD) and voice/data lines.

Tracking data is processed at JPL to accurately determine spacecraft present location and predict future flight path, whereas spacecraft attitude determination and prediction is accomplished at PMOC.

Two Sigma 5 Computers are dedicated for on-line real time data processing and display at PMOC. A third Sigma 5 is used for offline specialized processing of data.

The PDP-11 Computer systems at PMOC encode spacecraft commands, messages, and format them for the HSDL.

2.1.3.2 PGDS Telemetry System. RF signals received at a DSS are detected, and the telemetry subcarrier demodulated. The serial bit stream is

synchronized, and the detected bits formatted for HSD lines. When the spacecraft data are convolutiously encoded, the segmential decoding process is employed. Display of selected engineering data for station operations takes place at some sites.

All telemetry data, together with associated DSM status data, are transmitted to PMOC via HSD. The PMOC has the capability to process and display data in real time from one or more DSS and spacecraft simultaneously without interference.

2.1.3.3 PGDS Command System. Command, mode control, and recall data are generated at the PMOC and formatted for transmission via HSD lines to a DSS. Timed commands are transmitted at the time defined, whereas non-timed commands are transmitted in the order defined in the HSD command messages. The DSS generates and formats acknowledge, alarm, configuration, abort, and recall response messages for transmission via HSD to the PMOC.

Initialization of the Command Systems takes place from the NCS, although backup initialization can be accomplished at the DSS in the event of an NCS failure. In order for the NCS to have access to the Command System at the DSS over the same single HSD line which interfaces with the PMOC, a special piece of hardware is required which replaces filler blocks coming from the PMOC with out-going messages from the NCS. There are three filler multiplexers (FMXR), two on-line and a spare. When station handovers require a third HSDL into the PMOC, the spare FMXR is utilized if available. If the spare FMXR is not available, then coordination is necessary between the PMOC and DSN operations control during time periods when both Multiprobe and Orbiter spacecraft are being tracked and a station handover occurs because of the communications switching that is required.

The command message construction, verification, and HSD block formatting functions are performed

by the PDP-11 computer systems located in the PMOC. Response message blocks returning from the DSS are routed to the PDP-11 for verification, and to the Sigma-5 system for post-transmission processing. Mode change and recall request messages are generated by the PDP-11.

2.1.3.4 PGDS Tracking System. The tracking system provides and processes precision radiometric data for determination of the present and future flight path of the spacecraft. This information is used for specific DSS tracking and specific mission planning.

The operating DSS provides measurements to the NCS at JPL of:

(a) Integrated Doppler
(b) Spacecraft range
(c) Differenced range versus integrated Doppler
(d) Antenna pointing control and angle data
(e) Frequency and timing information
(f) Ground weather data measurements, and
(g) Various station status information such as receiver lock status.

The Network Control System at JPL selects, calibrates, and then uses the radiometric data for generation of present corrected spacecraft flight path, predicted spacecraft flight path, and predicted DSS tracking requirements. This information is then routed to cognizant DSS, and mission control at PMOC. The predicts are also compared with incoming data for use in an iterative-correction cycle to increase the accuracy of the predicts.

2.1.4 Mission Description. An overview Mission description and description of significant events are given below.

2.1.4.1 Overview Mission Description. The Orbiter Spacecraft travels from launch (20 May to 2 June 1978) until the Venus Orbit Insertion (VOI) point is reached (point of closest approach to Venus) for a nominal cruise period duration of 183 to

202 days. The predominant orientation during this cruise period consists of the spin axis (spacecraft + Z axis) pointing normal to the ecliptic plane and in a northerly direction (the Multiprobe System spin axis points in a southerly direction during cruise).

For insertion into Venus orbit, the Orbit Insertion motor is fired to produce a thrust generally opposite to the Orbiter velocity vector, thus causing the Orbiter to follow an elliptical orbit about Venus. The orbit is maintained for at least 243 earth days, during which time the 12 scientific instruments collect data that is transmitted back to the earth via the despun high gain antenna.

2.1.4.2 Mission Events Description. The significant mission events within each of the nine adopted Mission time phases are described below.

2.1.4.2.1 Prelaunch Through Despin of the HGA. (L-5 Minutes to L+6 Hours). Approximately five minutes before launch, the Orbiter Spacecraft electrical loads will be switched to internal battery power.

The Orbiter spacecraft will be launched with an Atlas SLV-3D/Centaur D-1AR Launch Vehicle (AC-52) from CCAFS, Florida. The launch will take place during a 1-hour window on one of 10 successive days in the period from 20 May through 2 June 1978. The launch window will place the Orbiter on the desired Type II interplanetary trajectory.

During the launch phase, spacecraft engineering telemetry will be transmitted via low transmitter power from the forward omni antenna at nominally 256 bps rate. Immediately prior to separation, the Centaur will orient the spacecraft in a normal-to-the-ecliptic attitude with the positive spin axis approximately in the direction of the North ecliptic pole (choice of this attitude over Southerly direction represents a net fuel savings because of its proximity to the VOI attitude). The separation switches will initiate redundant command sequences stored in the two command

processor memories that results in spacecraft spin-up to nominally 6.5 rpm.

It is expected that ground station acquisition from the Honeysuckle DSS will occur nominally 1-1/2 hours after launch, at which time the spacecraft bit rate will be increased to 2048 bps nominally, by ground command.

After preliminary attitude and spin rate determination to insure conditions are acceptable for deployment, and adjustment, if required, the magnetometer boom will be deployed by ground command. The spacecraft will then be spun-up to 15 rpm and checked out. The spin axis will be precessed a nominal 90° to align it normal to the ecliptic plane. (This will provide the first opportunity to calibrate an axial jet in pulsed mode in-flight.) The HGA will then be despun and pointed to the earth for HGA usage in the command and telemetry links.

OCPP, OMAG, and OGBD are scheduled to have been turned on before L+6 hours, and, except for OCPP, will operate continuously throughout the in-transit cruise period. OUVS, OEFD, ORPA and OPA will be turned on automatically after spacecraft separation and before initial ground station acquisition; ORPA will then be turned off ≈ 2H30M via ground control.

2.1.4.2.2 Attitude Determination, Star Survey & Cruise. (L+6 Hrs to L+1 Day). The scheme for attitude determination at this position will use at least two detectable stars. The star survey portion of the attitude determination effort may be repeated several times during the transit to Venus for sensor bias calibration.

Use of a 26-Meter (Dish) Ground Station on a four-hours-per-day basis during cruise periods, while in-transit, will be initiated.

2.1.4.2.3 Jet Calibration; TCM #1; Science Checkout; and Cruise. (L+1 Day to L+20 Days). As early as L+1 day, in-flight calibration of the axial and radial thrusters will be performed in preparation

for the first trajectory correction maneuver (TCM #1) at L+5 days. For a known duty cycle (spin rate and pulse width selection) and number of pulsed firings for each required thruster; impulse, and alignment will be determined - for efficient thruster(s) usage during TCM #1.

TCM #1 requires a ΔV of 7.5 m/sec (worst case) to correct launch vehicle injection errors. The maneuver will be performed nominally in the existing normal-to-the-ecliptic attitude via use of the radial thrusters in pulsed mode and, in addition, use of the axial thrusters in pulsed or continuous modes.

spacecraft transmitter high power and a 64 meter ground station will be used during the calibration and during TCM #1, with the cruise status applying before, between, and after these events.

ORPA, OIMS, OETP, ONMS, OIR and ORAD are scheduled to be checked out while the spacecraft is in-transit, starting right after TCM #1. OUVS and OGBD are also checked out right after TCM #1.

An HGA calibration is scheduled for execution after TCM #1.

2.1.4.2.4 <u>TCM #2 and Cruise (L+20 Days to L+60 Days)</u>. Nominally at L+20 days, the second TCM scheduled to correct for residual velocity errors will be performed. TCM #2 will be functionally the same as TCM #1, except the nominal telemetry bit rate will be 1024 bps for the forward omni, high transmitter power, and 64 m DSN usage. Nominally, the ΔV will be small (less than Δ 0.6 m/sec) in magnitude compared to TCM #1 and accomplished in the cruise attitude.

2.1.4.2.5 <u>Cruise (L+20 Days to L+163 Days or L+182 Days (= E -20 Days, where E ≡ Encounter)</u>. Type II transit to Venus, causes the spacecraft to initially travel outside the earth's orbit; reaching a maximum distance of 161 x 10^6 km from the sun before coasting inward towards rendezvous with Venus.

Throughout this long cruise period, the HGA can support 2048 bps telemetry rate, even with use of spacecraft transmitter low power and the 26 m DSN. The highest supportable bit rate via use of either omni drops gradually until, at the end of this cruise period, only 16 bps can be supported, even with use of spacecraft transmitter high power and the 64 m DSN.

The batteries remain in essentially a fully charged state and at a trickle charge rate. As the solar intensity increases, excess solar panel current absorbed by the bus limiters and the general temperature of the spacecraft increase.

2.1.4.2.6 TCM #3 and Cruise (E -20 Days to E -2 Days). Nominally, at E -20 days, where E ≡ Encounter closest approach to Venus, the third TCM scheduled to correct for residual velocity errors will be performed. TCM #3 will be functionally the same as TCM #2, except that the nominal telemetry bit rate will be 16 bps via use of the fwd omni, high transmitter power, and the 64 m DSN; and the nominal ΔV will be $\Delta 0.1$ m/sec.

2.1.4.2.7 Precess to VOI (Venus Orbit Insertion) Attitude Through VOI (E -2 Days to E + ≈12 Minutes). Approximately 2 days before nominal arrival at Venus, spacecraft downlink will be switched to the fwd omni in high transmitter power mode, and ground operation will be switched to the 64 meter DSN.

The orbiter spin axis will be precessed to the orbit insertion attitude, and then the HGA allowed to spinup (via turnoff of DCE motor drive) to the prevailing rotor rate (15 rpm). In this attitude with the positive spin axis directed approximately towards the ecliptic North pole, the sun angle ranges from 85 to 95 degrees w.r.t. (with respect to) the spacecraft +Z axis, and the communications (Earth) angle from 64.5 to 78.5 degrees, w.r.t. the spacecraft +Z axis, depending upon the launch day.

The spacecraft, including the HGA, will then be spunup to 30 rpm (via use of radial thrusters) to

provide acceptable gyroscopic stiffness during OIM burn. (The HGA is not usable in this attitude; it is spinning during OIM burn to keep velocity dispersions low). The HGA will then be despun (via turnon of DCE motor drive) but communications links still maintained via the fwd omni, to avoid contributing errors in a final attitude determination and trim (if required) before OIM burn. The HGA will be spunup to synchronize again with the prevailing rotor rate of 30 rpm prior to OIM burn.

The orbit insertion motor will be fired near the closest approach to Venus (350 to 450 km altitude) to generate an approximate ΔV of 1055 m/sec and place the Orbiter in an elliptical orbit about Venus. Since motor firing occurs behind the planet relative to Earth, the necessary commands will be stored in the command memory. The selected orbit is inclined 105 degrees to the Venus equator, periapsis located at 15 to 32 degrees North latitude, and a 24-hour nominal period.

2.1.4.2.8 Configure Spacecraft and Trim Orbit (E + ≈12 Minutes to E +1-1/2 Days). Twelve minutes after OIM burn, the spacecraft is expected to exit occultation. Reacquisition of the spacecraft communications links via the forward omni and the 64 meter DSN will take place. Tracking will begin to determine the spacecraft orbital parameters, concurrent with an engineering checkout of the spacecraft to insure integrity following OIM burn.

Then, the HGA will again be despun, while the communication links are maintained via the fwd omni, again to avoid contributing errors in a subsequent attitude and spin rate determination effort.

The HGA will first be spunup to the present rotor rate of 30 rpm and the spacecraft rotor plus the HGA will then be despun to 15 rpm (via radial jets firing), in preparation for the large "inversion" maneuver. The inversion maneuver consists of precessing the spacecraft spin axis a

nominal Δ153° so that it is again normal to the ecliptic (but not to the orbital plane) and pointed south (not north, as during the in-transit flight). This maneuver will be performed near the first apoapsis (E + 12 hours nominally) to gain the advantage of ΔV coupling that results in lowering of the periapsis.

After the inversion maneuver, the HGA is again despun, followed by an attitude determination and attitude trim, if required. The HGA electrical boresight axis is pointed at the virtual earth, and the spacecraft communications links are transferred from the fwd omni to the HGA.

Following extensive tracking, the orbit period is trimmed to 24 hours via a ΔV maneuver near the first periapsis (E + 24 hours, nominally).

The Spacecraft is further despun to 5 rpm, the nominal orbital spin rate for science operations. (The preceding set of actions may vary in order and time of occurrance, depending on post-insertion errors).

2.1.4.2.9 In-Orbit Operations (E + 1-1/2 Days to E + 243 Days). (NOTE: Occurances of occultations and eclipses, and eclipse durations given ahead correspond to a May 24 launch date).

The nominal Orbiter mission will last for one Venus sidereal day or 243 earth days. For this duration, the period will be adjusted to remain at approximately 24 hours so that periapsis will remain in view of the same earth tracking station. Solar perturbations, if uncorrected, will cause periapsis altitude to increase. Axial jet firings made near apoapsis will, however, lower periapsis altitude to 150 km periodically during the 243 day mission. Operation of the scientific instruments will be most intense near periapsis, although several instruments will operate throughout the orbit. The despun high gain antenna will provide both uplink and downlink capability throughout the mission. During the first 40 orbits, an occultation experiment will be conducted using both S and X-

band frequencies radiated towards the Earth through the fringe of the Venus atmosphere as the spacecraft passes behind the planet. The 64-meter DSN stations will be utilized for this experiment. Occultations will occur during two periods, near periapsis for the first 81 days of the orbital mission and near apoapsis between E + 157 and E + 166 days. Since the maximum expected periapsis eclipse duration is at most 24 minutes; full spacecraft operations can be maintained using energy supplied by the battery. The apoapsis umbral eclipses that occur later in the mission from E + 173 days to E + 180 days, are up to 3.3 hours in duration; and require that the Orbiter be powered down to a survival mode. Extensive use will be made of the data storage capability and the command memory during orbital operations in order to simplify real-time operations via the ground. The two data storage units will be used to store scientific instrument data recorded during the rapidly changing environment in the vicinity of periapsis, and generally during occulations and high density operations. The 26-meter DSN net will be utilized for data gathering and spacecraft command.

2.2 INTERFACES

Interfaces between the Orbiter spacecraft and each of the launch vehicles, and science instruments are described ahead.

2.2.1

Orbiter/Launch Vehicle Interface. The Orbiter combined integrated system interfaces with the Atlas SLV-3D/Centaur D-1AR, two-stage launch vehicle. To the top of the Centaur mounts a one-piece spacecraft attach fitting which interfaces with the spacecraft. The spacecraft attach fitting remains with the Centaur at Spacecraft separation. Figure 2.2.1-1 shows the Orbiter combined integrated system as encapsulated within the payload fairing (shroud).

2.2.1.1 Mechanical Interfaces

Station and Coordinate System. The relationship of the integrated launch system coordinate system and Station location in the region forward of the Centaur equipment module is shown in Figure 2.2.1.1-1.

Payload Fairing. The basic internal fairing is composed of a 120-inch diameter cylindrical afterbody, 208.33 inches in length, and a 194.74-inch long conical forebody. The allowable spacecraft dynamic envelope provided by the payload fairing is shown in Figure 2.2.1-1. For on-stand access to the spacecraft, three fairing doors will be provided for access to the Orbiter umbilical connector area, battery and pyrotechnic flight plugs, star sensor, and orbit insertion motor safe and arm device. Except for the fairing split-line latches, the fairing provides rf transparency forward of Centaur Station 137.7 because of its fiberglass construction. Extending aft of this Centaur station, a 56-inch long aluminum cylindrical section imposes some rf opacity.

Spacecraft Attach Fitting. The spacecraft attach fitting shown in Figure 2.2.1.1-2 is the intermediate structure between the launch vehicle and the Orbiter. It is 32 inches in height with a top diameter of 44 inches and a bottom diameter of 63 inches. The fitting interfaces the Centaur equipment module forward attach ring at Centaur Station C163.90 or Spacecraft Station 8.0. The spacecraft attach fitting will be vented during ascent through the atmosphere to provide essentially a zero pressure differential. The spacecraft is secured to the attach fitting forward ring before launch by a two-piece Marman-type separation clamp located at Spacecraft Station 40.0. Two pyrotechnic separation bolts located 180 degrees apart which hold the clamp assembly in tension are actuated by signals from the Centaur. Upon actuation of either or both of the explosive bolts, band tension is released permitting the clamp assembly to be pulled down onto the attach fitting. Separation of the

spacecraft from the attach fitting is effected by the relaxation of four springs equally spaced around the perimeter of the forward ring. There are four separation switches located 90 degrees apart, adjacent to each separation spring location, two of which provide separation signals to the Centaur. The spacecraft attach fitting also contains bracketry and nut plate installation of GFE OIM destruct device, accelerometers, pressure transducer, temperature transducer, and associated harnesses.

Aft Actuator Fitting Interface. The spacecraft attach fitting/launch vehicle mechanical interface occurs at the aft actuator fitting interface. Two actuator spring fittings are located 180 degrees apart on the attach fitting. These serve as part of the fairing separation function. Two compressive springs apply a load against these spring fittings to "kick" the lower end of the fairing clear of the launch vehicle. The helper spring fitting shown in Figure 2.2.1.1-2 provides a direct load path to the aft ring.

Electrical Interface Connector Bracket. The spacecraft attach fitting/launch vehicle mechanical interface also occurs at the airborne electrical interface connector bracket. This connector panel provides a break-point for the electrical interface between the Centaur and the spacecraft attach fitting. Two connectors are used for the mild detonating fuse to the OIM destruct unit.

2.2.1.2 Electrical Interface. This interface occurs in the connectors mounted to spacecraft attach fitting electrical interface connector bracket. These connectors pass the firing currents from the Centaur used to actuate the pyrotechnic bolts in the Marman type separation clamp and causes subsequent separation of the spacecraft from the spacecraft attach fitting. Wiring from two separation sensing switches mounted on the spacecraft attach fitting near spacecraft Station 40 pass through these connectors for telmetering by the Centaur. Other circuitry routed through

this interface connector pass through GFE harnessing which terminates in GFE hardware.

Unlike other spacecraft which commonly telemeter information via the Centaur transmission system, the Orbiter spacecraft has no such telemetry interface.

2.2.2 Orbiter/Orbiter Instruments Interface. The Orbiter spacecraft provides the necessary operating environment and means of communication for twelve scientific instruments as described ahead.

2.2.2.1 Mechanical and Thermal Interface. The Orbiter spacecraft accommodates twelve scientific instruments by providing footprint areas and inserts on the equipment shelf as seen in Figure 2.2.2.1-1. All mounting brackets are GFE. The magnetometer boom is supplied by Hughes. The composite fields of view of the instruments are seen in Figures 2.2.2.1-2 and 2.2.2.1-3.

2.2.2.1.1 Retarding Potential Analyzer (ORPA). The ORPA is positioned at $\theta = 259°31'$ on the edge of the equipment shelf. This position along with the pointing direction of 25° to the spacecraft +Z axis, affords an unobstructed 2π steradian FOV.

A cutout is provided in the spacecraft forward barrier for instrument mounting and removal, along with the mating of the electrical connector. For thermal control purposes, an auxilliary thermal blanket extends from the spacecraft forward thermal barrier to the instrument mating ring just under the aperture. The spacecraft ground plane on the instrument to spacecraft thermal blanket is terminated just under the 11" instrument ground plane.

2.2.2.1.2 Ion Mass Spectrometer (OIMS). The requirement for a 2π steradian FOV and a pointing direction along the +Z axis, results in the OIMS being positioned on the outer edge of the equipment shelf at $\theta = 279°16'$. This shelf position is also remote from large spacecraft subsystem units

and, therefore, in a location where the spacecraft magnetic environment is low.

To maintain the instrument at its specified temperature, and to provide a conducting surface for low energy ion measurements, the thermal blanket and conducting surface is extended from the spacecraft forward barrier up to an instrument provided interface just below the aperture.

2.2.2.1.3 Electron Temperature Probe (OETP). The OETP consists of two units, one containing the electronics and a sensor probe and the other a sensor probe only. The electronics sensor probe support extends forward parallel to the spacecraft Z axis through cutouts in the thermal blanket and ground-plane for a distance of 16.5" above the forward barrier. The sensor probe itself is an additional 4" high. By locating this sensor at the outer edge of the shelf, the spacecraft provides this probe a field of view of 2Π steradian with unavoidable blockage due only to the spacecraft mast and high gain antenna. The second sensor is attached to the solar panel and when deployed, extends radially outward 40" normal to the spacecraft Z-axis. Deployment signals are provided from the spacecraft pyrotechnic unit after orbit insertion. The boom release mechanism and squibs are GFE. This sensor is also provided 2Π steradian unblocked field of view. Both sensors are located near θ = 75° of the shelf, which is nearly 180° removed from the magnetometer.

2.2.2.1.4 Ultraviolet Spectrometer (OUVS). The OUVS consists of a single unit mounted on a GFE bracket at the outboard edge of the spacecraft shelf at approximately θ equals 290°. The optical axis of the instrument is parallel to θ equals 300° on the shelf (as are the OCPP and OIR instruments) and at an angle of 60°, with respect to the +Z axis of the spacecraft. A notch is provided in the solar panel forward edge and in the forward thermal barrier for the instrument aperture light shield to protrude through, and provides 60° full width conical clear field of

view. The forward thermal blanket and solar panel substrate blanket attach to this light shield to provide the thermal closure for the instrument.

2.2.2.1.5 Neutral Mass Spectrometer (ONMS). The ONMS is mounted on the outer edge of the equipment shelf at $\theta = 321^{\circ}0'$. This location along with the pointing direction of 27° with respect to the spacecraft +Z axis, provides an unobstructed FOV at 2π steradian within the radius of 36" from the aperture.

For thermal control purposes, the thermal blanket is extended from the forward barrier up to a mating ring under the instrument aperature.

The large power consumption of the instrument (9 watts average) required that it be positioned over a thermal louver and the use of RTV between the instrument base and the equipment shelf.

Pyrotechnic signals are provided by the spacecraft to eject the instrument sealing cap after Venus orbit insertion.

2.2.2.1.6 Cloud Photo Polarimeter (OCPP). The requirement that the OCPP spin scan the planet throughout the Venus orbit necessitated its position on the edge of the equipment shelf and viewing through the upper solar substrate. The travel of the instrument telescope from +43° through +145° with respect to the +Z axis, is through a 3.5" wide slot in the substrate running from just below the equipment shelf through to and including the hat brim. The solar diffuser is attached to the forward barrier directly over the telescope and is positioned to allow stowage of the telescope directed along the +Z axis (i.e., at 0° angle) during launch and when operated in the diffuser looking mode.

The telescope optical axis is positioned on the equipment shelf at $\theta = 306^{\circ}$. Both the OIR and the OUVS instruments are coaligned to the optical axis of the OCPP.

Thermal control of the instrument is afforded by its location under the thermal barrier, and a GFE supplied fiberglass moulding which interfaces between the telescope and the thermal blanket at the inside of the upper solar substrate. The external finish on the moulding is VDA. No thermal blanket is provided at present over the solar diffuser.

2.2.2.1.7 Infrared Radiometer (OIR). The OIR instrument consists of a single unit mounted about 2/3 of the spacecraft radius, outboard from the spacecraft Z axis toward the spacecraft edge, at approximately θ equals 300° on the shelf. The optical axis of the instrument is parallel to θ equals 306° of the shelf, (as are the OCPP and OUVS instruments) and at an angle of 45°, with respect to the +Z axis of the spacecraft. An opening is provided in the thermal barrier for the instrument aperture light shade to protrude through. An unobstructed 5° full width field of view along the optical axis is provided. The thermal blanket attaches to the forward surface of the instrument immediately adjacent to the light shade.

2.2.2.1.8 Magnetometer (OMAG). The magnetometer instrument consists of a shelf mounted electronics unit, and two boom mounted sensors.

The electronics unit is mounted near to the magnetometer boom root at θ = approximately 250°, a position that minimizes the cable length to the boom sensors.

To minimize the spacecraft contributed magnetic fields, the P/T magnetometer sensor and the G sensor are mounted on a three-segment 15'6" deployable boom. This boom length along with a limited magnetic control of all spacecraft materials and units, is required to meet the magnetic specifications at the P/T sensor.

The following are the operating magnetic specification requirements of PC-410 (Reference: Paragraph 1.5.4) and the measured levels

resulting from magnetic testing of the spacecraft (Reference: Paragraph 1.5.15):

(a) DC Magnetic Field

	PC-410 Requirements	Hughes TP32011-206 Measurements
Outboard Sensor (19.5')	(GAMMA)	(GAMMA)
Parallel to Spin	2.0	1.0
Spin Plane	6.0	2.12
Inboard Sensor (14.5')		
Parallel to Spin	4.8	1.67
Spin Plane	14.4	5.0
ORPA Positioned Sensor (1.0')		
Parallel to Spin	None	1.5

(b) AC Magnetic Field

Outboard Sensor (19.5')		
Parallel to Spin	0.1 rms (0.28 p-p)	0.25 p-p
Boom Axis	0.1 rms (0.28 p-p)	0.25 p-p
90 Degrees to Boom and Spin	0.1 rms (0.28 p-p)	0.25 p-p
Inboard Sensor (14.5')		
Parallel to Spin	0.24 rms (0.67 p-p)	0.5 p-p
Boom Axis	0.24 rms (0.67 p-p)	0.5 p-p
90 Degrees to Boom and Spin	0.24 rms (0.67 p-p)	0.5 p-p
ORPA Positioned Sensor (1.0')		
Parallel to Spin	None	3.0

The following are the remanent magnetic field specification requirements of PC-410 (Reference: Paragraph 1.5.4) and the measured levels

resulting from magnetic testing of the spacecraft (Reference: Paragraph 1.5.15):

(a) Remanent Field Magnetized (25 Gauss)

	PC-410 Requirements	Hughes TP32011-206-1 Measurements
Outboard Sensor (19.5°)	(GAMMA)	(GAMMA)
Parallel to Spin	7.0	-1.915
Spin Plane	21.0	3.870
Inboard Sensor (14.5°)		
Parallel to Spin	17.0	-4.441
Spin Plane	51.0	9.354
ORPA Positioned Sensor (2.0°)		
Parallel to Spin (Bz)	None	+104
Radial - (Br)	None	+104
90 Degrees to Radial and Spin (Bφ)	None	+25

(b) Remanent Field Demagnetized (50 Gauss)

Outboard Sensor (19.5°)		
Parallel to Spin	0.5	-0.370
Spin Plane	1.5	0.543
Inboard Sensor (14.5°)		
Parallel to Spin	1.2	-0.943
Spin Plane	3.6	1.330
ORPA Positioned Sensor (2.0°)		
Parallel to Spin (Bz)	None	+20
Radial - (Br)	None	-30
90 Degrees to Radial and Spin (Bφ)	None	-5

The boom is deployed at approximately 2.2 hours after spacecraft launch from the equipment shelf mounted cradle by the initiation of redundant pyrotechnic devices, spacecraft spin, and small kick-springs. The P/T sensor is mounted at the boom end (19°6" from the spacecraft spin axis) and the G sensor is 14°6" from the spin axis. The boom design along with the alignment of all three sensors and the boom, maintain the sensor within the required one-degree half cone angle with respect to the corresponding spacecraft axes during flight.

The temperature of both the P/T sensor and the inboard G sensor is controlled by a multi-layer Kapton blanket around five sides of each unit. The boom mounting flange to sensor interface has a VDA finish. The boom is also thermally wrapped.

2.2.2.1.9 Plasma Analyzer (OPA). The OPA consists of a single unit mounted at the outboard edge of the shelf at approximately θ equals 130°. The instrument outboard looking aperture is provided a 140° vertical by 15° azimuth field of view through notches provided in the solar panel substrate and forward thermal barrier. The centerline of the azimuth field of view is parallel to θ equals 114° 40' of the spacecraft. The centerline of the vertical field of view is perpendicular to the spacecraft Z axis. The forward thermal blanket covers the forward end of the instrument which protrudes through a cutout provided in the blanket support. The solar panel thermal blanket extends just up to the edge of the cutout in the panel for the FOV. A special protective shield provided by the spacecraft is attached to the instrument to prevent the thermal blanket from intruding into the gold-plated area immediately adjacent to the aperture.

2.2.2.1.10 Radar (ORAD). The ORAD consists of two units; the antenna and the electronics. The antenna is located at the extreme edge of the equipment shelf at θ equals 45°. A large notch is provided in the solar panel substrate and the forward thermal barrier to allow the antenna to be

positioned from 0° to 140°, with respect to the spacecraft +Z axis.

The antenna is stowed at 90°, with respect to the Z axis during launch and deployed by redundant pyrotechnic pin pullers after orbit insertion. In this position, the antenna slightly intrudes within the beam of the High Gain Antenna (HGA). The motion of the radar antenna about the pivot point contributes 0.04° to the spacecraft wobble budget (0.2°).

The antenna is thermally isolated from the equipment shelf with insulated mounts. Since this also electrically isolates the antenna, a ground strap is provided for an electrical bond to the spacecraft. A closure is provided around the antenna well sides between the equipment shelf and the forward thermal barrier.

The electronics is located nearby the antenna to minimize the length of cables required. The large power consumption necessitated positioning the electronics over a thermal louver.

2.2.2.1.11 Electrical Field Detector (OEFD). The OEFD consists of a single unit mounted to the spacecraft shelf beneath the thermal barrier at θ equals 175°. Two removal antenna whips extend radially through a hole provided in the solar panel. They extend 24 inches from the solar panel at an angle of 60°, with respect to each other, and in a plane perpendicular to the spacecraft Z axis. During launch, the 4" tip spheres of the antennas are constrained by the launch vehicle shroud. Deployment of these antennas will occur upon shroud separation. The design of the antennas precludes their tips from impacting each other or the solar panel during deployment.

2.2.2.1.12 Gamma Burst Detector (OGBD). The OGBD consists of three units, one containing the electronics and two sensors. One of these sensors is located on the spacecraft shelf at θ equals 203° 49' and the other is located nearly 180° away on the shelf at θ equals 22°30'. Both are located near

the edge of the shelf and thus obtain a minimum combined sensor field of view of 4Π steradians. Cables supplied with the instrument connect the sensors to the electronics located at approximately θ equals ∠60°. All units are contained beneath the forward thermal barrier.

Each of these sensors straddle doublers and as a result, some of the mounting feet are shimmed to obtain equal distance to the equipment shelf. The use of these shims and being mounted to an isolated doubler caused sensor B at θ equals 22°30' to be isolated from the structure. Grounding of the unit is obtained via a ground strap to an adjacent unit. Sensor A obtains chassis ground via ground straps to the doubler to which it is attached.

2.2.2.2 Command Interface. There are 75 discrete commands and 6 quantitative commands allocated for all twelve instruments. Thirteen of the 75 discrete commands are presently unassigned.

2.2.2.3 Data Handling and Timing Signals Interface. Allocation of telemetry channels for all twelve instruments consists of 18 serial digital, 31 analog, and 24 bi-level channels. Assigned channels are 17 serial digital, 24 analog, and 20 bi-level channels.

Additionally, temperature of each of six instruments is telemetered as part of the Structures/Harness "subsystem" (SORPAT, SOIMST, SOEFPT, SONMST, SOPAIT, and SORADT), and the instruments' collective load currents are monitorable via PSCICI.

Timing and status signals made available to each scientific instrument consist of the following:

(a) Word rate
(b) Minor frame rate
(c) Major frame rate
(d) Head clock
(e) Analog read envelope
(f) Digital read envelope
(g) 32,768 Hz clock

(h) 2,048 Hz clock
(i) Bit rate status
(j) Roll Index Pulse (RIP)
(k) 2^{10} P_S Sector pulses
(l) RAM and NADIR pulses
(m) P_S Roll Reference pulse

2.2.2.4 Power and Pyro Interface. The Orbiter Spacecraft provides, via a fused line to each instrument from the 28 VDC science bus, the measured operating power shown in Table 2.2.2.4-1.

Either of the two fully redundant halves of the Orbiter PCU provides the commandable control to fire the squib that releases the MAG BOOM (ORD 14 or ORD A4); or to fire the squib that simultaneously releases the ONMS breakoff nut, deploys the OETP Boom, and unlocks the ORAD mechanism (ORD 15 or ORD A5).

TABLE 2.2.2.4-1

ORBITER SCIENCE INSTRUMENTS
MEASURED OPERATING POWER

INSTR	TURN-ON CURRENT (AMPS)	PEAK CURRENT (AMPS)	FUSE SIZE (AMPS)
ONMS	0.28	0.53	1.50
OIMS	0.05	0.05	0.25
ORPA	0.08	0.08	0.50
ORAD	0.364	1.00	3.00
OUVS	0.58	0.67	0.25
OETP	0.133	0.133	0.5A
OIR	0.180	0.330	0.75A
OPA	0.185	0.185	0.5A
OCPP	0.183	0.410	1.00
OMAG	0.069	0.069	0.50
OEFD	0.022	0.022	0.25
OGBD	0.038	0.080	0.25

2.2.3 Interactions Between Subsystems. The fundamental interfaces between the previously listed Orbiter Subsystems can be classified as mechanical, thermal, and electrical.

The mechanical interface between Orbiter Subsystems is essentially rigid and unchangeable, except for that between the Control Subsystem and the Communications Subsystem. The despin control, via the BAPTA, maintains the HGA, forward omni and HGA backup antennas spatially fixed while the remainder of the spacecraft is spinning to provide gyroscopic stiffness. The HGA Elevation Control portion of the Control Subsystem maintains the HGA elevation slew to the last commanded stepping rate or at the fixed elevation resulting from commanded termination of the stepping rate.

The thermal interface between Orbiter Subsystems, resulting from the primarily passive thermal design that includes nearly all subsystem equipment mounted to a single shelf, tends to be distributive in nature for cold case conditions, and localized in nature for hot case conditions. (Either near-Venus operation, or operation with the sun L.O.S. in an aft direction, is expected to produce a generally hot spacecraft environment). The shelf-mounted equipment are enveloped in a compartment bounded by the forward side of the shelf, inner wall of the solar panel substrate, and the aft side of the forward blanket that isolates the equipment from a space environment.

If the local shelf temperature on the RF half of the equipment shelf drops to +35° ±5°F, and there is sufficient excess of solar panel power, eight heaters that are distributed across the aft side of the RF half of the shelf come ON via closure of a thermostat circuit, providing a uniform heating of the equipment. The same statement is true for the other half (Battery Shelf) of the equipment shelf: Eight heaters distributed across it are controlled by a separate thermostat circuit that closes at +35° ±5°F. Each of the

two thermostat circuits are set to open at +55° ±5°F.

All shelf-mounted units which dissipate more than two to three watts are located over thermostatically-controlled louvered radiator areas. There are 15 thermal louvers (13 general type, and 2 battery type, i.e., used near the spacecraft battery) distributed over the APT side of the shelf, each one controlled by its own locally-mounted thermostat. When the temperature rises above (55° ±5°F for the general type, and 45° ±5°F for the battery type) at a thermostat, the louver begins to open. At (80° ±5°F for the general type, and 70° ±5°F for the battery type), the louver is fully open, radiating heat away from the local hot spot. With such a locally-controlled thermal radiation design, it is seen that the thermal interface between subsystem equipments is minimal in the hot case (equipment from the same subsystem are generally mounted on the shelf in close physical proximity).

All subsystems electrical interfaces, excluding thermal, and including Orbiter Science, are summarized in the interaction matrix, Table 2.2.3-1. The power subsystem interface with other subsystems has been described in Section 2.1.1. The remaining subsystems have numerous interfaces that are primarily electrical.

The numbers in Table 2.2.3-1 refer to the following electrical interface items:

(1) Power (= nominal cruise power)

(2) Commands: ((D) = Discrete;
(Q) = quantitative - <u>excludes redundant channels</u>).

(2a) These are all the unique commands that pass between the Communications Subsystem and the Command Subsystem at 48 bits per command, 4 bps, and on an FSK subcarrier ("0" tone = 100 Hz; "1" tone = 250 Hz).

(2b) These are all the unique commands that pass from one or more Command Subsystem COMs to the first affected subsystem.

(2c) These are all the commands that are processed in the first affected subsystem and pass to the final affected subsystem for terminal action. Most of the time, the first and final affected subsystems are the same.

NOTE: Some single commands can finally affect two or more subsystems, and are counted for each such subsystem, whereas, they are each counted only once in category (2b). Therefore, the total of all (2c) commands from the first affected subsystem does not always equal the quantity of (2b) commands for that subsystem. An example of such a command is: (ORD13 or ORDA3): ARM MAG BOOM, DEPLOYMENT, ONMS B/O MATS AND ORAD RELEASE MECHANISM. This command is indicated as a (2c) command for each of OMAG, OETP, ONMS and ORAD. But it is counted only once as a (2b) command in the first affected subsystem (The Command Subsystem).

The sum total of all (2b) commands does equal the (2a) commands. Again, redundancy in COM assignments is not included. This is a total of unique commands.

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(3) Telemetry (= number of parameters, either analog (AN), serial digital (SD), or Bilevel (BL). - Excludes redundant channels).

(3a) These are all the unique parameters that pass from the source subsystem to one or more Data Handling Subsystem DIMs, either directly, or via an intermediate subsystem. Those in the latter category are listed twice, for the source subsystem (including identification of the intermediate subsystem), and for the intermediate subsystem.

(3b) These are all the unique parameters that pass from the output of the Data Handling Subsystem, via a 16,384 Hz subcarrier, to the Communications Subsystem.

(4) Telemetry timing signals (= number of data timing signal lines supplied directly to the subsystem). Timing signal types include major frame rate, minor frame rate, bit rate status, analog read envelope, digital read envelope, read clock, and word rate.

(5) Attitude timing signals. These timing signal types to the Scientific Instruments include RIP, VPDI (MAM), VPDII (NADIR), Spin Period Sector pulses (1024 pulses per spin period) and Roll Reference Pulse (Fs).

(6) Clock Reference (= number of clock references supplied directly to the subsystem).

TABLE 2.2.3-1. ORBITER SUBSYSTEMS INTERACTION MATRIX

Source Subsystem \ Affected Subsystem		Controls S/S	Propulsion S/S (Includ. OMS)	Data Handling S/S	Command S/S	Communications S/S	Power S/S	Structure & Harness S/S	ORBITER SCIENCE: ORPA	OMS	OETP	OUVS	ONMS	OCPP	OIR	OMAG	OPA	ORAD	OEFD	OGBD
Controls S/S		2c 14(D); 5(Q)	2c 4(D)	3a 16(AN) 3(SD); 2(BL).					1b RIP, I, & Sector			1b RIP, II, & Sector	1b I, & Sector	1b RIP & Sector	1b RIP & Sector	1b RIP & Sector	1b Fs & Sector	1b RIP, II, & Sector		
Propulsion S/S				3a (a) Via Pwr S/Ss: 5(BL) (b) 21(AN) 2(BL)																
Data Handling S/S		4 12		2c 9(D); 1(Q) 3a 2(AN); 1(SD); 4(BL). 4 5	4 9 5 4096 Hz (two lines) 4 1	3b 99 (AN); 29(SD); 79(BL).	4 1	4 1	4 4	4 6	4 3 6 32768 Hz	4 4 6 2048Hz	4 4 6 32768 Hz	4 4 6 32768 Hz	4 4 6 32768 Hz	4 7	4 2	4 5		4 9 6 32768 Hz.
Command S/S		2b 18(D); 5(Q)	2c 2(D)	2b 9(D); 4(Q) 3a 5(SD); 4(BL)	2b 11(D); 2(Q) 2c 7(D); 2(Q)	2b 27(D) (Excludes "Test Call" cmd.)	2b 39(D)		2b 8(D)	2b 8(D)	2b 2(D), 1(Q) 2c 2(D)	2b 4(D); 1(Q)	2b 4(D); 1(Q) 2c 2(D)	2b 5(D); 1(Q)	2b 8(D)	2b 8(D) 2c 2(D)	2b 7(D)	2b 8(D); 1(Q) 2c 2(D)	2b 2(D)	2b 3(D); 1(Q)
Communications S/S				3a 18(AN) 15(BL)	2a 155(D) 17(Q)	2c 27(D)														
Power S/S		1 0.937A 2c 1.336A 10(D)	1 1.336A 2c 10(D)	1 0.429A 3a 17(AN) 23 (BL)	1 0.330A	1 (a) 1.878 A (one Pwr amp ON) (b) 3.453A (Two Pwr amps ON)	1 0.466A 2c 28(D)	1 Zero A	1 ← Total of 1.423A Maximum (@ 40 Watts Maximum) → 2c 1(D)	2c 1(D)	2c 1(D)	2c 1(D)	2c 1(D)	2c 1(D)	2c 1(D)	2c 1(D)	2c 1(D)	2c 1(D)	2c 1(D)	2c 1(D)
Structure/ Harness S/S				3a 18(AN)																
ORBITER SCIENCE	ORPA			3a (a) Via Str. S/S 1(AN) (b) (AN); [illegible]					2c 8(D)											
	OMS			3a (a) Via Str. S/Ss: 1(AN) (b) 2(AN); 2(SD); 1(BL)						2c 9(D)										
	OETP			3a (a) Via Str. S/Ss: 1(AN) (b) 1(AN); 2(SD); 2(BL)							2c 8(D); 1(Q).									

TABLE 2.2.3-1. ORBITER SUBSYSTEMS INTERACTION MATRIX (Continued)

Source Subsystem	Affected Subsystem	Controls S/S	Propulsion S/S (Includ. OMS)	Data Handling S/S	Command S/S	Communications S/S	Power S/S	Structure & Harness S/S	ORBITER SCIENCE: ORPA	GEMS	OETP	OUVS	ONMS	OCPP	OIR	OMAG	OPA	ORAD	OEFD	OGBD
ORBITER SCIENCE (Continued)	OUVS			3a a(AN)₁; 2(SD)₁; 1(BL)₁								2c 2(D)₂; 1(Q)								
	ONMS			3a a)via Str. S/S 1(AN) /b)2(AN)₁; 1(SD)₁; 1(BL)₁									2c 4(D)₁; 1(Q)							
	OCPP			3a 2(SD)₁; 1(BL)₁										2c 5(D)₁; 1(Q)						
	OIR			3a 1(AN)₁; 1(SD)₁; 1(BL)₁											2c 3(D)					
	OMAG			3a a)via Cont. S/S₁ 1(BL)₁ /b)1(AN)₁; 1(SD)₁; 1(BL)₁												2c 6(D)₁				
	OPA			3a a)via Str S/S₁ 1(AN)₁ /b)2(AN)₁; 1(SD)₁; 1(BL)₁													2c 8(D)			
	ORAD			3a a)via Str S/S₁ 1(AN) /b)3(AN)₁; 2 (SD)₁; 6(BL)₁														2c 6(D)₂; 1(Q)		
	OEFD			3a 1(AN)₁; 1(BL)₁															2c 2(D)	
	OGBD			3a 1(AN)₁; 2(SD)₁; 2(BL)₁																2c 3(D)₁; 1(Q)

2.3 CAPABILITIES AND LIMITATIONS FOR THE ORBITER SYSTEM

The nominal spacecraft equipment configuration during the nine mission phases, as well as spacecraft system-level capabilities and limitations, are described below.

2.3.1 Nominal Configurations. Spacecraft equipment configurations for each of the nine nominal mission phases described in Paragraph 2.1.4 is shown in Table 2.3.1-1. Of course, alternate equipment of a redundant pair could be used optionally. Some sequences of actions could be chronologically ordered differently and accomplish essentially the same result. However, this nominal mission sequence and corresponding nominal equipment configurations have been adopted from numerous sources as listed in Section 1.5, or, in some instances, for some net advantage. This has been done for the purpose of communicating what can be expected nominally from the equipment, and act as a reference for discussing what capabilities exist beyond nominal planned usage, and what limitations exist for the equipment as designed.

2.3.2 System Capabilities and Limitations. The capabilities and limitations of system performance in the following disciplines are discussed below.

(a) Commanding
(b) Telemetering
(c) Power Control
(d) Thermal Control and
(e) Maneuver Control

Subsystem level details of these capabilities and limitations are discussed in Sections 3 and 4.

2.3.2.1 Commanding. The maximum field-of-view with respect to the Spacecraft Coordinate system for spacecraft reception of commands is essentially omni-directional because one spacecraft receiver can be connected to one hemispherically directional antenna, while the other receiver is simultaneously connected to another antenna that

TABLE 2.3.1-1. ORBITER NOMINAL EQUIPMENT CONFIGURATIONS DURING THE MISSION

ORBITER PHASE → / STATUS (Within, or at end of Phase)	Prelaunch Thru Despin of HGA	Attitude Determination; Star Survey: & Cruise	Jet Calibration; TCM#1; Sci. Checkout; & Cruise	TCM #2	Cruise	TCM #3 & Cruise	Precess to VOI Attitude Through VOI	Configure S/C & Trim Orbit	In-Orbit Operations
Typical Start Time (Relative)	L-5 Min. (Switch to S/C Battery Pwr) (L=Launch=20 May to 2 June 1978)	L+6 Hr.	~L+1 Day: Jet Cal. ~L+5 Days TCM#1. ~L+5 Days: Start of Periodic Sci. Checkout (Every 3 Days)	L+20 Days	~L+20 Days	E-20 Days (E=Encounter=Closest approach to Venus).	E-2 Days	E+≈12 Min. (Expected exit from occultation)	E+1½ Days.
Typical Phase Duration & Remarks	① Δ6 Hrs. ② Phase includes Δ19° (nominal) precession of S/C +Z axis to ecliptic normal & pointed North.	① Δ3 Hr. for att. Det. and Star Survey (includes Att. Trim (if required) ② Δ18 Hr. overall.	① ~19 Days overall (10 hr. for jet calib.; Δ4 hrs. for TCM #1). ② Nominal ΔV of 7.5 M/sec. ③ 40 Watts max. for all science.	① Δ4 Hr. ② Nominal ΔV of 0.6 m/sec.	① Δ183 Days to Δ202 Days from launch to encounter Δ143 Days to Δ162 Days for this phase.	① Δ18 Days ② Nominal ΔV of 0.1 m/sec.	① ≈Δ2 Days ② Nominal Δ34° precession to VOI attitude.	① ≈Δ2 Days. ② Nom. Δ153° precession to on-orbit attitude (S/C +Z axis pointed South). ③ Nominal Δ24.4 m/sec & Δ13.5 m/sec for 1st periapsis & period adj.	① ≈241½ Days minimum. 2. Nom. Δ8m/sec, Δ1 m/sec, & Δ0.05 m/sec for typical periapsis, apoapsis & attitude corrections.
Nominal Earth-Orbiter Distance	0 to 1 x 10^5 km.	1 x 10^5 to 4 x 10^5 km.	4 x 10^5 to 8 x 10^5 km.	≈8 x 10^6 km.	8 x 10^6 to 33 x 10^6 km.	33 x 10^6 to 53 x 10^6 km.	≈53 x 10^6 km.	≈53 x 10^6 km.	53 x 10^6 to 260 x 10^6 km.
Nominal Sun L. O. S. (W. R. T. Spin Axis)	90°	90°	89° to 90°	89° to 90°	87° to 91°	90° to 91°	85° to 96° (VOI Attitude)	75° to 100° during inversion.	86.5° to 93.5°
Nominal Earth L. O. S. (W. R. T. Spin Axis)	72° to 82°	72° to 82°	72° to 82°	73° to 82°	72° to 87°	87° to 90°	64° to 78.5° (VOI Attitude)	20° to 90° during inversion	86° to 92°
Earth Station(s) in Use:	26 M.		64 M. for TCM #1; 26 M. otherwise.	64 M. for TCM#2; 26 M. otherwise.		64 M for TCM #3; 26 M, otherwise.	To 64 M. prior to switching from HGA to fwd Omni.	64 M.	64 M during occultation experiment; 26 M otherwise.

TABLE 2.3.1-1. ORBITER NOMINAL EQUIPMENT CONFIGURATIONS DURING THE MISSION (Continued)

ORBITER PHASE → / STATUS (Within, or at end of Phase)	Prelaunch Thru Despin of HGA	Attitude Determination; Star Survey; & Cruise	Jet Calibration; TCM#1; Sci. Checkout; & Cruise	TCM #2	Cruise	TCM #3 & Cruise	Precess to VOI Attitude Through VOI	Configure S/C & Trim Orbit	In-Orbit Operations
Communications S/S: Mod Index (Radians)	1.18					① 1.18 (HGA; Lo pwr; 26 m. DSN) ② 0.65 (Fwd Omni; Hi pwr; 64 M. DSN)			
Transmit Antenna	Fwd Omni until after Sep. & Despin of HGA; then HGA.		To HI Gain Backup for ORAD checkout, then return to HGA.				To fwd omni prior to precession.	① To HGA after inversion maneuver. ② To Fwd Omni for despin to 5 rpm, then return to HGA.	To HGA for S-band (and also X-band during occultation experiment).
Receive Antenna	Rcvr 1 to fwd omni, then to HGA as above; Rcvr 2 to aft omni.		Rcvr 1 to H. G. Backup for ORAD checkout, then to HGA.				Rcvr 1 to fwd omni as above; Rcvr 2 to aft omni.	Rcvr 1 to fwd omni or HGA as above; Rcvr 2 to aft omni.	Rcvr 1 to HGA; Rcvr 2 to aft omni.
Exciter	1 ON; 2 OFF. Coh. Mode Enabled.								
Power Amplifier(s)	1 ON; 2, 3, & 4 OFF. (=Lo Pwr)						1 & 3 ON; 2 & 4 OFF. (=Hi Pwr)	Hi Pwr for Fwd Omni use; Lo pwr for HGA use.	Lo Pwr.
States of Transfer Switches	① Lo pwr to despun antennas. ② HGA Selected. ③ Fwd omni selected ④ Exciter 1 selected ⑤ AMP 1 to LO. ⑥ Rcvr. Norm. select. ⑦ AMP 3 Selected.		For ORAD checkout: ② Backup/Fwd select, and ③ Backup select Then return to 1st column status (i.e., HGA).				① Hi pwr to despun antennas ② Backup/fwd omni select. ③ AMP 1 to HI.	① Same as 1st column for HGA Use. ② Same as col. at left for fwd omni use.	① Same as first column

TABLE 2.3.1-1. ORBITER NOMINAL EQUIPMENT CONFIGURATIONS DURING THE MISSION (Continued)

ORBITER PHASE → / STATUS (Within, or at end of Phase)		Prelaunch Thru Despin of HGA	Attitude Determination; Star Survey; & Cruise	Jet Calibration; TCM#1; Sci. Checkout; & Cruise	TCM #2	Cruise	TCM #3 & Cruise	Precess to VOI Attitude Through VOI	Configure S/C & Trim Orbit	In-Orbit Operations
X-Band Transmitter		OFF		ON for HGA Calib.; then OFF						ON for occultation experiment; then OFF.
Controls S/S: Star Sensor Channels		ℓ_1* & ℓ_2* Both ON.	Appropriate threshold & BP selected for one or both, when req'd.							
Sun Sensors		Mid-Range Selected.								
Attitude Data Processor	ON/OFF State	Both ON for auto spinup; then only ADPI ON.								
	Measurement Select	Nominal config. (for spin rate & sun L.O.S. measurement): A: ℓ_1 to ℓ_1 B: ℓ_1 to ℓ_2	Nominal config., then (for star l.O.S. measurement): A: ℓ_1 to ℓ_2 B: ℓ_2 to gated Star B							
	Jet Control	① R1 & R3 fired continuously for spinup to 6.5 rpm (automatic). ② R1 & R3 fired continuously for spinup to 15 rpm (via gnd control). ③ A5 & A6 fired in 128 ms pulses to precess spin axis $\Delta 19^{\circ}$ to ecliptic normal.	For Attitude Trim: (R1 & R4) or (R2 & R3) or A5 or A6 or (A5 & A6) selec. for pulse precession.	For Jet Calib: ① Radials (Pulsed), then; ② Axials (Pulsed) For TCM #1: ① A5 or A6 or A7 continuously for axial ΔV.	① A5 or A6 or A7 contin. for axial ΔV. ② R3 & R4 in 512 ms pulses for radial ΔV.		① A5 or A6 or A7 contin. for axial ΔV. ② R3 & R4 in 128 ms pulses for radial ΔV.	① A5 & A6 in 128 ms pulses for preces. to VOI attitude. ② Spin rate detector inhibited, then R1 & R3 contin. for spinup to 30 rpm.	① Spin rate detector disabled, then R2 & R4 contin. for despin to 15 rpm. ② PLL spin rng. of 8.0 - 17.7 rpm select. then spin rate detector enabled.	① PLL spin range of 4.0 - 8.85 rpm selected, then spin rate detector enabled. ② A6 & A7 continuously for periapsis or apoapsis correction. ③ A5 or A6 or A7 in 128 ms pulses for typical attitude correction.

TABLE 2.3.1-1. ORBITER NOMINAL EQUIPMENT CONFIGURATIONS DURING THE MISSION (Continued)

ORBITER PHASE → / STATUS (Within, or at end of Phase)		Prelaunch Thru Despin of HGA	Attitude Determination; Star Survey; & Cruise	Jet Calibration; TCM#1; Sci. Checkout; & Cruise	TCM #2	Cruise	TCM #3 & Cruise	Precess t VOI Attitu. Through VO	Configure S/C & Trim Orbit	In-Orbit Operations
Attitude Data Processor (Cont'd)	Jet Control (Cont'd)	④ Spin rate detector inhibited until after spinup to 15 rpm; then enabled.		For TCM#1 (Continued) ② R3 & R4 in 512 ms pulses for radial ΔV.				③ PLL spin range of 16.0 - 35.4 rpm selected, then spin rate detector enabled for OIM burn.	③ A5 & A6 in 128 ms pulses for precession to on-orbit attitude. ④ A6 & A7 continuously for periapsis adj. ⑤ A5 continuously for period adj. ⑥ Spin rate detector disabled. then R2 & R4 continuous to despin to 5 rpm.	
	ADP Mode Select	① PLL loss of lock inhibited until after spinup to 15 rpm; then enabled. ② Sun, SRR, norm all selected. ③ Sun gate disabled for initial sun acq., then enabled. ④ SRR adv. inhib. ⑤ Star gate B enl 1 (ℓ*) selected. ⑥ PLL spin range of 8.0 - 17.7 rpm selected after spinup to 15 rpm.		② Sim SRR for ORAD checkout, then back to SRR. ⑥ 4.0 to 8.85 rpm for ORAD checkout; then return to 8.0 - 17.7 rpm.				PLL spin range of 16.0 - 35.4 rpm selected as above.	PLL spin range of 8.0 - 17.7 rpm selected as above.	PLL spin range of 4.0 - 8.85 rpm selected as above.

TABLE 2.3.1-1. ORBITER NOMINAL EQUIPMENT CONFIGURATIONS DURING THE MISSION (Continued)

ORBITER PHASE → / STATUS (Within, or at end of Phase)		Prelaunch Thru Despin of HGA	Attitude Determination; Star Survey; & Cruise	Jet Calibration; TCM#1; Sci. Checkout; & Cruise	TCM #2	Cruise	TCM #3 & Cruise	Precess to VOI Attitude Through VOI	Configure S/C & Trim Orbit	In-Orbit Operations
Attitude Data Processor (Cont'd)	ADP Configure Commands 3 through 11	① JCE buffer output enabled for spinup to 6.5 rpm, then disabled. ② JCE buffer output enabled for spinup to 15 rpm & precession to ecliptic normal attitude, then disabled.	For Attitude Trim: ① JCE enabled. ② PLL spin period magnitude loaded. ③ JCE countdown magnitude loaded. ④ ACS angle magnitude loaded. ⑤ Jets fired; then JCE disabled.	① For Pulsed Firing: Same as for Attitude Trim. ② For Continuous Firing: Same as for Spinup to 6.5 rpm.	Same as for TCM #1.	JCE Disabled.	Same as for TCM #1.	Same as for TCM #1.	Same as for TCM #1.	Same as for TCM #1.
	Spin Rate	① Auto spinup to 6.5 rpm. ② 5.4 after deployment. ③ Commanded spinup to 15 rpm.						Spinup to 30 rpm for VOI burn.	Despin to 15 rpm; then, after orbit & attitude adj., to 5 rpm.	
Despin Control Elect. (DCE)	ON/OFF State	1 ON; 2 OFF.						1 ON; 2 OFF throughout spinups & despin (to retain az. & el. zero rates).	1 ON; 2 OFF throughout spinup & despin (to retain az. & el. zero rates.	1 ON; 2 OFF.
	Motor Drive	1 ON; 2 OFF		Both OFF for ORAD checkout, then 1 ON, 2 OFF.				① Both OFF after precession to VOI. ② 1 ON; 2 OFF after spinup to 30 rpm. ③ Both OFF after final attitude determ.	① Both OFF for any HGA spinup. ② 1 ON; 2 OFF for any despin.	

TABLE 2.3.1-1. ORBITER NOMINAL EQUIPMENT CONFIGURATIONS DURING THE MISSION (Continued)

ORBITER PHASE → / STATUS (Within, or at end of Phase)		Prelaunch Thru Despin of HGA	Attitude Determination; Star Survey; & Cruise	Jet Calibration; TCM#1; Sci. Checkout; & Cruise	TCM #2	Cruise	TCM #3 & Cruise	Precess to VOI Attitude Through VOI	Configure S/C & Trim Orbit	In-Orbit Operations
Despin Control Elect. (DCE) (Cont'd)	DCE Configure Commands 0 Through 5	① HGA Positioner ON for elevation positioning, then O. . . ② HGA electrical boresight despun and pointed to Earth Station. ③ RAM VPD & NADIR VPD reset (not used).		① HGA slewed in el & az for calib. ② HGA spunup for ORAD ckout; then despun as before. ③ VPD used for science ckout as required.				① HGA spunup after precession to VOI attitude. ② HGA despun after spinup to 30 rpm (comm links retained via fwd omni) ③ HGA spunup after final attitude determination & trim (before OIM burn).	① HGA despun after OIM burn. ② HGA spunup after att. det. & before despin of S/C to 15 rpm. ③ HGA despun after inversion maneuver. ④ HGA electrical boresight pointed to earth station after attitude determination. ⑤ HGA spunup for S/C despin to 5 rpm, then despin & pointed to earth station.	① HGA Az & El changed systematically to track earth. ② VPDI & VPDII used as required.
COMMAND S/S : Command Processor:		① 1 & 2 ON. ② Sep. Sw. armed. ③ Spinup seq. is stored in both prelaunch; executed @ L+45 min; then SCL1 & 2 go to Standby.						① OIM burn sequence loaded in CP1 & CP2 both initiated @ [E] -40.5 hrs. ② OIM burn seq. timing updated seq'ly in CP1 & CP2) (Result of latest tracking).		1 Occultation sequences used in both CP1 & CP2 to update HGA Az & El, and to operate science. ② High density sequences used when required.

TABLE 2.3.1-1. ORBITER NOMINAL EQUIPMENT CONFIGURATIONS DURING THE MISSION (Continued)

ORBITER PHASE → / STATUS (Within, or at end of Phase)	Prelaunch Thru Despin of HGA	Attitude Determination; Star Survey; & Cruise	Jet Calibration; TCM#1; Sci. Checkout; & Cruise	TCM #2	Cruise	TCM #3 & Cruise	Precess to VOI Attitude Through VOI	Configure S/C & Trim Orbit	In-Orbit Operations
Command Processor (Continued)							③ OIM fired at E. ④ SCLs later cleared.		
Command Output Modules	1 Through 6 ON.								
Pyrotechnics	PCU Disarmed; then armed & fired for Mag Boom; then disarmed.						Via Cmd memories at E: Arm OIM; fire OIM; then disarm PCU.		Arm; then release ONMS B/O hat, deploy OETP boom, unlock ORAD; then DISARM.
DATA HANDLING S/S: Nominal Bit Rate (bps)	256 until 26M station acquisition of downlink after initial spinup; then 2048 (via HGA).	2048 (HGA or OMNI; Lo pwr; 26 M. DSN).	① 2048 (HGA; Lo pwr; 26 M. DSN). ② 2048 to 1024 (fwd omni; Hi pwr; 64 M. DSN)	① 2048 (HGA; Lo pwr; 26 M. DSN). 2) 1024 (fwd omni; Hi pwr; 64 M. DSN)	(1 2048 (HGA; Lo pwr; 26 M. DSN). ② 1024 to 16 (fwd omni; Hi pwr; 64 M. DSN.)	① 2048(HGA Lo pwr; 26 M. DSN). ② 16 (fwd omni; Hi pwr; 64 M. DSN).	① 2048 (HGA; Lo pwr; 26 M. DSN). 2 To 16 bps after att. det. and before xfer to fwd omni.	(1) 2048 (HGA; Lo pwr; 26 M. DSN). ② 16 (fwd omni; Hi pwr; 64 M. DSN).	(1 2048 (Orbit Day 0) to 64 (Orbit Day 243) (via HGA; Lo xmtr pwr; & 26 M. DSN).
Data Format	Launch/Cruise, then to Orbiter Eng.	① ACS for att. det. & star survey. ② Orb. Eng. otherwise.	① For jet cal. & TCM #1; ACS. ② Orb. Eng. otherwise.	① For TCM #2; ACS. ② Orb. Eng. otherwise.	Orbiter Eng.	① For TCM #3; ACS. 2 Orb. Eng. otherwise.	To ACS for all att. det. & precessions; Orb. Eng. otherwise.	To ACS for all Maneuvers; Orb. Eng. otherwise.	1 ACS for maneuvers. ② Any other format when required.
PCM Encoder	1 ON; 2 OFF								
TM Processor	1 ON; 2 OFF. Subcarrier conv. enc. & data all ON.								

TABLE 2.3.1-1. ORBITER NOMINAL EQUIPMENT CONFIGURATIONS DURING THE MISSION (Continued)

ORBITER PHASE → / STATUS (Within, or at end of Phase)	Prelaunch Thru Despin of HGA	Attitude Determination; Star Survey; & Cruise	Jet Calibration; TCM#1; Sci. Checkout; & Cruise	TCM #2	Cruise	TCM #3 & Cruise	Precess to VOI Attitude Through VOI	Configure S/C & Trim Orbit	In-Orbit Operations
Data Input Modules	All 8 ON.								
Data Storage Units	TBD						TBD		Used during occultations & high-density operations
POWER S/S: Typical Solar Panel Power					205 W (at L+48 Days)				≈ 324 W.
Typical Maximum Batteries D. O. D.	20%	0%							① 31% (periapsis eclipse) ② 3% (periapsis albedo) ③ 60% (apoapsis eclipse)
Batteries Charge/Discharge State (Typical)	① Lo charge rate selected. ② Primary discharge regulator selected.								① Periapsis eclipse season: 4Hr. Max. In Hi; Lo, otherwise. ② Apoapsis eclipse season 10 Hr. Max. In Hi; Lo otherwise.
Protective Circuits	① All 7 limiters enabled. ② UV/OL protec. OFF @ launch; then ON @ Acq. ③ Precharge OFF.								
Science Bus	ON (to power OIR heater).								

TABLE 2.2.1-1. ORBITER NOMINAL EQUIPMENT CONFIGURATIONS DURING THE MISSION (Continued)

ORBITER PHASE → / STATUS (Within, or at end of Phase)	Prelaunch Thru Despin of HGA	Attitude Determination; Star Survey; & Cruise	Jet Calibration; TCM#1; Sci. Checkout; & Cruise	TCM #2	Cruise	TCM #3 & Cruise	Precess to VOI Attitude Through VOI	Configure S/C & Trim Orbit	In-Orbit Operations
PROPULSION S/S: Latch Valves	1 & 2 opened for auto spinup to 6.5 & gnd cntlld. spinup to 15; closed otherwise.	1 & 2 opened prior to jet firings, and closed afterward.	Same	Same	1 & 2 Closed.	1 & 2 opened prior to jet firings and closed afterward.	Same	Same	Same
Heaters	① Primary tanks, fwd & aft jet heaters all ON. ② OIM heater 1 ON; 2 OFF.								① Primary tank htrs ON. ② Fwd & aft axial jet htrs may be OFF. ③ OIM heaters OFF.
Nominal Fuel Usage (lbs.)	① ^0.37 for spinup to 6.5 rpm. ② ^0.58 for spinup to 15 rpm. ③ ^0.18 for precess to ecliptic plane normal.		① ^3.0 for axial ^V. ② ^3.1 for radial ^V.	① ^0.25 for axial ^V. ② ^0.26 for radial ^V.		① ^0.04 for axial ^V. ② ^0.04 for radial ^V.	① ^0.32: precess to VOI attitude. ② ^0.85: spinup to 30 rpm for VOI.	① ^0.82: despin to 15 rpm. ② ^1.25: precess ③ ^8.77: periapsis Adj. 4) ^0.488: Orbit Adj. 5) ^0.54: despin to 5 rpm	① ^2.84:typical periapsis correction. ② ^0.36:typical apoapsis correction. ③ ^0.01:typical attitude correction.
SCIENCE INSTPUMENTS: ORPA	ON@ ≈ L + 40 M OFF@ ≈ L + 2H30M		Checked out periodically, starting after TCM #1.						Operation TBD
OIMS	ON@ ≈ L + 40 M OFF@ ≈ L + 2H30M		"						"
OETP			"						Boom deployed; operation TBD.
OUVS	ON @ ≈ L + 31 M		Checked out after TCM #1.						Operation TBD

TABLE 2.3.1-1. ORBITER NOMINAL EQUIPMENT CONFIGURATIONS DURING THE MISSION (Continued)

ORBITER PHASE → / STATUS (Within, or at end of Phase)	Prelaunch Thru Despin of HGA	Attitude Determination; Star Survey; & Cruise	Jet Calibration; TCM#1; Sci. Checkout; & Cruise	TCM #2	Cruise	TCM #3 & Cruise	Precess to VOI Attitude Through VOI	Configure S/C & Trim Orbit	In-Orbit Operations
ONMS			Checked out periodically, starting after TCM #1.						B/O hat released; operation TBD.
OCPP	ON at L+3H		OFF						Operation TBD.
OIR	Heater ON at launch, then OFF after gnd. acq.		Checked out periodically, starting after TCM #1.						"
OMAG	Boom deployed; then on at ≈ L+2H								"
OPA	ON at ≈ L+50 M								"
ORAD			"						Unlocked; Operation TBD
OEFD	ON at ≈ L+50M								Operation TBD
OGBD	ON at L+5H30M		Checkout after TCM #1.						"

is hemispherically directional in the remaining hemisphere. The two antennas (Fwd Omni and Aft Omni) can be interchanged in their connections to the receivers by command. Additionally, the Forward Omni can be replaced by the HG backup antenna or the HGA. Either replacement provides more gain, but considerably less field-of-view. Both receivers are powered on at all times and cannot be commanded off. Each receiver is also factory set to receive at a fixed S-band frequency carrier.

Spacecraft-to-earth range is a major factor in determining commanding field-of-view, (command threshold is -136.01 dBm at the receive antenna output port, for a BER (bit error rate) of 3.5 x 10^{-6}), but that factor will not be discussed here. Each omni antenna essentially provides approximately 0 dBi gain over most of the hemisphere it sees.

The real-time command rate is fixed at 4 bits per second. Real time commanding, therefore, is limited to a maximum rate of 5 commands per minute (48 bits per command). The maximum stored commanding rate, however, is much faster, at 8 commands per second.

The two command memories onboard the spacecraft can each store 128 commands maximum (including time-delay commands), and can be operated independently, or serially (for a total capacity of 256 unique commands). The two command memories provide not only the capability of rapid commanding, but also commanding when the spacecraft is not in view of an earth station, such as during the launch-to-first ground station acquisition period, and during any occultation.

2.3.2.2 Telemetering. The field-of-view with respect to the spacecraft coordinate system for spacecraft transmission of telemetry depends on the transmitter power amplifier/antenna configuration chosen by command. Any one of the forward omni (provides ≥ -2.5 dB over ±90° w.r.t. the spacecraft +Z axis), the aft omni (provides ≥ -6 dB over ±90° w.r.t. the spacecraft -Z axis), the

HGA (provides > +24 dB over ±15° w.r.t. the HGA boresight axis), or the high gain backup antenna (provides +3.5 to +7.0 dB gain over ±15° w.r.t. the normal to the spacecraft +Z axis), can be selected for telemetry use. The HGA electrical boresight axis can be pointed between 0° and 360° azimuthally, and ±15° w.r.t. the spacecraft X-Y plane. Not only can low (one amplifier) or high (two amplifiers) power transmitting mode be selected for any one of the four antennas, but the AFT omni and any one of the other three antennas can be fed simultaneously (each by one amplifier) in low power mode. Again, spacecraft range is a major factor, but to enhance downlink signal-to-noise ratio, the downlink frequency (S-band) can be phase-locked to the uplink frequency. Alternatively, the downlink transmits a multiple of an on-board crystal oscillator if no uplink is available.

The X-band transmitter transmits a carrier only that is (11)/3 of the S-band downlink carrier frequency. X-band transmission is via the HGA only.

The telemetry data rate is selectable by command (choice of thirteen rates between 8 bps and 4096 bps). The all-digitized telmetry is sampled at various rates based on the format selected (choice of thirteen fixed and one programmable). The programmable format was created for ground test purposes. However, it provides the versatility to select any combination of eight analog, serial digital, or bilevel words maximum (or any single parameter selected eight times), and increase the sampling of the selected parameters to the limit of the selected bit rate.

Telemetry bit rate versus time in the mission is shown in Figures 3.7.3-1 through 3.7.3-3 and is extracted from Reference: Paragraph 1.5.32.

Each of the two telemetry memories (Data Storage Units) onboard the spacecraft can store 524,288 bits of data. They are primarily intended for use during any occultation. Data is stored and read out at the commanded bit rate.

The Orbiter telemetry bit stream phase shift keys (PSK) a 16,384 Hz subcarrier that, in turn, phase-modulates (PM) the downlink S-band carrier.

2.3.2.3 Power Control. The voltage on all four bus lines (essential, science, switched loads, and RF transmitter buses) is limited to 30.0 volts maximum by the bus limiters. When the voltage drops to nominally 27.8 volts, the two batteries (nominal 15 amp hours total capacity) come on line in a load-sharing configuration to support the solar panel. When either battery drops to nominally 27.55 volts, (or the spacecraft load current exceeds nominally 16.25 amps), an under-voltage trip (or an overload trip, respectively) occurs that may dump all loads except the essential bus loads (e.g., the command receivers).

Maximum batteries depth of discharge (DOD) expected for the nominal mission is 60 percent, although the batteries can take a 70 percent DOD with little risk. Most of the cruise portion of the nominal mission is estimated to be flown with the batteries at full charge, necessitating the excess solar panel power to be dumped by the bus limiters nearly continuously into their load resistors. The excess will increase as Venus is approached and the solar intensity factor increases.

The tolerable sun aspect angle range (centered on the optimum 90° Sun aspect angle w.r.t. the spacecraft +Z axis) to maintain power balance (i.e., not discharge the batteries) during the mission depends primarily on load level, solar intensity factor, solar cells temperature, and solar cells degradation. The range increases in magnitude as Venus is approached, as the nominal excess solar panel power increases from 0.6 amp to 5.6 amps for the cruise conditions specified in Table 2.3.1-1.

2.3.2.4 Thermal Control. Thermal control has been designed to be as passive as possible; augmented by onboard thermostatic control (cannot be overridden by command) of thermal louvers, and

shelf heaters (associated with two bus limiters that can be enabled or disabled by command). Onboard thermostatic control also automatically turns off battery charging when battery temperatures exceed 95° ±5°F. Battery charging is automatically resumed when battery temperatures decrease by approximately 10°F. Battery charging can be resumed optionally by ground command sooner (before the battery temperature decreases by 10°F). Radial jet heaters and propulsion tank heaters are on continuously and cannot be commanded off.

All ON/OFF commandable heaters on the Orbiter Spacecraft (fwd axial jet heaters, two aft axial jet heaters and one of two redundant OIM heaters) are expected to be required in an ON state for the cruise portion of the mission, while the spacecraft spin axis is normal to the ecliptic. (After any maneuver that results in the sun impinging on the Fwd or Aft thrusters of the spacecraft, the heater(s) for the illuminated thruster(s), will be commanded OFF). Although only one OIM heater is expected to be required ON during cruise, the redundant OIM heater can be commanded ON also, in the event of a thermal deficiency for the OIM or its S&A device. The OIM heater will be turned OFF after the spacecraft is on-orbit, and the axial jet heaters may or may not be turned OFF.

The only remaining possible thermal control via the ground in the face of a thermally malfunctioning spacecraft are (1) unscheduled ON/OFF cycling of equipment or (2) unscheduled maneuvering to change the sun aspect angle. Either approach can result in a major impact on some or all of the other performance disciplines discussed in Section 2.3.2, and must be considered carefully.

The tolerable sun aspect angle range (centered on the optimum 90° sun aspect angle w.r.t., the spacecraft +Z axis) to maintain thermal balance during the mission depends on the same factors as for power control. However, the range decreases in magnitude as Venus is approached, unlike the

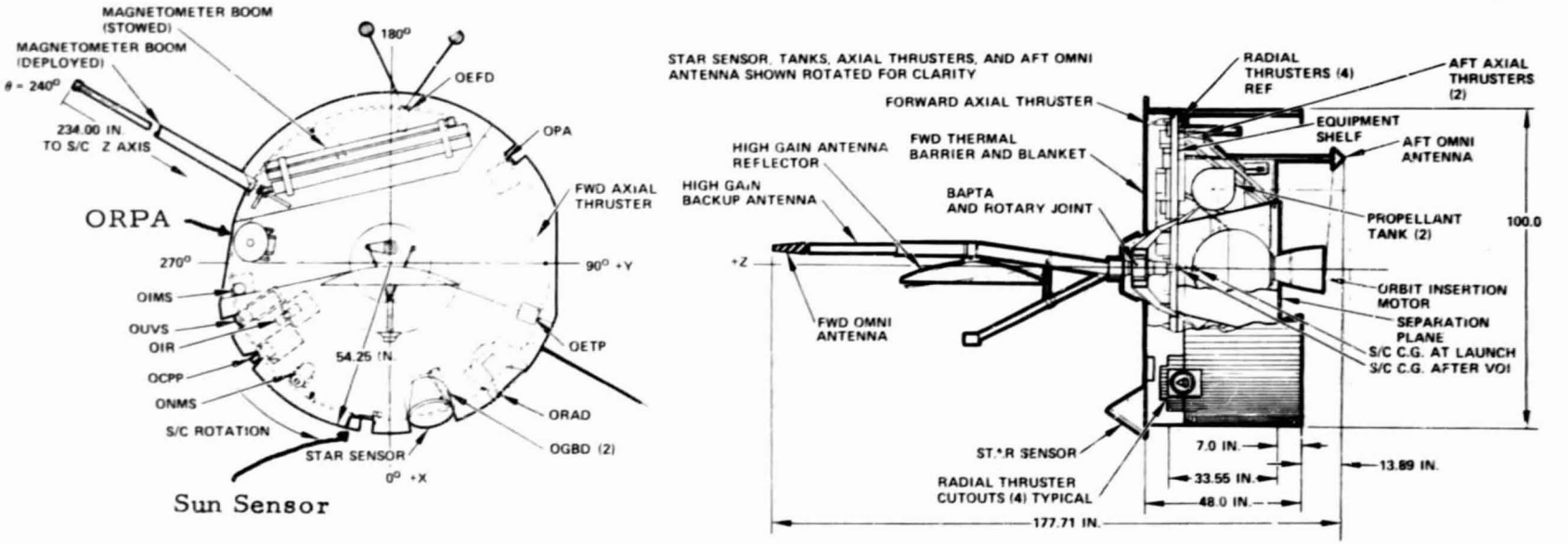
trend for power balance maintenance. The thermal design will accommodate a minimum of 0.3 hour of solar illumination directly on the -Z axis spinning end, or a minimum of 4 hours of solar illumination directly on the +Z axis spinning end.

2.3.2.5 Maneuver Control. Maneuverability is versatile on this spacecraft. Various combinations of jets and their firing timing (128 ms pulsed, 512 ms pulsed, or continous) provide redundant means to generate very accurate precession or translation in any direction. (Sections 3.3, 3.4 and 4.1 provide extensive details). Since the tanks will likely be completely filled at launch (total capacity: 70 lb of hydrazine), and the nominal mission is expected to consume conservatively 55.6 lb, the reserve is substantial.

Maneuver accuracy is not only dependent on resolution of jet firing timing, but also on accuracy of attitude determination. Redundant sun sensors and redundant star sensor channels, each with narrow slit fields-of-view, to provide better than 2.5° accuracy in attitude determination. Each star sensor channel signal threshold and bandwidth is selectable in several discrete steps by command. In the normal mode, the Attitude telemetry measurement system is designed to trigger in response to only one star at a time - the first one encountered azimuthly within a selected azimuth range (11.25° wide Gate B) in each spin period from occurrance of the sun pulse that meets selected threshold and bandwidth levels.

Spin rate control nominally uses redundant combinations of radial jets that fire continuously. The MAG boom is deployed while the spacecraft is nominally spinning 6.5 rpm. The nominal cruise spin rate of 15 rpm is a compromise between keeping attitude changes due to solar torque small, and fuel consumption low during precessions. The spinup to nominally 30 rpm prior to OIM burn is to produce sufficient gyroscopic stiffness to limit insertion attitude errors due to asymmetric mass properties and OIM

thrust misalignment. A despin to 15 rpm following OIM burn is again to compromise between several effects during on-orbit operation, followed by despin to 5 rpm for science operations. This broad spin rate range (5 to 30 rpm) for the nominal mission implies the non-science portion of the spacecraft can perform adequately at any spin rate within the range at any time in the mission, with the exception of the OIM burn cited above.



Equipment shelf (forward surface is S/C station H73.55)

NOTES: (1) Spacecraft Station H8.00 (Atlas - -Centaur station C163.9) is the reference plane.
(2) Star sensor, tanks, axial thrusters and aft omni antenna shown rotated for clarity.
(3) All dimensions are in inches.

Figure 2.1.1-1. Orbiter Configuration and Coordinate System

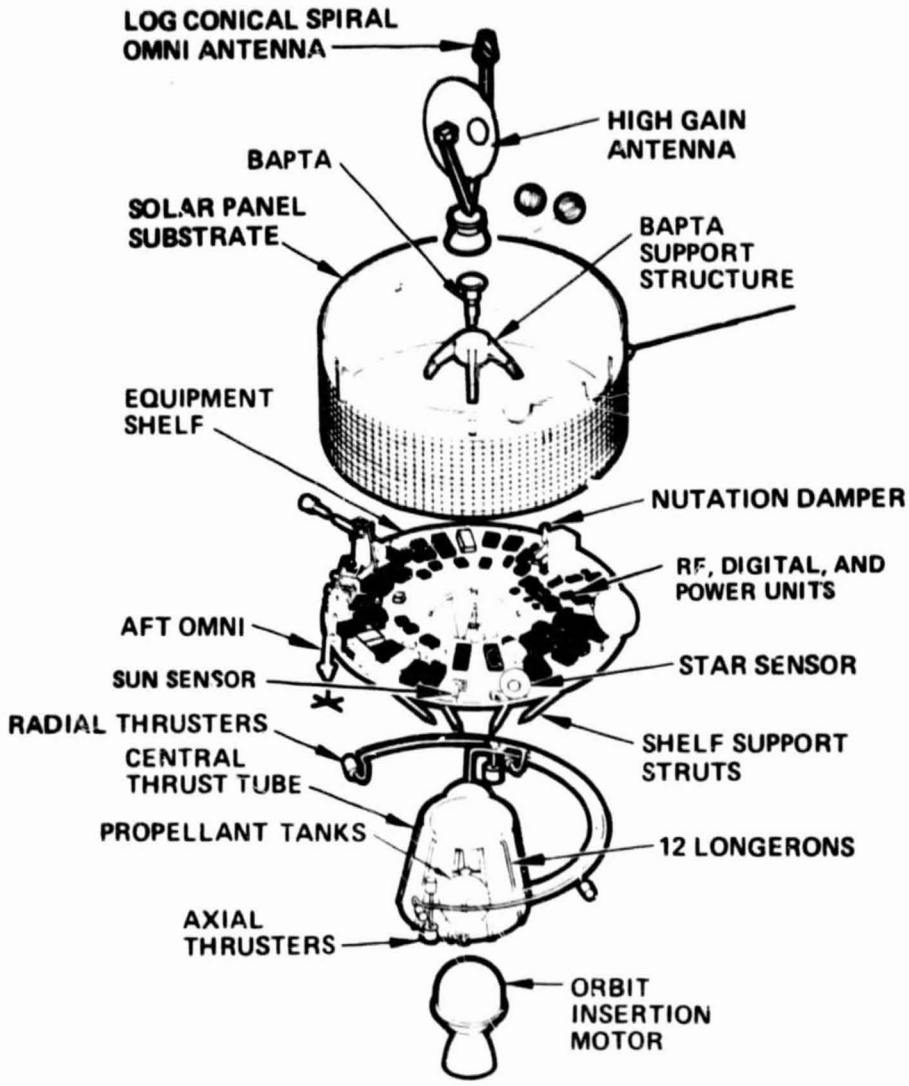


Figure 2.1.1-2. Exploded View of Orbiter

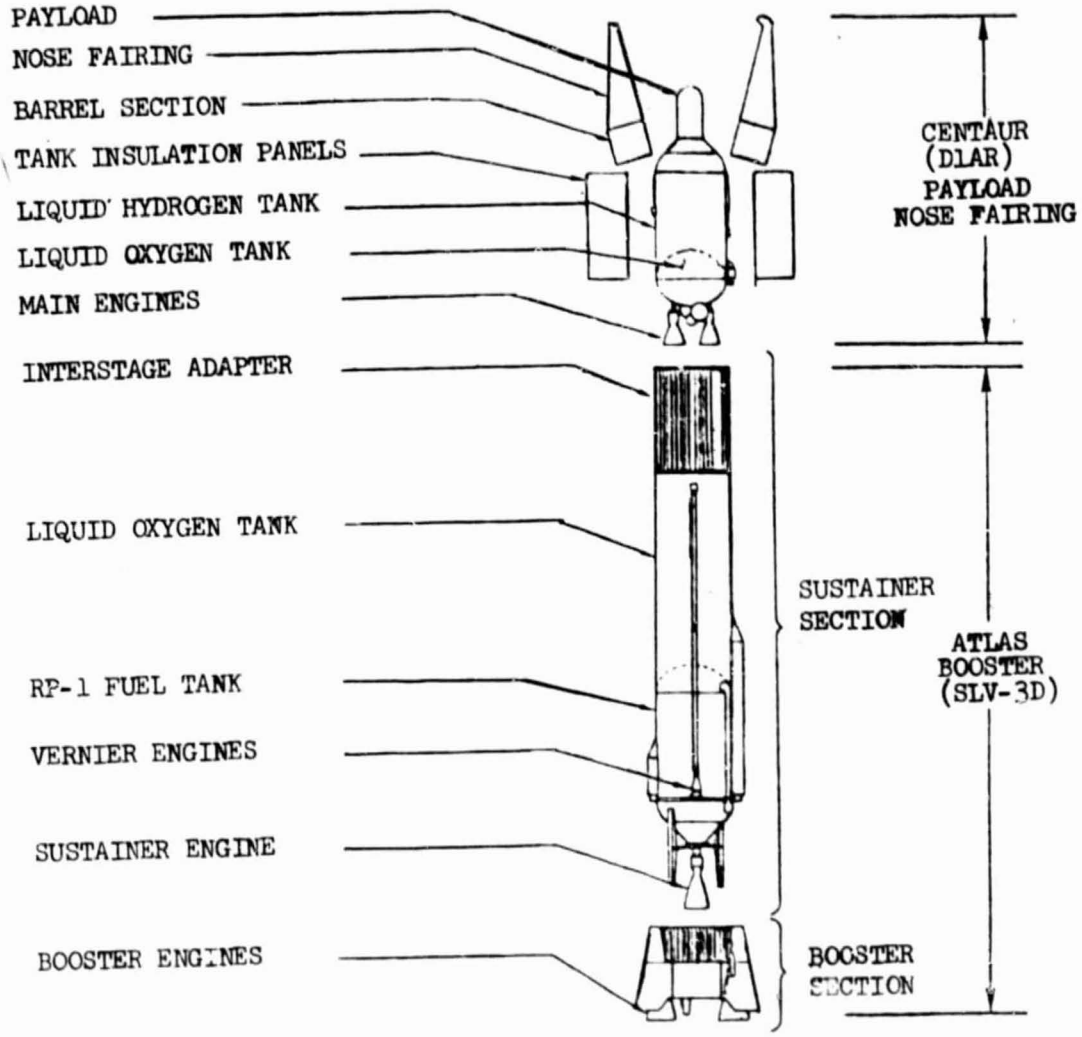


Figure 2.1.2-1. The Atlas Centaur D-1AR Vehicle

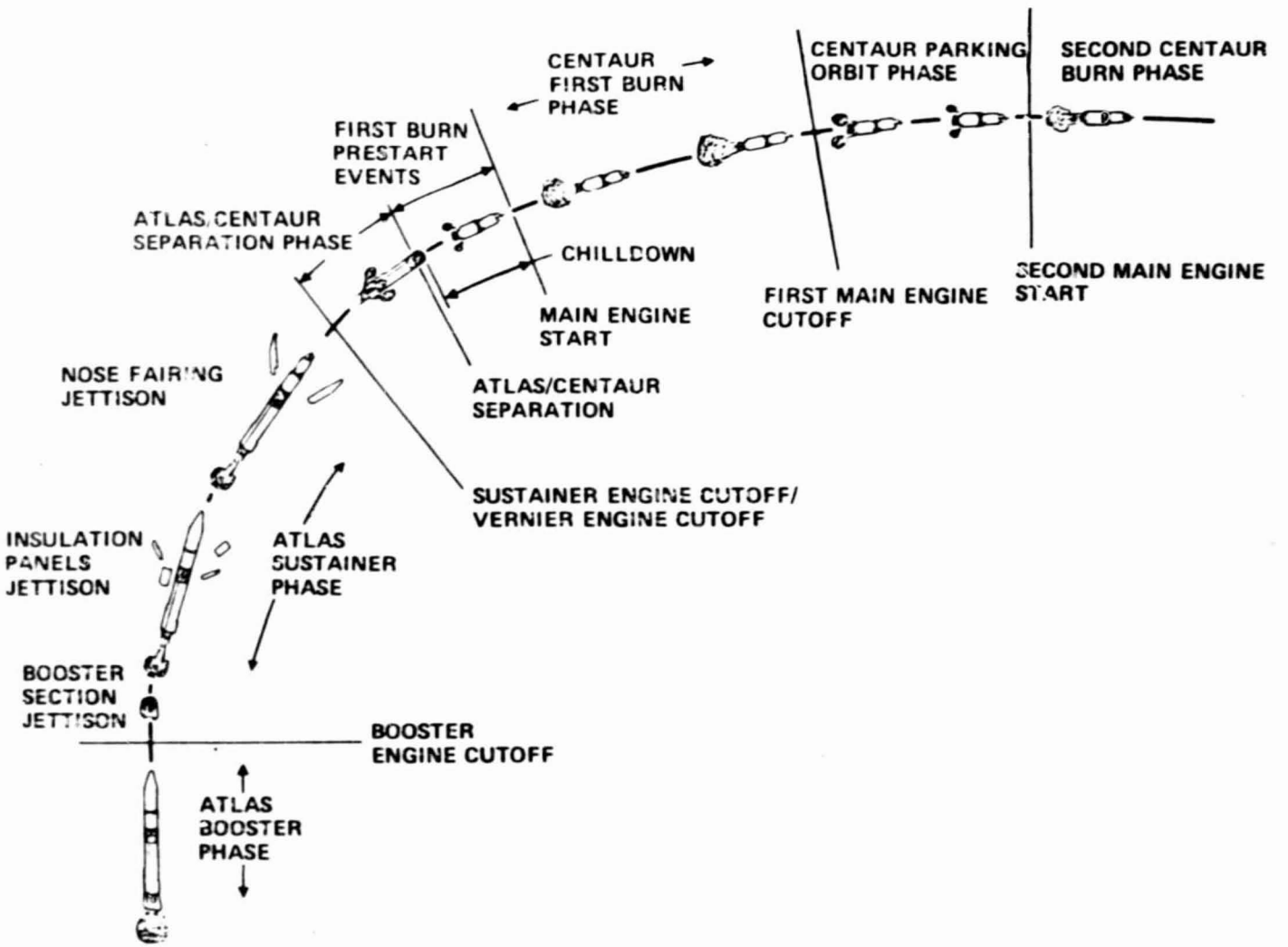


Figure 2.1.2.3-1A. Atlas/Centaur Flight Profile

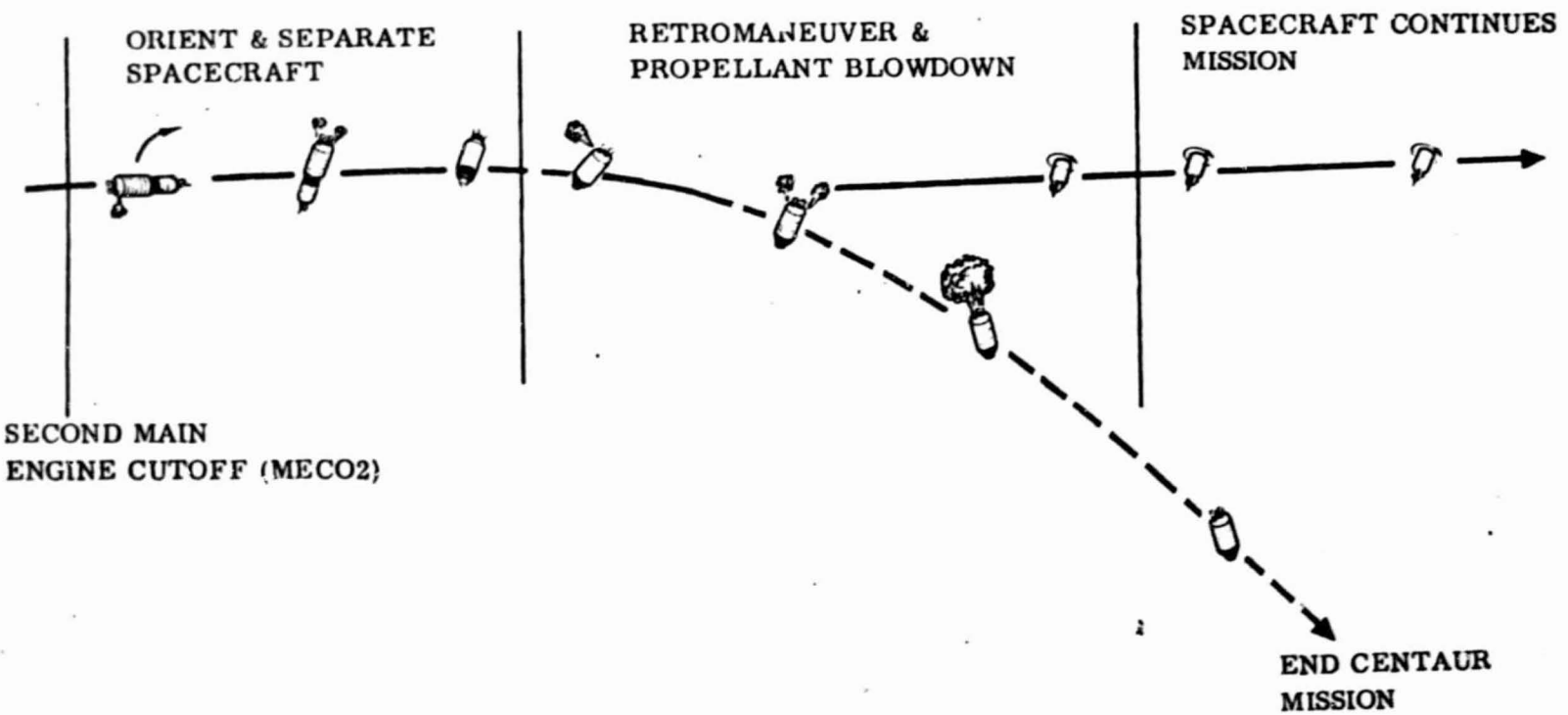


Figure 2.1.2.3-1B. Atlas/Centaur Flight Profile (Continued)

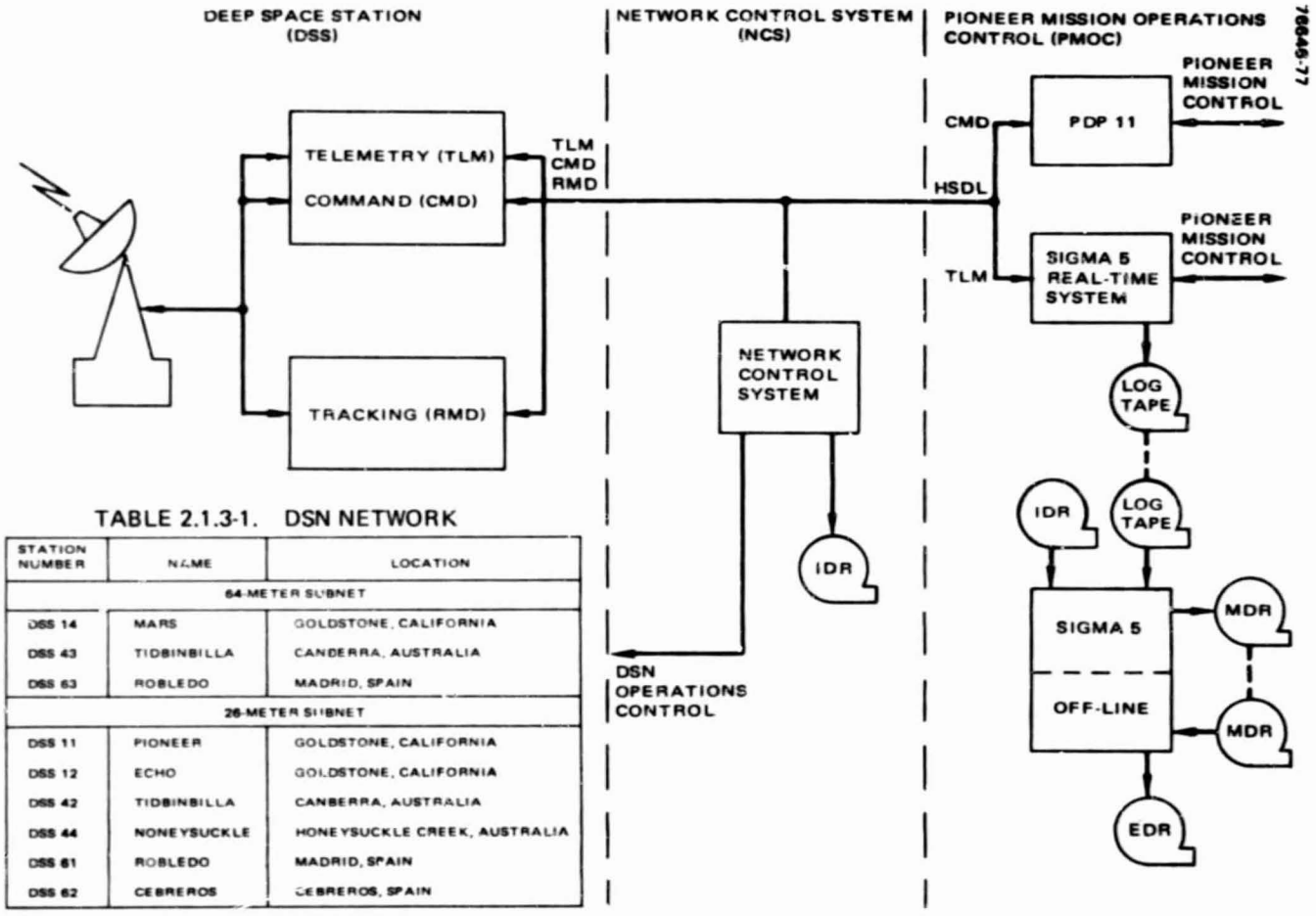


TABLE 2.1.3-1. DSN NETWORK

STATION NUMBER	NAME	LOCATION
64-METER SUBNET		
DSS 14	MARS	GOLDSTONE, CALIFORNIA
DSS 43	TIDBINBILLA	CANBERRA, AUSTRALIA
DSS 63	ROBLEDO	MADRID, SPAIN
26-METER SUBNET		
DSS 11	PIONEER	GOLDSTONE, CALIFORNIA
DSS 12	ECHO	GOLDSTONE, CALIFORNIA
DSS 42	TIDBINBILLA	CANBERRA, AUSTRALIA
DSS 44	HONEYSUCKLE	HONEYSUCKLE CREEK, AUSTRALIA
DSS 61	ROBLEDO	MADRID, SPAIN
DSS 62	CEBREROS	CEBREROS, SPAIN

FIGURE 2.1.3-1. PGDS NETWORK CONFIGURATION (MARK III-74, DIRECT MODE)

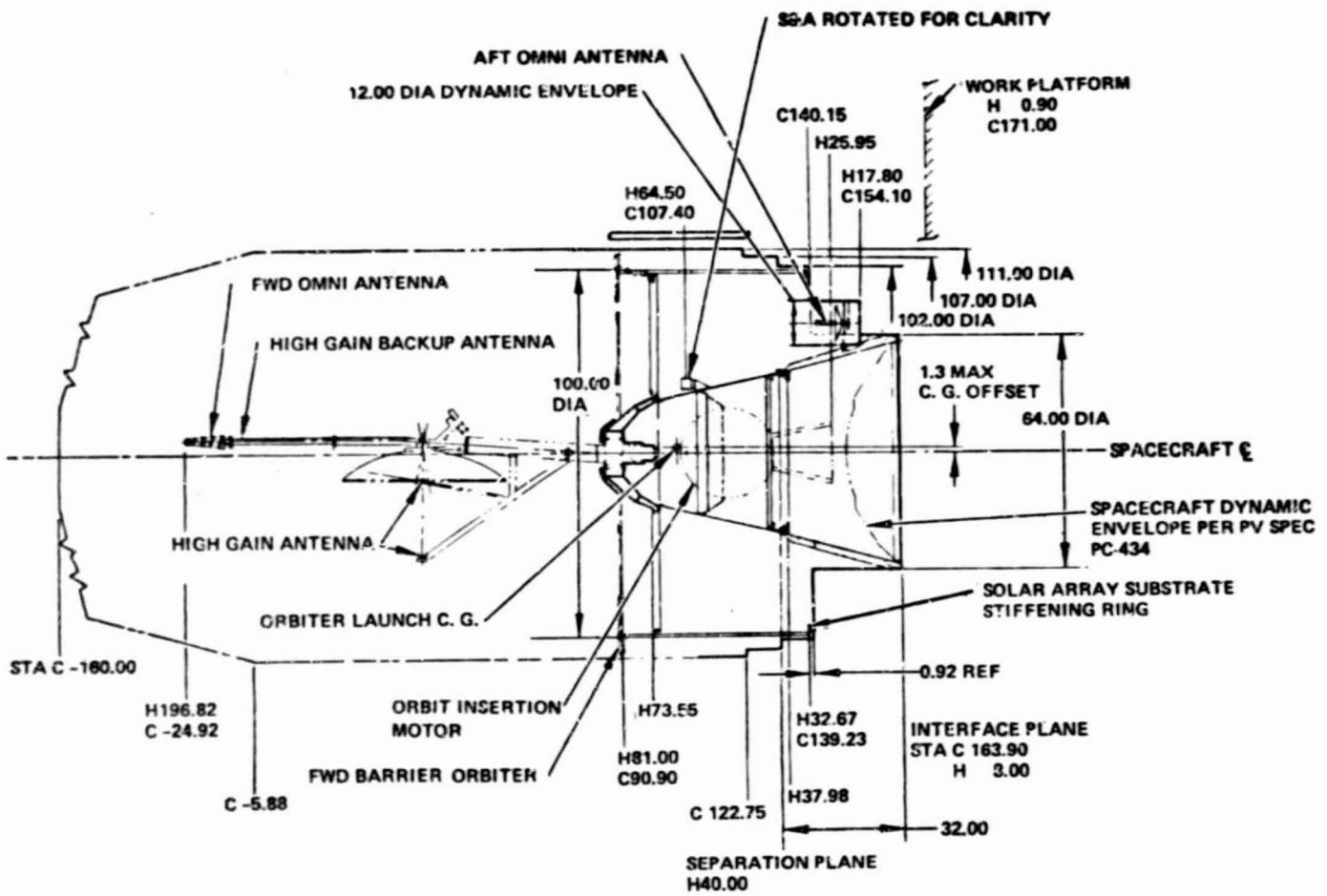


Figure 2.2.1-1. Orbiter Launch Configuration Envelope

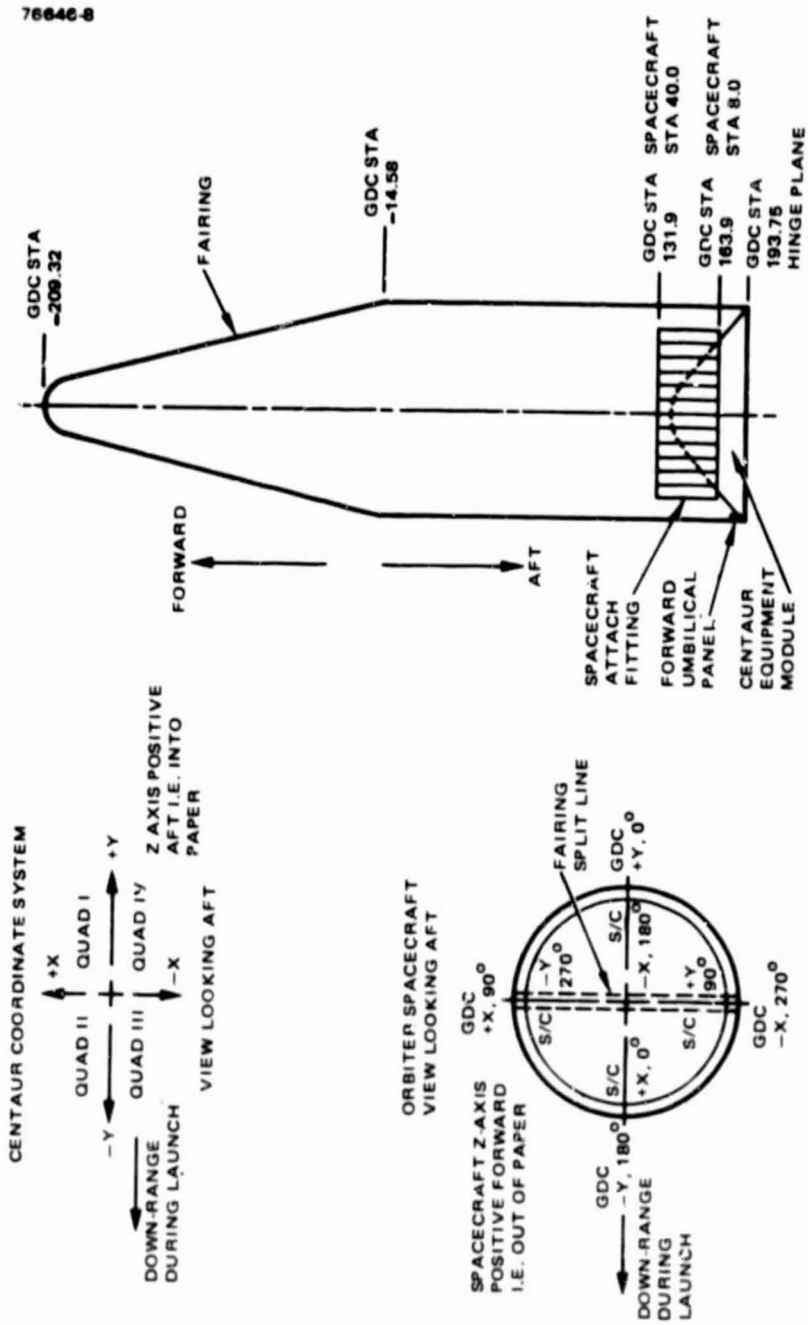


FIGURE 2.2.1.1-1. RELATIONSHIP BETWEEN ORBITER SPACECRAFT AND LAUNCH VEHICLE COORDINATE SYSTEMS

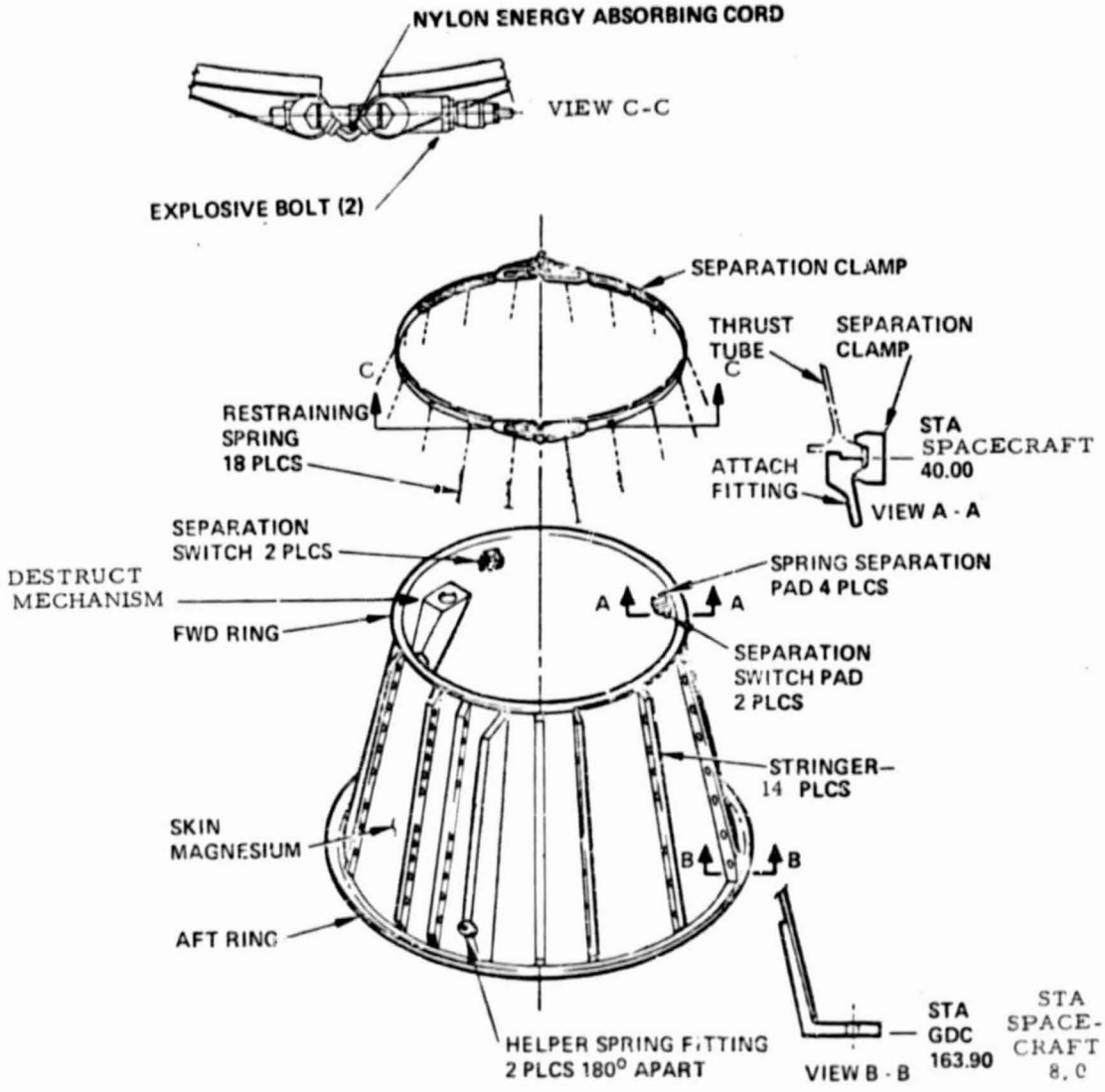


Figure 2.2.1.1-2. Spacecraft Attach Fitting

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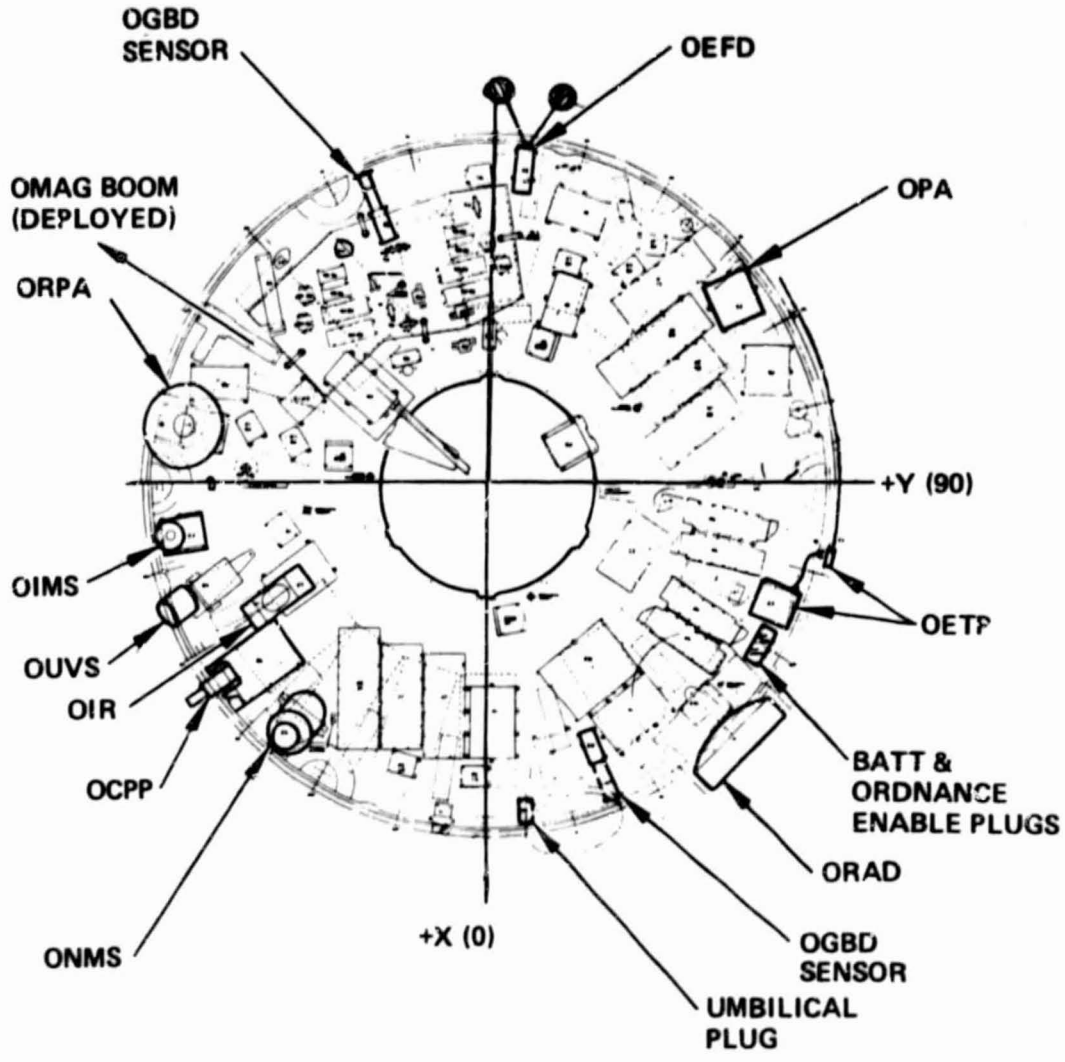


Figure 2.2.2.1-1. Orbiter Instruments Layout on Equipment Shelf

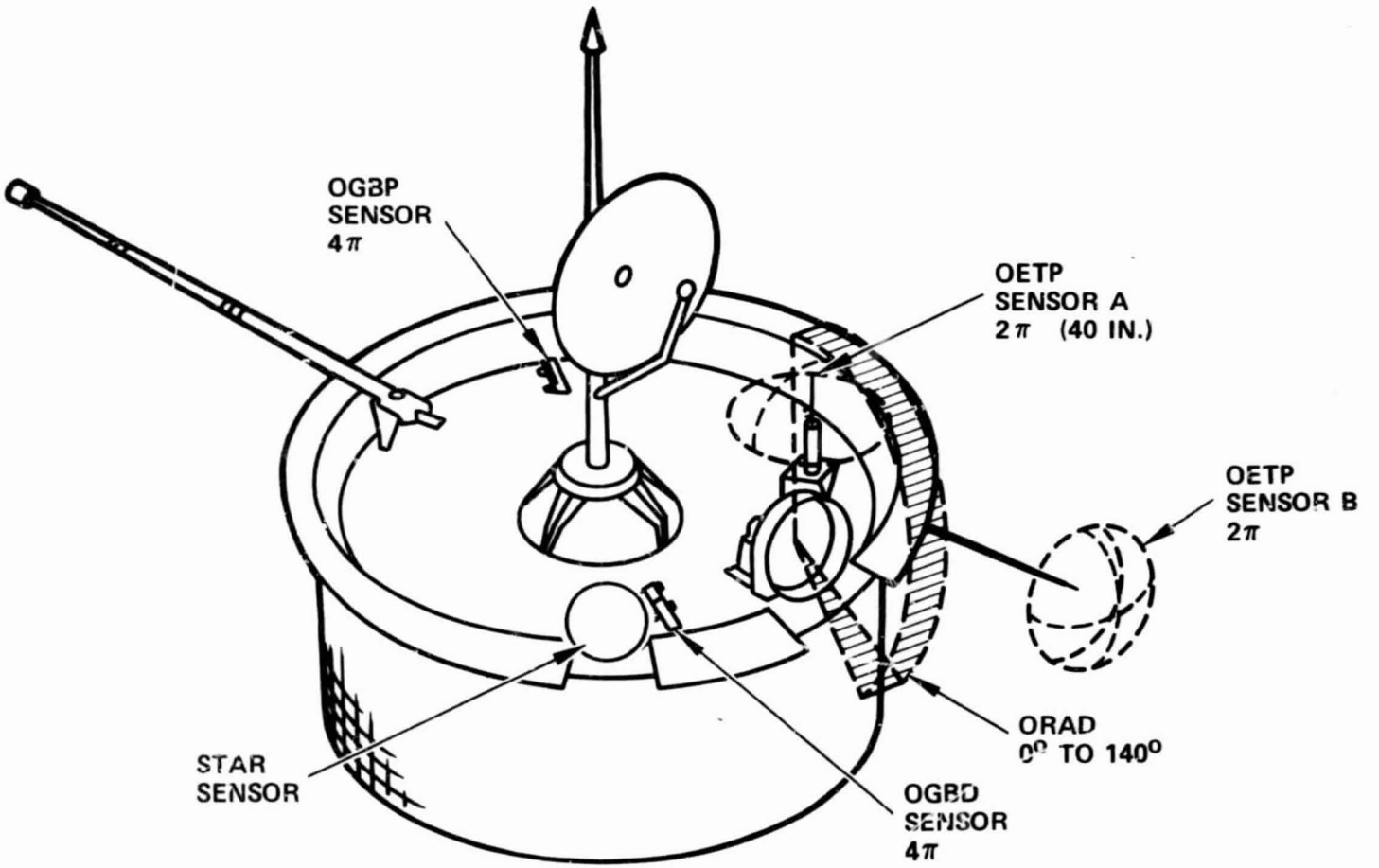


Figure 2.2.2.1-2. Orbiter Instrument FOV's 0 = 0° View

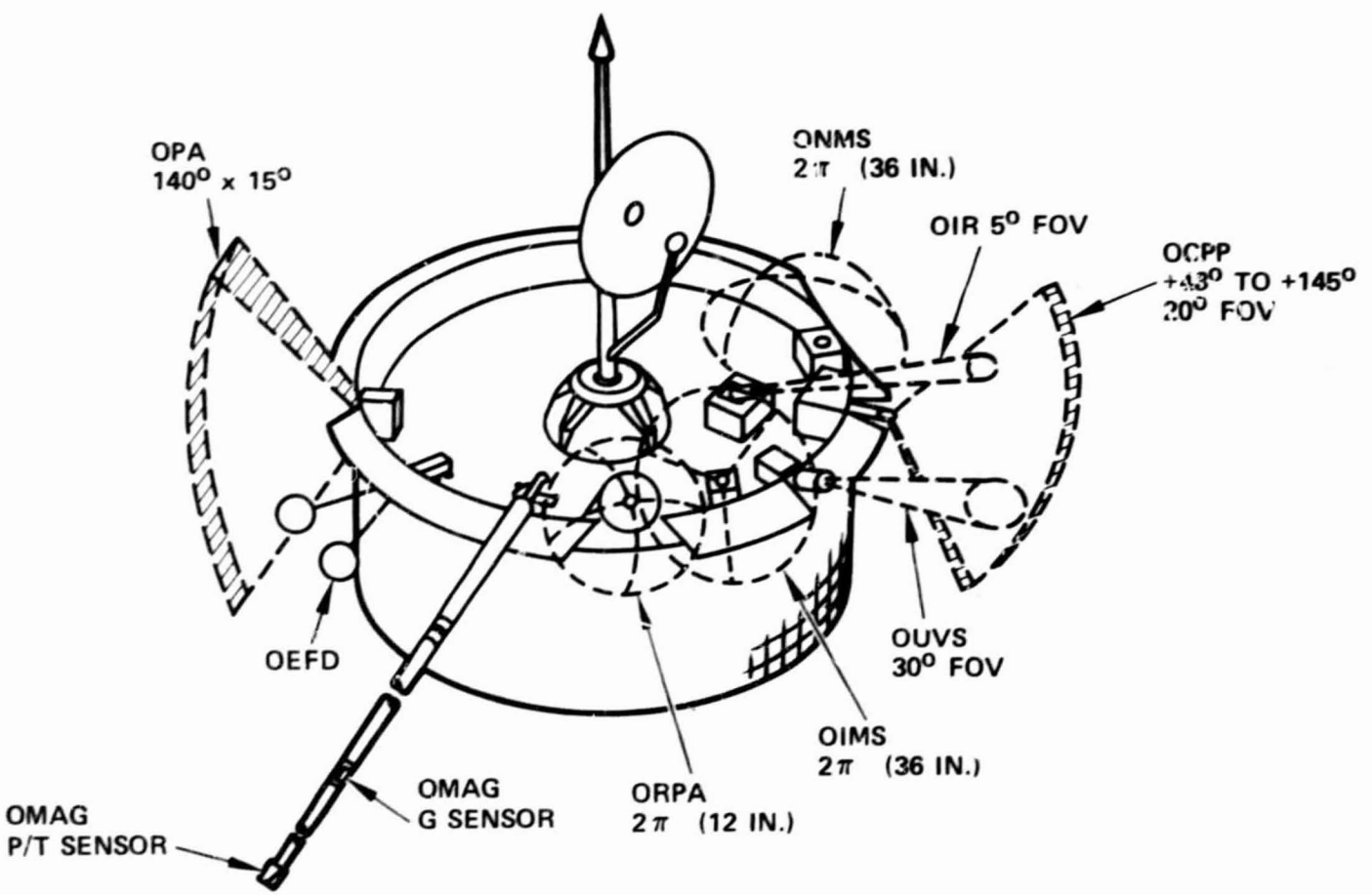


Figure 2.2.2.1-3. Orbiter Instrument FOV's, S/C θ = 240° View

3.0 ORBITER SUBSYSTEMS DESCRIPTIONS

Functional and Operational descriptions of the Orbiter Spacecraft at the subsystem and unit level are presented below.

3.1 MECHANICAL DESIGN

3.1.1 Spacecraft Configuration

3.1.1.1 General Description. The mechanical design of the Orbiter was primarily determined by the use of spin stabilization, the Venusian orbital operations including high data rate transmissions back to earth, the use of solar cells for power generation and the design goal of commonality with the sus spacecraft. Spin stabilization and solar cell generated power led to a cylindrical solar panel with its axis of symmetry coincident with the spacecraft spin axis. The diameter of the cylindrical solar panel was made as large as possible within the limitations of the launch vehicle shroud in order to maximize the roll to pitch moment of inertia ratio and thus improve spacecraft stability. The solar panel is 100 inches in diameter.

The equipment shelf is circular in order to fit inside the solar panel. Spacecraft units and scientific instruments are mounted near the periphery of the shelf for improved view angles and to increase the roll to pitch moment of inertia ratio. Since the majority of the shelf mounted units and the solar panel are mounted near the edge of the shelf, struts are provided to support this outer edge. The struts terminate at the spacecraft thrust tube.

A mechanically despun high gain antenna is provided in order to permit high data rate transmissions during transit and orbital operations without requiring excessive electrical power. A despun antenna requires a despun mast to support the antenna, a bearing and power transfer assembly (BAPTA) to serve as the mechanical and electrical interface between the

spun and despun portions of the Orbiter and pedestal legs to support the BAPTA.

A solid propellant orbit insertion motor is required to retard the Orbiter sufficiently to achieve the desired orbit. The motor must be mounted with its thrust axis coincident with the spin axis to prevent wild gyrations during the motor firing. The orbit insertion motor is mounted inside the open ended thrust tube and is supported by a conical section that attaches to the thrust tube.

The thrust tube is the primary structural element of the Orbiter. All other structural elements are supported either directly or indirectly by the thrust tube. Since the thrust tube is the only structural element that attaches to the launch vehicle, it must react all loads imposed by the launch vehicle.

The requirement for maximum commonality with the Bus does not affect the general mechanical configuration of the Orbiter; however, the size and thickness of certain items such as the thrust tube skins were affected by the commonality requirement.

A scale drawing of the Orbiter is shown in Figure 3.1.1.1-1 (Appendix C). Table 3.1.1.1-1 shows the location and identification of some units.

3.1.1.2 Spacecraft Coordinate System Definition. The spacecraft coordinate system is a right-handed orthogonal set of axes: X, Y and Z. The positive Z axis coincides with the geometric axis of symmetry of the spacecraft structure and is in the direction of the forward omni antenna. The X and Y axes lie in a plane perpendicular to the Z axis at launch vehicle station 171.9. The positive Y axis lies in the plane defined by the Z axis and the shelf split line and will be in the direction of the forward axial thruster. The X axis completes the orthogonal set. The spacecraft attach fitting/launch vehicle interface is parallel to the X-Y plane is located at launch vehicle station 163.9 and spacecraft

station 8.0. The spin of the spacecraft will be a positive rotation about the Z axis; i.e., the angular momentum vector will be in the nominal direction of the positive Z axis which coincides with the nominal launch velocity vector. The spacecraft coordinate system is illustrated in Figure 3.1.1.2-1.

TABLE 3.1.1.1-1

LOCATION AND IDENTIFICATION OF CMD, DATA HANDLING, AND ADP UNITS

NAME	PART NUMBER	S/N	LOCATION (Φ ANGLE; TOP IS CLOSEST TO S/C + Z AXIS)
Comm. Output Module	3280540-100	22	0° Bottom
Comm. Output Module	3280540-100	23	0° Middle
Comm. Output Module	3280540-100	20	268° Middle
Comm. Output Module	3280540-100	24	178° Middle
Comm. Output Module	3280540-100	26	178° Bottom
Comm. Output Module	3280540-100	21	268° Bottom
PCM Encoder	3280620-100	5	120° Bottom
PCM Encoder	3280620-100	7	120° Top
Dual Data Input Module	3280630-100	22	0° Top
Dual Data Input Module	3280630-100	23	268° Top
Dual Data Input Module	3280630-100	24	178° Top
Dual Data Input Module	3280630-100	21	278°
Telemetry Processor	3385034-100	2	140° Bottom
Telemetry Processor	3385034-100	4	140° Top
Data Storage Unit	3385035-100	3	130° Bottom
Data Storage Unit	3385035-100	2	130° Top
Command Processor	3385042-100	2	150° Bottom
Command Processor	3385042-100	3	150° Top
Attitude Data Processor	3385055-100	2	337° Bottom
Attitude Data Processor	3385055-100	3	337° Top

The coordinate system for the high gain antenna (HGA) on the despun mast is a right-handed orthogonal set of axes (X_a, Y_a, Z_a) with the origin located at antenna reflector pivot point. The Z_a axis is parallel to Orbiter spin axis and positive in the direction of spacecraft +Z axis. The X_a axis is coincident with HGA boresight at zero elevation (gimbal) angle and normal to the pivot axis and to Orbiter spacecraft Z axis. The Y_a axis is parallel to pivot axis and completes the system. The HGA coordinate system is illustrated in Figure 3.1.1.1-1. The coordinate system is illustrated in Figure 3.1.1.2-2.

Polarities of offset angles and rates shall follow the right-hand rule. A positive elevation angle would result from the antenna boresight axis being depressed toward the Orbiter shelf. Positive boresight elevation angle rate is obtained by counter-clockwise rotation of antenna reflector about $+Y_a$ axis.

Azimuth offset angles and rates are obtained by rotating HGA coordinate system with respect to the spacecraft coordinate system. Positive polarities would result from counter-clockwise rotations of the mast about the spacecraft +Z axis.

3.1.1.3 Structural Elements

3.1.1.3.1 Thrust Tube. The thrust tube is the primary structural element for reacting the launch and orbit insertion loads. The thrust tube is a truncated conical frustrum with its axis of symmetry coincident with the spacecraft spin axis. The larger end of the trust tube faces aft and serves as the separation plane between the spacecraft and the attach fitting. The forward end of the thrust tube supports the center area of the equipment shelf and the BAPTA pedestal legs.

The thrust tube is a semimonocoque design with magnesium skins, three aluminum rings and twelve machined aluminum longerons. Two of the rings are used as structural interfaces at the ends of

the thrust tube while the third is used to support the orbit insertion motor and part of the propulsion subsystem.

The spacecraft is attached to the attach fitting during launch by a Marman clamp that is shown in Figure 2.2.1.1-2. The "U" shaped segments or shoes of the clamp prevent vertical movement while the mating surfaces of the separation rings prevent lateral movement. The shoes are attached to two steel bands to form two semicircular segments. The two segments are joined together and tensioned by two explosive bolts; detonation of either explosive bolt causes the band to release which allows the spacecraft to separate from the launch vehicle with the aid of kick-off springs. The spacecraft separation is controlled from the launch vehicle. During spacecraft separation from the attach fitting, two separation switches on the spacecraft are actuated which initiate the preprogrammed initial spin-up sequence. This sequence is presented in Section 4.2.1.

3.1.1.3.2 Equipment Shelf. The equipment shelf supports the spacecraft units and scientific instruments. It is a circular honeycomb plate 97.5 inches in diameter with a 2.0 inch thick core and 0.010 inch aluminum facesheets. The center area of the equipment shelf is supported by the forward end of the thrust tube while the outer edge is supported by twelve one-inch diameter beryllium struts that terminate at the intersection of the twelve thrust tube longerons and the separation ring. The shelf is manufactured and assembled to the spacecraft in two semicircular segments. After assembly to the spacecraft, the two shelf halves are spliced together to reduce the vibration levels along the split line to levels comparable to those elsewhere on the shelf. The split line is oriented along the $\theta = 90^{\circ}, 270^{\circ}$ line in spacecraft coordinates.

The layout of spacecraft units and scientific instruments on the forward side of the equipment shelf is shown in Figure 3.1.1.3.2-1 (Appendix C). The aft side of the equipment shelf is used

as a thermal radiator; the locations of the thermal louvers and heaters are shown by dotted lines in this same figure. Mounting points for the spacecraft units and scientific instruments are provided by implanting threaded inserts in the honeycomb core of the shelf with rigid foam.

3.1.1.3.3 Solar Array Substrate. The purposes of the solar array substrate are to support the solar cells that supply spacecraft power and to provide a physical closeout along the periphery of the spacecraft. The substrate is a 100-inch diameter cylinder that is 48 inches long. It is constructed from aluminum honeycomb and fiberglass facesheets. The substrate is aligned with its axis of symmetry coincident with the spacecraft centerline, and it is supported at twenty-four places along the periphery of the equipment shelf.

3.1.1.3.4 Mast Assembly. The purpose of the mast assembly is to support the despun antennas. The interface between the despun mast and the spinning spacecraft body is the BAPTA. Mechanically, the BAPTA consists of two ball bearing races and an electric motor to despin the mast assembly. The BAPTA is discussed more fully in Section 3.3.2.5. The BAPTA is supported above the equipment shelf by four pedestal legs and the mast is attached to the BAPTA. The mast is a tubular structure composed of beryllium tubes and machined aluminum fittings. Since the high gain antenna and its feed assembly are not symmetric with respect to the spacecraft centerline, the lower segment of the mast is canted to bring the center of gravity of the despun assembly close to the spacecraft centerline.

3.1.1.3.5 Magnetometer Boom. The magnetometer boom is a deployable boom designed to place the outboard magnetometer sensors 19.5 feet from the spacecraft centerline. The boom is composed of three tubular graphite segments connected by titanium hinges. The boom is folded and stowed above the forward thermal barrier in two cradle assemblies during launch. Each cradle is held closed by two redundant pyrotechnic pinpullers.

The magnetometer boom is discussed in detail in Section 3.3.2.7. Deployment of the magnetometer boom is discussed in Section 4.1.2.

3.1.1.4 Electrical Harness. All unit electrical interconnections including ground returns are accomplished with the spacecraft harness. Except for the scientific instruments and the data input modules, all wiring is redundant either by unit or by wire; if two units are redundant, a single wire is used for each function of each unit, but if only one unit performs a function, the wires are looped or doubled so that any one wire break does not affect operation. In general, all signal and power grounds are kept separated from inside the unit to the spacecraft single point ground. The only exceptions to this are the transponder, power amplifier, drivers and certain science instruments. Sensitive signal lines and all pyrotechnic lines are shielded to prevent electromagnetic interference, with each shielded line grounded only at one point to avoid ground loops.

The wire harness is constructed from kapton insulated wire varying in size from 16 AWG to 26 AWG. Low current circuits (command, telemetry) use 26 AWG, the smallest size reliably capable of withstanding the mechanical environment. Except for the 16 AWG high current carrying wires, all power lines are 20 AWG for units using Cannon D connectors and 22 AWG for those using Microdot connectors. These are the largest sizes these connectors will take. When the current to be carried exceeds the capability of a single wire, multiple wires are used. The individual wires are bundled and laced together and then tied to the equipment shelf. The harness is constructed in such a manner that minimum stress is applied to the wires. The connectors on the Orbiter are Cannon Golden D's except that Microdot connectors are used where wire density is too great. The harness is composed of five subharnesses - the two equipment shelf harnesses, the pyrotechnic harness, the propulsion harness and the mast harness.

There is an equipment shelf harness on each half of the equipment shelf. It is routed around the thrust tube on the forward side of the shelf; wire bundles radiate outward from this loop as required. Bus power is taken from the solar panel at θ = 115° and 245°.

The pyrotechnic harness carries the firing current to all pyrotechnic devices including the OIM safe and arm device detonators. For safety, all wires are completely shielded in a Faraday cage and only ordnance circuits are allowed in the pyrotechnic harness and connectors.

The propulsion harness connects the propulsion subsystem electrical components to the equipment shelf harness. Since the propulsion units are located on the aft side of the shelf, the propulsion harness is routed through slots near the outer edge of the shelf.

The mast harness connects the HGA positioner and RF switches to the equipment shelf through the BAPTA.

3.1.2 Mass Properties. From launch until the end of the mission, the mass properties of the Orbiter change significantly due to magnetometer boom deployment, propellant usage and orbit insertion motor firing. During the mission, the center of mass moves in the +Z direction due to hydrazine usage and orbit insertion motor firing. The longitudinal position of the center of mass as a function of propellant usage and key mission events is shown in Figure 3.1.2-1 for a nominal mission. After magnetometer boom deployment, the lateral position of the center of mass of the Orbiter is essentially coincident with the Z axis.

The roll moment of inertia increases when the magnetometer boom is extended and decreases slightly during the mission due to hydrazine usage for orbit insertion motor firing. The change in roll moment of inertia as a function of propellant usage and key mission events is shown in Figure 3.1.2-2 for a nominal mission. The

minimum and maximum lateral moments of inertia change in the same manner as the roll moment of inertia. The changes in minimum and maximum lateral moments of insertia are shown in Figure 3.1.2-2.

The details of the changes in mass, center of mass and moments and products of inertia throughout the nominal Orbiter mission are given in Table 3.1.2-3.

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TABLE 3.1.2-1

SEQUENCED MASS PROPERTIES DATA

Code	Description	Current Weight (Pounds)	Center of Gravity (In. From Ref. Datum) Z	X	Y	Moment of Inertia (Slug-Ft. Square) IZ	IX	IY	Products of Inertia (Slug-Ft. Square) IZX	IZY	IXY
3000	Orbiter EOI - AP JETT	753.11	72.2	-0.0	0.0	233.9	189.4	147.7	-0.68	0.19	13.67
	Spun Section	710.19	69.4	0.0	0.0	233.1	161.5	119.4	0.0	0.00	13.69
	Despun Section	42.92	117.8	-0.6	0.3	0.8	7.4	7.8	-0.43	0.05	-0.01
3350	N2H4 Increment	59.23	60.8	0.0	0.0	8.9	8.3	0.6	0.0	0.0	-2.22
3380	ONMS CAP-Jettisoned	0.82	85.0	36.2	-29.0	0.0	0.0	0.0	0.0	0.0	0.0
3400	Post VOIM Burn*	813.16	71.3	0.0	-0.0	243.4	189.4	150.1	-0.60	0.12	11.27
	Spun Section	770.24	68.8	0.0	-0.0	242.4	170.9	121.1	0.12	-0.08	11.28
	Despun Section	42.92	117.8	-0.6	0.3	0.8	7.4	7.8	-0.43	0.05	-0.01
3500	VOIM Expendables	397.91	55.0	0.0	0.0	5.9	5.2	5.2	0.0	0.0	0.0
3600	VOIM Ignition	1,211.07	66.0	0.0	-0.0	249.1	220.0	170.7	-0.59	0.12	11.27
	Spun Section	1,168.15	64.1	0.0	-0.0	248.3	186.8	137.1	0.15	-0.10	11.28
	Despun Section	42.92	117.8	-0.6	0.3	0.8	7.4	7.8	-0.43	0.05	-0.01
3750	N2H4 Increment	9.00	61.0	0.0	0.0	0.9	0.9	0.1	0.0	0.0	-0.24
3800	Magnetometer Extended*	1,220.07	65.9	0.0	-0.0	250.0	220.4	179.4	-0.59	0.12	11.03
	Spun Section	1,177.15	64.1	0.0	-0.0	249.2	187.7	137.1	0.15	-0.10	11.05
	Despun Section	42.92	117.8	-0.6	0.3	0.8	7.4	7.8	-0.43	0.05	-0.01
3950	N2H4 Increment	0.27	61.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	-0.01
4000	Orbiter - Separation	1,220.34	65.9	0.3	0.9	211.4	190.1	163.1	0.70	4.74	-6.69
	Spun Section	1,177.42	64.1	0.3	0.9	210.6	156.8	129.4	1.56	4.96	-6.68
	Despun Section	42.92	117.8	-0.6	0.3	0.8	7.4	7.8	-0.43	0.05	-0.01
4100	Attach Fitting	59.12	26.0	0.2	0.2	9.7	6.9	7.5	0.01	0.04	0.06
4200	Orbiter - Launch	1,279.46	64.1	0.3	0.8	220.6	215.9	190.8	0.72	5.09	-6.48
4250	Science Contingency	4.62	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
4300	S/C Contingency	24.54	0.0	0.0	0.0	6.0	0.0	0.0	0.0	0.0	0.0
4400	TOTAL LAUNCH WEIGHT	1,308.62									

*CG Affects Jet Alignment.

(Table Continued)

TABLE 3.1.2-1 (Continued)

Code	Description	Current Weight (Pounds)	Moment of Inertia (Slug-Ft. Square) IZ	IP (Max)	IP (Min)	PHI (Max)	IZ/IP (Max)	IZ/IP (Min)	IR/IP (Eff.)
3000	Orbiter EOL-CAP JETT	753.11	233.9	193.5	143.6	163.4			
	Spun Section	710.19	233.1	165.6	115.3	163.5	1.205	1.624	1.358
	Despun Section	42.92	0.8	7.8	7.4	88.3			
3350	N2H4 Increment	59.23	8.9	8.9	0.0	15.0			
3380	ONMS CAP-Jettisoned	0.82	0.0	0.0	0.0	0.0			
3400	Post-VOIM Burn	813.16	243.2	201.8	147.6	167.7			
	Spun Section	770.24	242.4	173.3	118.7	167.8	1.201	1.642	1.359
	Despun Section	42.92	0.8	7.8	7.4	88.3			
3500	VOIM Expendables	397.91	5.9	5.2	5.2	0.0			
3600	VOIM Ignition	1,211.07	249.1	222.4	168.3	167.7			
	Spun Section	1,168.15	248.3	189.2	134.6	167.8	1.116	1.476	1.235
	Despun Section	42.92	0.8	7.8	7.4	88.3			
3750	N2H4 Increment	9.00	0.9	0.9	-0.0	15.0			
3800	Magnetometer Extended	1,220.07	250.0	223.2	168.5	168.1			
	Spun Section	1,177.15	249.2	190.0	134.8	168.2	1.116	1.479	1.236
	Despun Section	42.92	0.8	7.8	7.4	88.3			
3950	N2H4 Increment	0.27	0.0	0.0	-0.0	15.0			
4000	Orbiter - Separation	1,220.34	211.4	191.6	161.5	13.2			
	Spun Section	1,177.42	210.6	158.4	127.8	13.0	1.099	1.304	1.173
	Despun Section	42.92	0.8	7.8	7.4	88.3			
4100	Attach Fitting	59.12	9.7	7.5	6.9	95.3			
4200	Orbiter - Launch	1,279.46	220.6	217.5	188.5	13.3			
4250	Science Contingency	4.62	0.0	0.0	0.0	0.0			
4300	S/C Contingency	24.54	0.0	0.0	0.0	0.0			
4400	TOTAL LAUNCH WEIGHT	1,308.62							

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See APPENDIX C - Page C-1/C-2

Figure 3.1.1.1-1. Orbiter General Arrangement

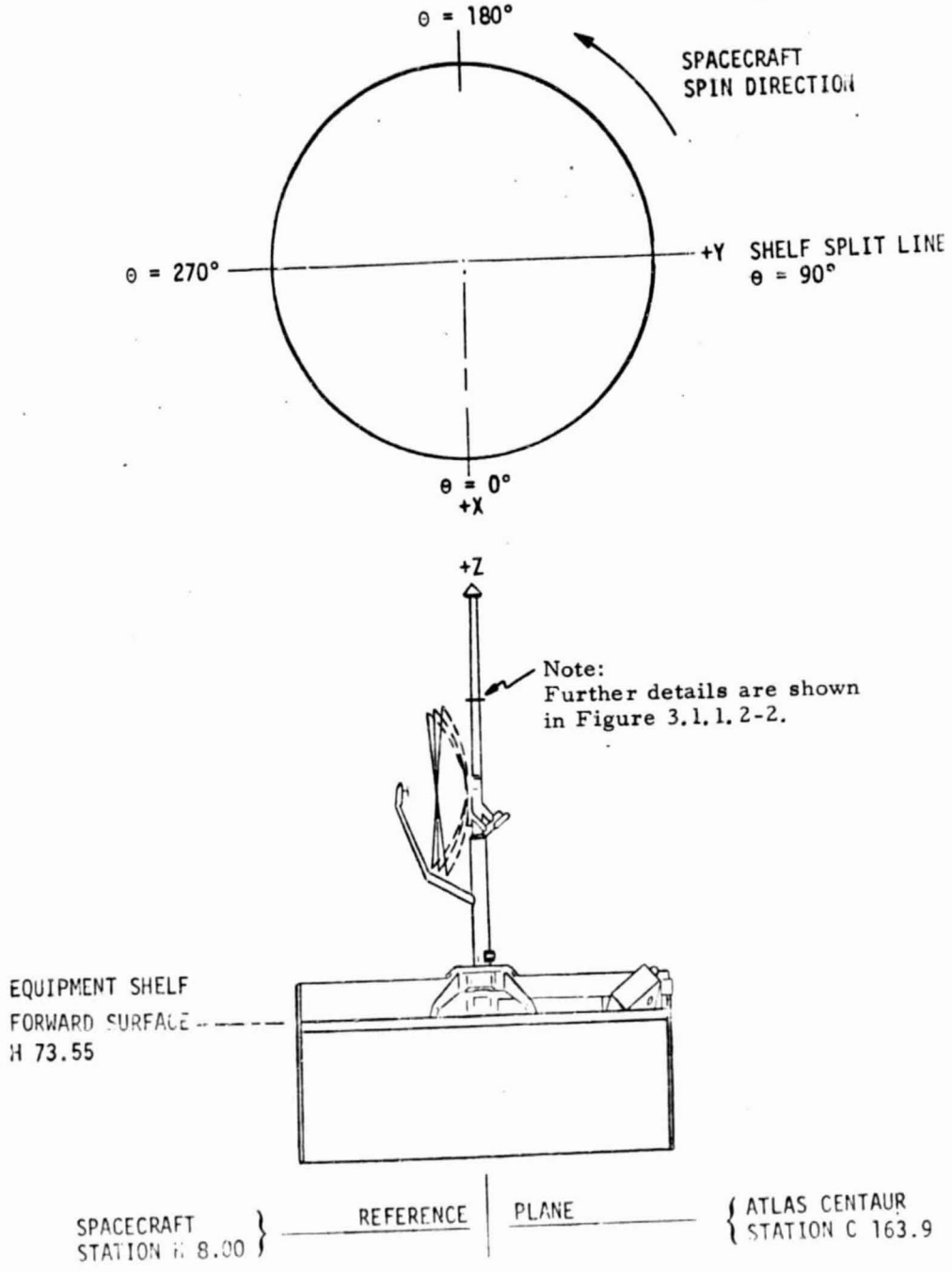
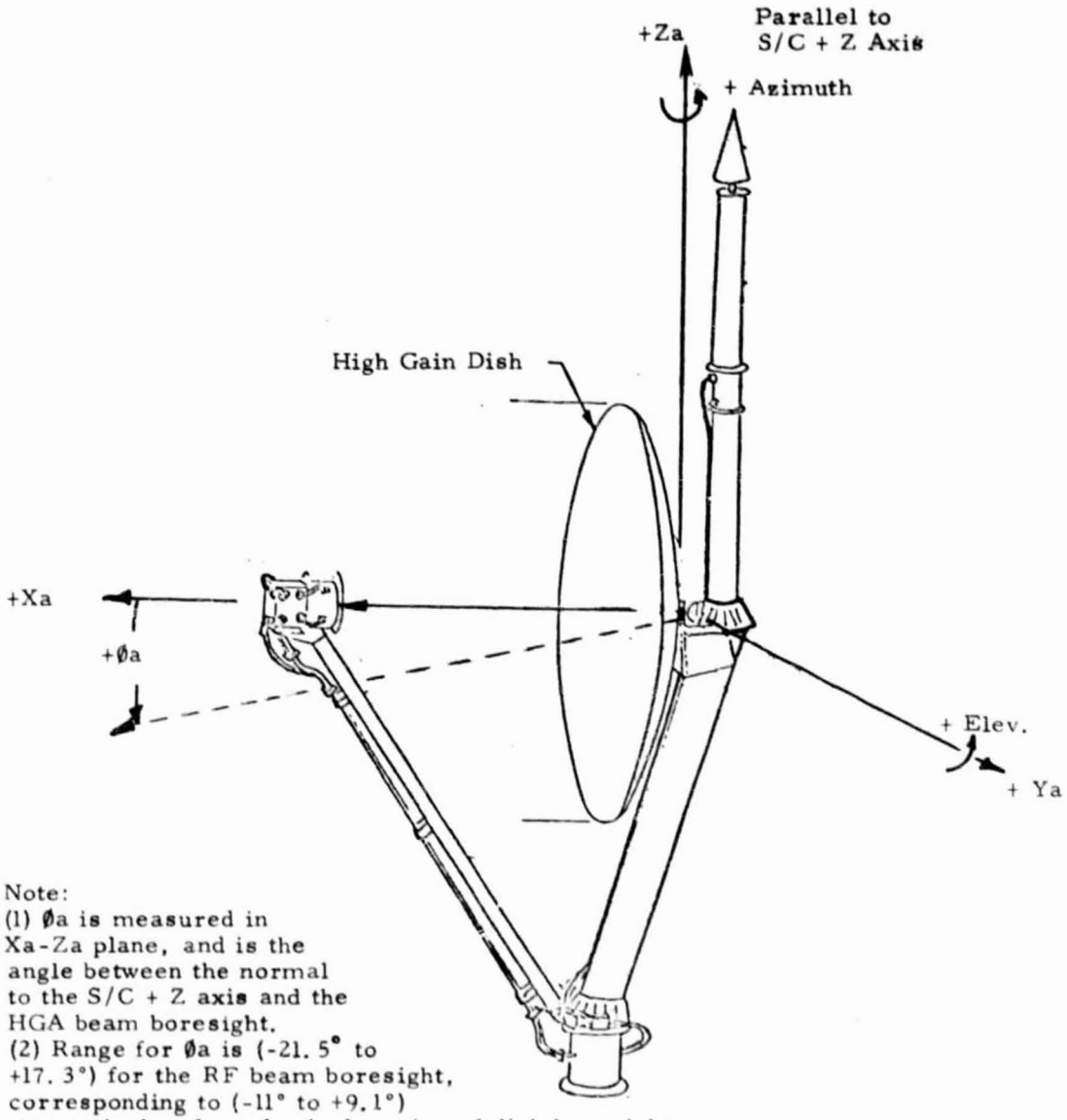


Figure 3.1.1.2-1. Orbiter Coordinate System



Note:
(1) Øa is measured in Xa-Za plane, and is the angle between the normal to the S/C + Z axis and the HGA beam boresight.
(2) Range for Øa is (-21.5° to +17.3°) for the RF beam boresight, corresponding to (-11° to +9.1°) respectively of mechanical motion of dish boresight.
+Ya completes the right-handed Orthogonal Coordinate System.

Figure 3.1.1.2-2. HGA Coordinate Definition

SHELF MOUNTED TEMPERATURE SENSORS

NO	MNEMONIC
TS1	SHLO1T
2	SHLO2T
3	SHLO3T
4	SHLO4T
5	SHLO5T
6	SHLO6T
7	SHLO7T
8	SHLO8T
9	SHLO9T
10	SHLO10T
11	SHLO11T
12	SHLO12T

CODE	NOMENCLATURE
E1	GNMS SENSOR
	GNMS ELECTRONICS
E2	BIMS SENSOR
R1	TRANSPONDER
R2 PA	POWER AMP
R2 DA	DRIVER AMP
R2 IS	ISOLATOR
R3	6 DB HYBRID
R4	3 DB HYBRID
R5	DIPLEXER
R6	[illegible] SWITCH
R7	AFT OMNI ANTENNA
R8	[illegible] SWITCH
R9	FWD OMNI ANTENNA
R10	SWITCH DRIVER
R11	MEDIUM GAIN HORN
D1	DUAL DATA INPUT MODULE
D2	PCM ENCODER
D3	TM PROCESSOR
C1	COMMAND PROCESSOR
C2	COMMAND OUTPUT MODULE
C3	PYRO CONTROL UNIT
A1	SUN SENSOR ASSEMBLY
A2	STAR SENSOR
A3	ATTITUDE DATA PROCESSOR
A4	--
A5	NUTATION DAMPER
A6	--
A7	--
A8	--
A9	--
A10	THERMAL LOUVER
P1	CHARGE/DISCHARGE CONTROLLER
P2	BUS VOLTAGE LIMITED
P3	UNDERVOLTAGE/OVERLOAD CONTROL CURRENT SENSOR
P4	POWER INTERFACE
P5	BATTERY PACK (2 PER BATTERY)
P6	HEATER
P7	SMALL PROBE (FD BRKT (REF)

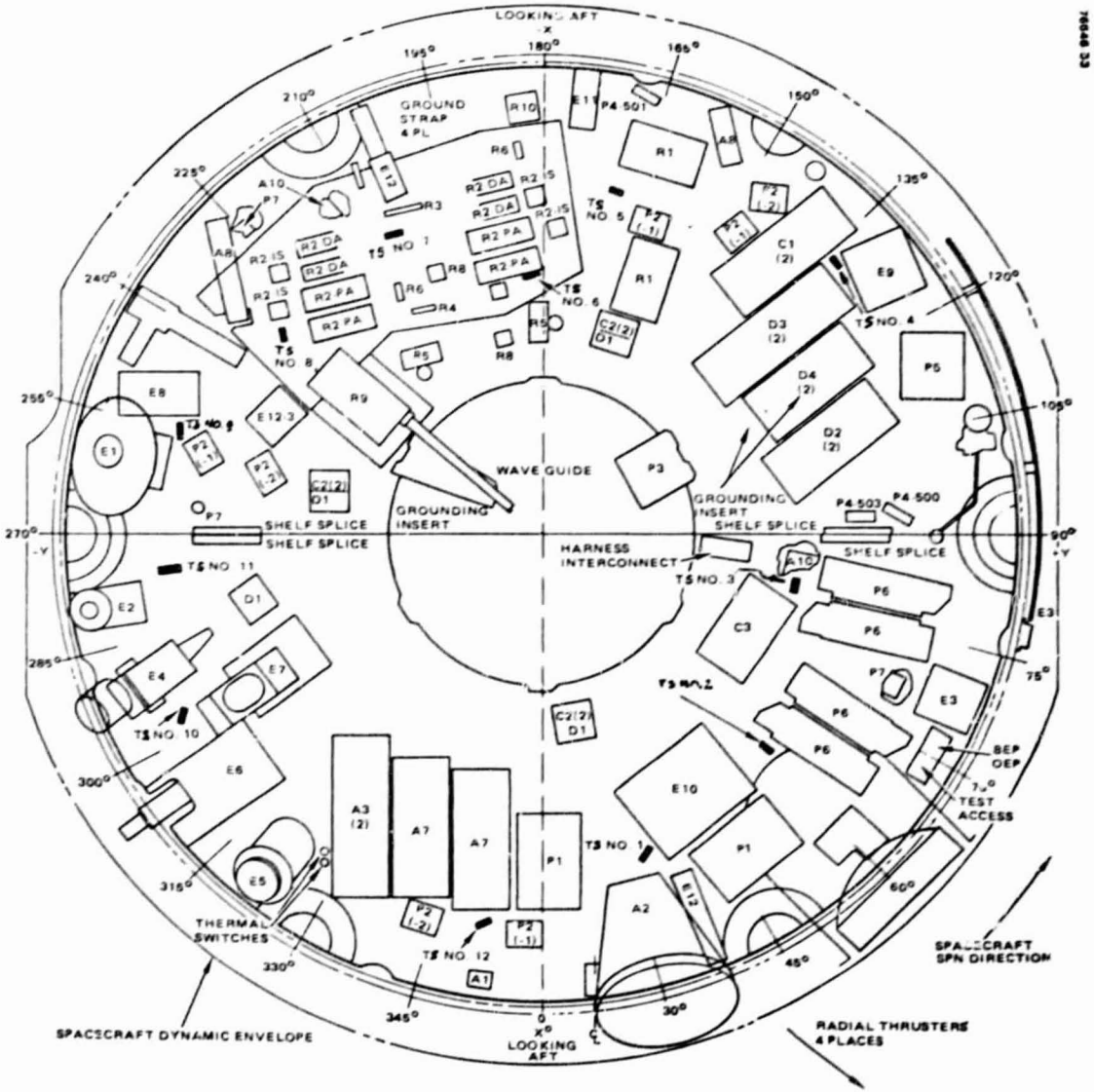


Figure 3.1.1.3.2-1. Orbiter Shelf Unit Location of Shelf-Mounted Temperature Sensors

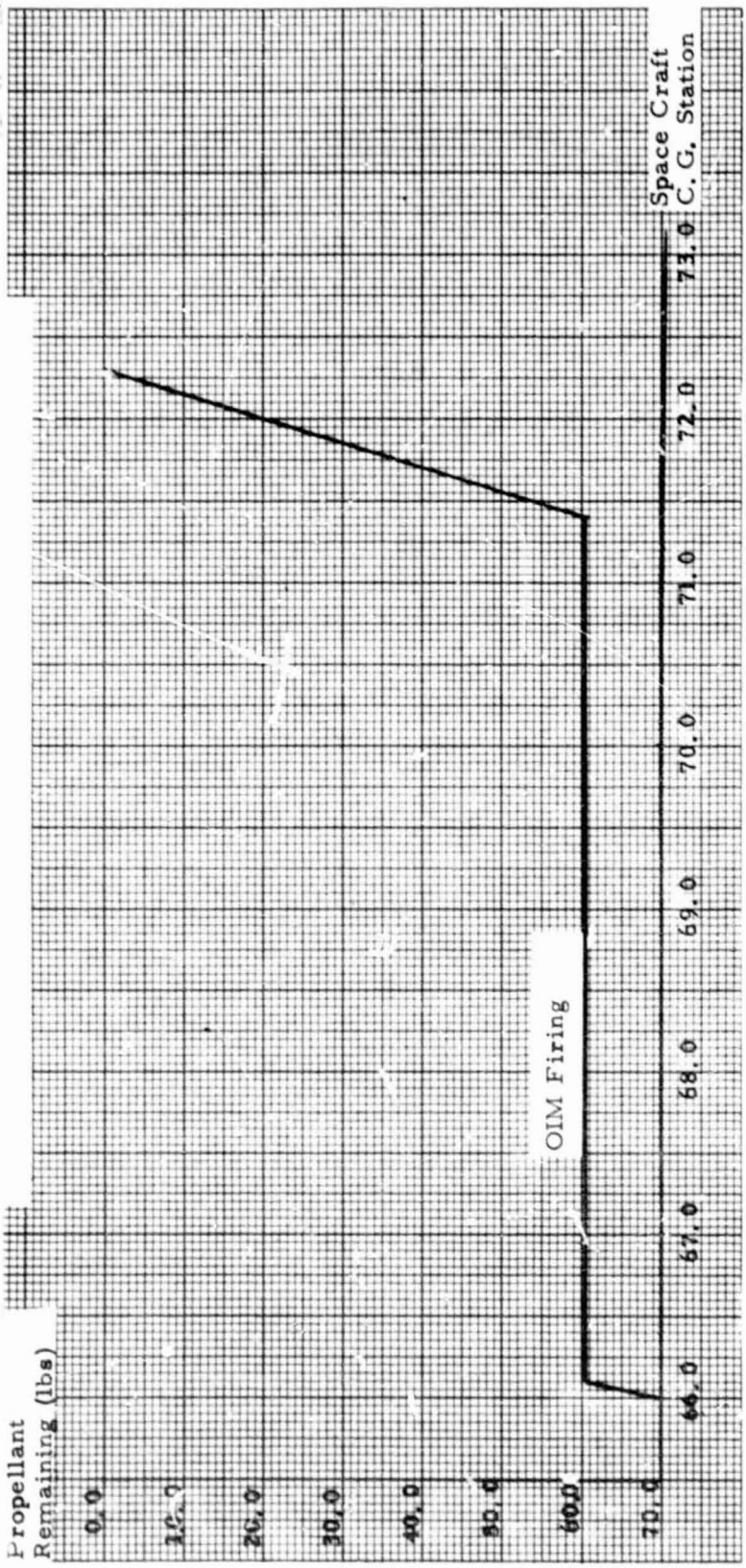


Figure 3.1.2-1. Orbiter C.G. Location Versus Propellant Remaining for a Nominal Mission

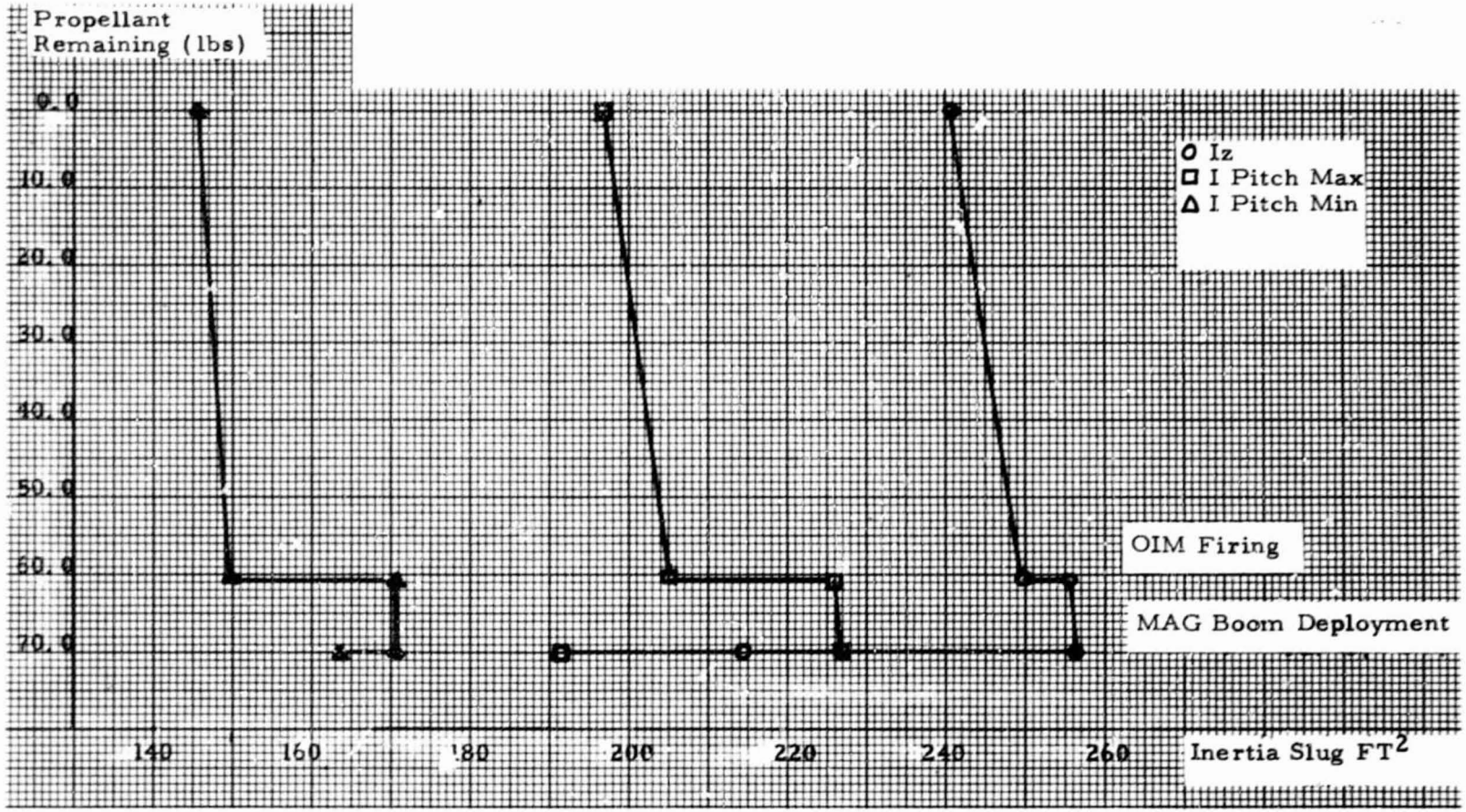


Figure 3.1.2-2. Orbiter Moments of Inertia Versus Propellant Remaining for a Nominal Mission

3.2 THERMAL DESIGN

3.2.1 General Description. The thermal design of the Orbiter is based on isolating the spacecraft units and scientific instruments from the external environmental extremes encountered during the mission to the maximum extent practical and using a combination of passive and active elements to maintain acceptable unit temperatures. The primary factors affecting the thermal design of the Orbiter are the increase in solar intensity as the spacecraft travels towards Venus, the variations in internal power dissipation during different phases of the mission, the variations in sun angle during the mission and the 3.8 hour solar eclipse that occurs late in the on-orbit phase of the mission. In addition, the Orbiter is designed to accommodate failure modes including extreme under powered conditions and extreme off nominal mission attitude conditions. These failure mode conditions plus the variation in environments encountered during the normal mission require the thermal design to be highly adaptive and almost completely automatic. The only elements that can be controlled from the ground are certain propulsion heaters, the OIM heaters and the bus limiters that control the shelf heaters.

The equipment compartment is the space defined by the equipment shelf, the forward thermal barrier and the solar array cylinder. Multilayer kapton insulation blankets are used to isolate this compartment from the external environment. The isolation is not complete due to the inherent limitations of any insulation system and the various penetrations for spacecraft units and scientific instruments that must view out to space. In addition, the aft side of the equipment shelf is used as the primary radiator for the electrical power dissipation inside the equipment compartment. Fifteen thermal louvers are mounted on this radiator surface to help accommodate the large variations in power dissipation by varying the effective thermal emittance as a function of shelf temperature.

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Heaters are provided to maintain all elements of the propulsion subsystem at 40°F or higher to prevent propellant freezing. Some of these heaters are commandable on-off, others are commandable primary-secondary, and some are continuously powered. Commandable heaters are provided to control minimum Orbit Insertion Motor and Safe and Arm temperatures. Heaters are also provided on the equipment shelf to provide additional dissipation during an extreme powered down condition such as an Undervoltage/Overhead (UV/OL) trip. Automatic shelf heater turn-on is required to counteract a sustained UV/OL trip that could occur when the spacecraft is not being monitored. In addition, a heater turn-on override is provided to prevent battery depletion if such a trip occurs because of insufficient solar array power or an extended eclipse condition.

Thermal finishes are used on externally mounted units such as antennas to limit their maximum temperatures and on external surfaces of the spacecraft to achieve thermal balance within the acceptable temperature range.

All heaters and temperature sensors, including applicable command and telemetry mnemonics are shown in Figures 3.1.1.3.2-1, 3.2.1-1, 3.2.1-2, 3.4.1-3, 3.8.1.2-1 and 3.8.1.2-2. All temperature sensors are listed and located in Table 3.2.2.5-1.

3.2.2 Thermal Control Units and Components

3.2.2.1 Heaters

3.2.2.1.1 Propulsion Heaters. All elements of the propulsion subsystem except the semicircular run of pressurant line are heated to maintain the propellant temperature at 40°F or higher. The mechanically redundant components and units which are located downstream of the latch valves have non-redundant heaters while the non-redundant components and units upstream of the latch valves have redundant heaters. Figure 3.2.2.1.1-1 shows

this division between redundant and non-redundant heaters schematically.

The non-redundant heaters on the radial thrusters and the lines and components are continuously powered from the essential bus; there is no way that these heaters can be turned off either by ground command or any on-board failure mode sequence. The satisfactory operation of these heaters can be verified by monitoring Radial jets 1 through 4 temperatures telemetry channels (VJET1T through VJET4T), R1 and R2 line heater (voltage) status (VJ12HS), and R3 and R4 line heater (voltage) status (VJ34HS).

The forward axial thruster and the aft axial thruster pair and associated lines and components both have on-off commandable non-redundant heaters. The forward axial thruster heater is commanded on by HTJ19 (HTJA9 is the redundant command) and commanded off by HTJ1Ø (HTJAØ). The aft axial thruster pair heaters are commanded on by HTJ29 (HTJB9) and commanded off by HTJ2Ø (HTJBØ).

All Axial Jet heaters should be ON at launch, and left ON, unless the following occurs: The Forward Axial Jet heater should be commanded OFF if (1) Forward Axial Jet 5 Temperature (VJET5T) reaches or exceeds +154°F or (2) Propellant Line Temperature 2 (VLIN2T) reaches or exceeds +145°F. The Aft Axial Jet heaters should be commanded OFF if (1) Aft Axial Jet 6 Temperature (VJET6T) or Aft Axial Jet 7 Temperature (VJET7T) reaches or exceeds +154°F or (2) Propellant Line Temperature 3 (VLIN3T) or Propellant Line Temperature 4 (VLIN4T) reaches or exceeds +145°F. Whichever of the Jet heaters had been turned OFF should be commanded ON again if any of the associated telemetry temperature indicators identified above reaches or decreases below +61°F. All of the Axial Jet heaters should be commanded ON prior to any eclipse. The Aft Axial Jet heaters are expected to be required OFF for Orbit insertion for a 100° sun aspect angle.

Revision

The redundant heaters upstream of the latch valves are command selectable between the primary and secondary circuits; one circuit is always powered even during extreme powered down conditions. The command for selecting the primary circuit is HTT19 (HTTA9); the command for selecting the secondary circuit is HTT29 (HTTB9). ON-OFF status of either of the circuits can be verified by the following telemetry: PRI/SEC Prop Tank Heater Status (PHTTKS), L/V Primary heater status (VLVPHS), and F and D Valve Primary Heater status (VFDVHS). The primary circuit should be ON throughout the mission; the secondary heater circuit should be switched in if a decrease in Tank 1 temperature (VTNK1T), or Tank 2 temperature (VTNK2T), or propellant Line Temperature 1 (VLIN1T) is observed which is unrelated to environmental changes (e.g., eclipse, heating effect of planet, attitude change w.r.t. the sun l.o.s.), or occurs for steady-state environmental conditions.

The Orbit Insertion Motor (OIM) and the Safe and Arm device (S&A) have fully redundant, on-off commandable heater circuits. The OIM uses two separate heater elements, one on the aft case and one near the mounting flange. The S&A uses a single heater element located on the side of the S&A case. The three heaters are connected in series and commanded on by HTM19 (HTMA9) and commanded off by HTM1Ø (HTMAØ). The redundant heaters are in the same physical locations and are commanded on by HTM29 (HTMB9) and commanded off by HTM2Ø (HTMBØ).

The primary heaters should be ON at launch, and left ON unless the following occurs: The heaters should be commanded OFF if any one of OIM throat temperature (VTHROT) or OIM case temperature (VCASET) or OIM Safe & Arm Temperature (VSAFET) reaches or exceeds +90°F. The heaters should be commanded ON again if (1) VTHROT or VCASET reaches or decreases below +45°F or (2) VSAFET reaches or decreases below +14°F. The heaters are expected to be ON through Orbit Insertion for a minimum sun aspect angle (~60°) or OFF for a

maximum sun aspect angle (~100°). The heaters are not required after OIM firing.

It should be noted that both heater circuits can be powered simultaneously if the OIM is in serious danger of becoming too cold.

3.2.2.1.2 Shelf Heaters. Heaters are required on the equipment shelf to provide additonal dissipation during an extreme powered down condition such as an Undervoltage/Overload (UV/OL) trip. Automatic heater turn-on is required to counteract a sustained UV/OL trip that could occur when the spacecraft is not being monitored. This is accomplished by using thermostatic switches on the equipment shelf to lower the set points on two of the bus voltage limiters (Nos. 2 and 6) of the power subsystem. Lowering of the bus voltage limiter set points forces the first 130 watts of excess solar panel power to be dissipated in the equipment shelf heaters rather than in the load resistors mounted on the back side of the solar panel. The thermal switches and shelf heaters on the two halves of the equipment shelf operate independently of each other. The thermostatic switches on each shelf half are set to turn the heaters on at 35 ±5°F and to turn them off at 55 ±5°F. The lower voltage set point on the battery shelf bus limiter is 28.5 volts while on the RF shelf bus limiter it is 29.0 volts. This difference in voltage set points means that if the two shelf halves are at the same temperature, the first 65 watts of excess solar panel power will go into the battery shelf. The location of the shelf heaters and thermostatic switches are shown in Figures 3.8.1.2-1 and 3.8.1.2-2.

Since the bus limiters dissipate only excess solar panel power, the shelf heater cannot operate during conditions such as apoapsis eclipse when there is not an excess of solar panel current. During apoapsis eclipse, unit temperatures are maintained at acceptable levels transiently by spacecraft heat capacity and thermal insulation. Shelf heaters can be turned off by turning off the appropriate bus limiter.

The functioning of the shelf heaters is also discussed in Sections 3.8.2.3 and 4.3.5.2.

3.2.2.2 Thermal Louvers. The fifteen thermal louvers mounted on the aft side of the equipment shelf are used to regulate the heat flow from the equipment shelf by varying the effective thermal emittance with movable louver blades. When the blades are fully closed, their polished aluminum surface has an effective thermal emittance of 0.1. When the louver blades are fully open, the exposed quartz mirror has an effective thermal emittance of 0.61.

Each thermal louver covers an 8-inch by 16-inch area and has four pairs of louver blades. Each pair of louver blades is actuated by a coiled bi-metallic strip mounted in a housing between the blades; the louver blade pair is rotated from the closed to the open position as the bi-metallic strip is heated. The bi-metallic actuators are conductively isolated from the louver structure so that the louver blade position is determined by the heat radiated from the equipment shelf. Each blade pair in a particular louver is set to open and close at the same temperature. However, there are two different set point ranges used in order to obtain the proper unit temperatures. Thirteen louvers are set to close completely at or above 50°F and to open completely at or below 85°F. The remaining two louvers which are used under the batteries are set to close completely at or above 40°F and to open completely at or below 75°F. The physical arrangement of these two different louver types is shown in Figure 3.1.1.1-1 (Appendix C).

The thermal dissipation from the louver varies with temperature due to the position of the louver blades and the temperature of the radiating surface. In addition, louver performance is affected by spacecraft spin induced radial acceleration that increases the bearing friction and thus creates a hysteresis effect. The performance of the two louver types is shown in Figures 3.2.2.2-1 and 3.2.2.2-2.

3.2.2.3 Thermal Doublers. Because of the honeycomb construction of the equipment shelf, lateral heat conduction is relatively low. In order to adequately spread out the heat from high power units so that it can be dissipated, a thermal doubler is used. The thermal doubler on the Orbiter is an 0.070-inch thick beryllium plate mounted between the high power communications equipment such as the power amplifiers and the equipment shelf.

To enhance the thermal conduction across the unit mounting interfaces, RTV-566 is applied to the mounting surfaces of units with high power dissipation.

3.2.2.4 Insulation Blankets. Insulation blankets are used on the spacecraft to help isolate it from the environmental extremes encountered during the mission. The blankets are made from fifteen layers of aluminized kapton. The interior layers of these blankets are 1/4 mil thick, aluminized on one side only, and crinkled to achieve the minimum contact between layers. To minimize weight, separators are not used between layers. The outer layer is 1 mil kapton, aluminized on one side only with the bare kapton side exposed.

The blankets are made in various segments to ease access and assembly. The blanket segments are attached to each other with Velcro fasteners. In order to minimize heat leaks through these attachments, the number of fasteners and the stitching used to attach the Velcro to the blankets is minimized.

3.2.2.5 Temperature Sensors. Temperature sensors are mounted throughout the spacecraft to measure and provide telemetry of the thermal status of the spacecraft and selected units. Table 3.2.2.5-1 lists each of these temperature sensors, its location, reference designator, and Hughes part number.

3.2.3 Solar Heating Restrictions. The normal attitude of the Orbiter is with the spin axis perpendicular to the sun's rays (sun angle of 90°

TABLE 3.2.2.5-1

TELEMETRY DESCRIPTION

Reference Designator	Telemetry Description	HAC Part Number
	Communications Subsystem	
RVCO1T	Receiver 1 VCO Temp	TI RTH-42
RVCO2T	Receiver 2 VCO Temp	TI RTH-42
ROSC1T	Transmitter Aux Oscillator 1 Temp	TI RTH-42
ROSC2T	Transmitter Aux Oscillator 2 Temp	TI RTH-42
RAMP1T	Power Amp 1 Temp	908683-4
RAMP2T	Power Amp 2 Temp	908683-4
RAMP3T	Power Amp 3 Temp	908683-4
RAMP4T	Power Amp 4 Temp	908683-4
RXMTRT	X-Band Transmitter Temp	908631-32
	Control Subsystem	
ASTART	Star Sensor Temp	908631-32
ABAP1T	BAPTA Inner Race Bearing Temp	908632-2
ABAPOT	BAPTA Outer Race Bearing Temp	908632-2
ADCE1T	DCE 1 Temp	908631-31
ADCE2T	DCE 2 Temp	908631-31
APOSMT	HGA Position Motor Temp	908683-5
	Power Subsystem	
PB1P1T	Battery 1, Pack 1 Temp	908635-4
PB1P2T	Battery 1, Pack 2 Temp	908635-4
PB2P1T	Battery 2, Pack 1 Temp	908635-4
PB2P2T	Battery 2, Pack 2 Temp	908635-4
PPAN1T	Solar Panel 1 Temp	908683-4
PPAN2T	Solar Panel 2 Temp	908683-4
	Propulsion Subsystem	
VTNK1T	Tank 1 Temp	908631-32
VTNK2T	Tank 2 Temp	908631-32
VJET1T	Radial Jet 1 Temp	908683-4
VJET2T	Radial Jet 2 Temp	908683-4
VJET3T	Radial Jet 3 Temp	908683-4
VJET4T	Radial Jet 4 Temp	908683-4
VJET5T	Fwd Axial Jet 5 Temp	908683-4
VJET6T	Aft Axial Jet 6 Temp	908683-4
VJET7T	Aft Axial Jet 7 Temp	908683-4

TABLE 3.2.2.5-1 (Continued)

Reference Designator	Propulsion Subsystem (Cont'd)	HAC Part Number
VLIN1T	Propellant Line Temp 1	908631-32
VLIN2T	Propellant Line Temp 2	908631-32
VLIN3T	Propellant Line Temp 3	908631-32
VLIN4T	Propellant Line Temp 4	908631-32
VTHROT	OIM Throat Temp	908631-32
VCASET	OIM Case Temp	908631-32
VSAFET	OIM Safe and Arm Temp	908631-32
	Structure-Harness Subsystem	
SHLO1T	Shelf Temp 1	908631-32
SHLO2T	Shelf Temp 2	908631-32
SHLO3T	Shelf Temp 3	908631-32
SHLO4T	Shelf Temp 4	908631-32
SHLO5T	Shelf Temp 5	908631-32
SHLO6T	Shelf Temp 6	908631-32
SHLO7T	Shelf Temp 7	908631-32
SHLO8T	Shelf Temp 8	908631-32
SHLO9T	Shelf Temp 9	908631-32
SHL10T	Shelf Temp 10	908631-32
SHL11T	Shelf Temp 11	908631-32
SHL12T	Shelf Temp 12	908631-32
SOPRAT	ORPA Temp	908631-32
SOIMST	OIMS Temp	908631-32
SOETPT	OETP Temp	908631-32
SONMST	ONMS Temp	908631-32
SOPAIT	OPA Temp	908631-32
SORADT	ORAD Temp	908631-32

as measured from the forward spin axis). Planned and unplanned excursions from this normal attitude affect the thermal balance of the spacecraft. Depending on the operating mode of the spacecraft and its distance from the sun, off attitude orientations can be sustained for varying lengths of time before spacecraft overheating occurs. Figure 3.2.3-1 shows the time limits for various sun angles as a function of time from launch and spacecraft operating mode.

3.2.4 Ranges of Expected Mission Temperatures. Table 3.2.4-1 gives the range of expected temperatures in a nominal mission for each telemetered temperature.

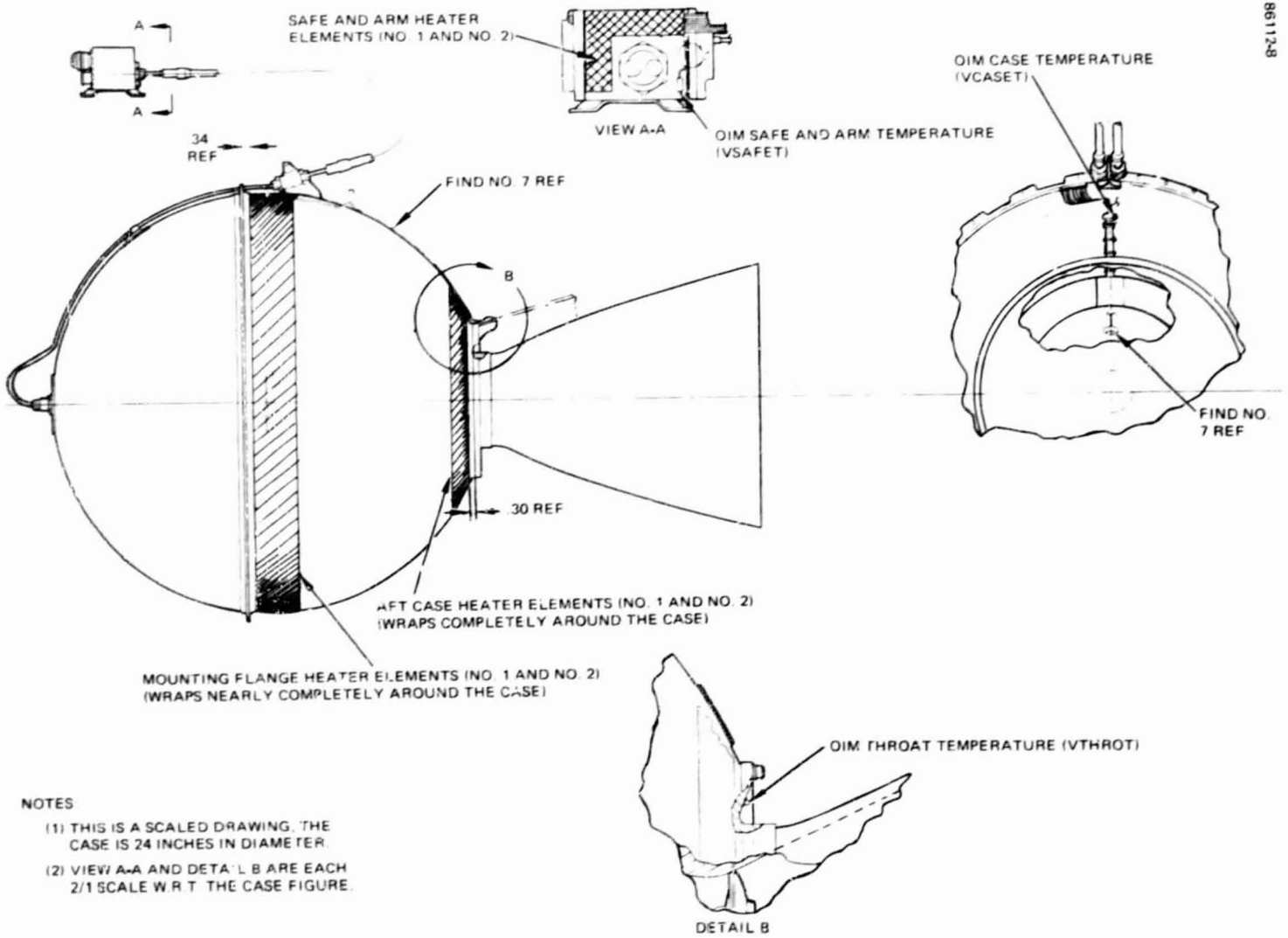
TABLE 3.2.4-1

RANGE OF EXPECTED MISSION TEMPERATURES
FOR EACH TELEMETERED TEMPERATURE
(NOTE: Representative nominal mission temperatures are described in Reference 1, 5, 35)

Reference Designator	Telemetry Description	Expected Mission Temp. Range (°F)	
		Maximum	Minimum
	Communications Subsystem		
RVCO1T	Receiver 1 VCO Temp	105	41
RVCO2T	Receiver 2 VCO Temp	104	44
ROSC1T	XMTR Aux Osc 1 Temp	105	38
ROSC2T	XMTR Aux Osc 2 Temp	101	42
RAMP1T	Power Amp 1 Temp	131	5
RAMP2T	Power Amp 2 Temp	122	8
RAMP3T	Power Amp 3 Temp	115	3
RAMP4T	Power Amp 4 Temp	105	4
RXMTRT	X-Band XMTR Temp	104	21
	Control Subsystem		
ASTART	Star Sensor Temp	121	-2
ABAPIT	BAPTA Inner Race Bearing Temp	81	42
ABAPOT	BAPTA Outer Race Bearing Temp	85	31
ADCE1T	DCE 1 Temp	97	26
ADCE2T	DCE 2 Temp	91	29
APOSMT	HGA Position Motor Temp	170	-130
	Power Subsystem		
PB1P1T	Battery 1, Pack 1 Temp	87	36
PB1P2T	Battery 1, Pack 2 Temp	87	37
PB2P1T	Battery 2, Pack 1 Temp	91	39
PB2P2T	Battery 2, Pack 2 Temp	95	38
PPAN1T	Solar Panel 1 Temp	274	-238
PPAN2T	Solar Panel 2 Temp	274	-232
	Propulsion Subsystem		
VTNK1T	Tank 1 Temp	102	60
VTNK2T	Tank 2 Temp	102	60

TABLE 3.2.4-1 (Continued)

Reference Designator	Telemetry Description	Expected Mission Temp. Range (°F)	
		Maximum	Minimum
	Propulsion Subsystem (Continued)		
VJET1T	Radial Jet 1 Temp	149	47
VJET2T	Radial Jet 2 Temp	171	56
VJET3T	Radial Jet 3 Temp	156	54
VJET4T	Radial Jet 4 Temp	163	58
VJET5T	Fwd Axial Jet 5 Temp	121	56
VJET6T	Aft Axial Jet 6 Temp	120	47
VJET7T	Aft Axial Jet 7 Temp	133	63
VLIN1T	Propellant Line Temp 1	112	73
VLIN2T	Propellant Line Temp 2	117	58
VLIN3T	Propellant Line Temp 3	151	86
VLIN4T	Propellant Line Temp 4	117	56
VTHROT	OIM Throat Temp	95	35
VCASET	OIM Case Temp	89	26
VSAFET	OIM Safe and Arm Temp	83	24
	Structure-Harness Subsystem		
SHL01T	Shelf Temp 1	96	20
SHL02T	Shelf Temp 2	82	31
SHL03T	Shelf Temp 3	80	35
SHL04T	Shelf Temp 4	111	46
SHL05T	Shelf Temp 5	109	19
SHL06T	Shelf Temp 6	115	14
SHL07T	Shelf Temp 7	112	7
SHL08T	Shelf Temp 8	104	10
SHL09T	Shelf Temp 9	94	10
SHL10T	Shelf Temp 10	94	13
SHL11T	Shelf Temp 11	100	10
SHL12T	Shelf Temp 12	99	17
SORPAT	ORPA Temp	103	10
SOIMST	OIMS Temp	108	14
SOETPT	OETP Temp		
SONMST	ONMS Temp		
SOPAIT	OPA Temp		
SORADT	ORAD Temp	89	26



NOTES

(1) THIS IS A SCALED DRAWING. THE CASE IS 24 INCHES IN DIAMETER.

(2) VIEW A-A AND DETAIL B ARE EACH 2/1 SCALE W.R.T. THE CASE FIGURE.

FIGURE 3.2.1-1. OIM HEATERS AND TEMPERATURE SENSORS

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See APPENDIX C - Pages C-8, C-9 and C-10.....

Figure 3.2.1-2.

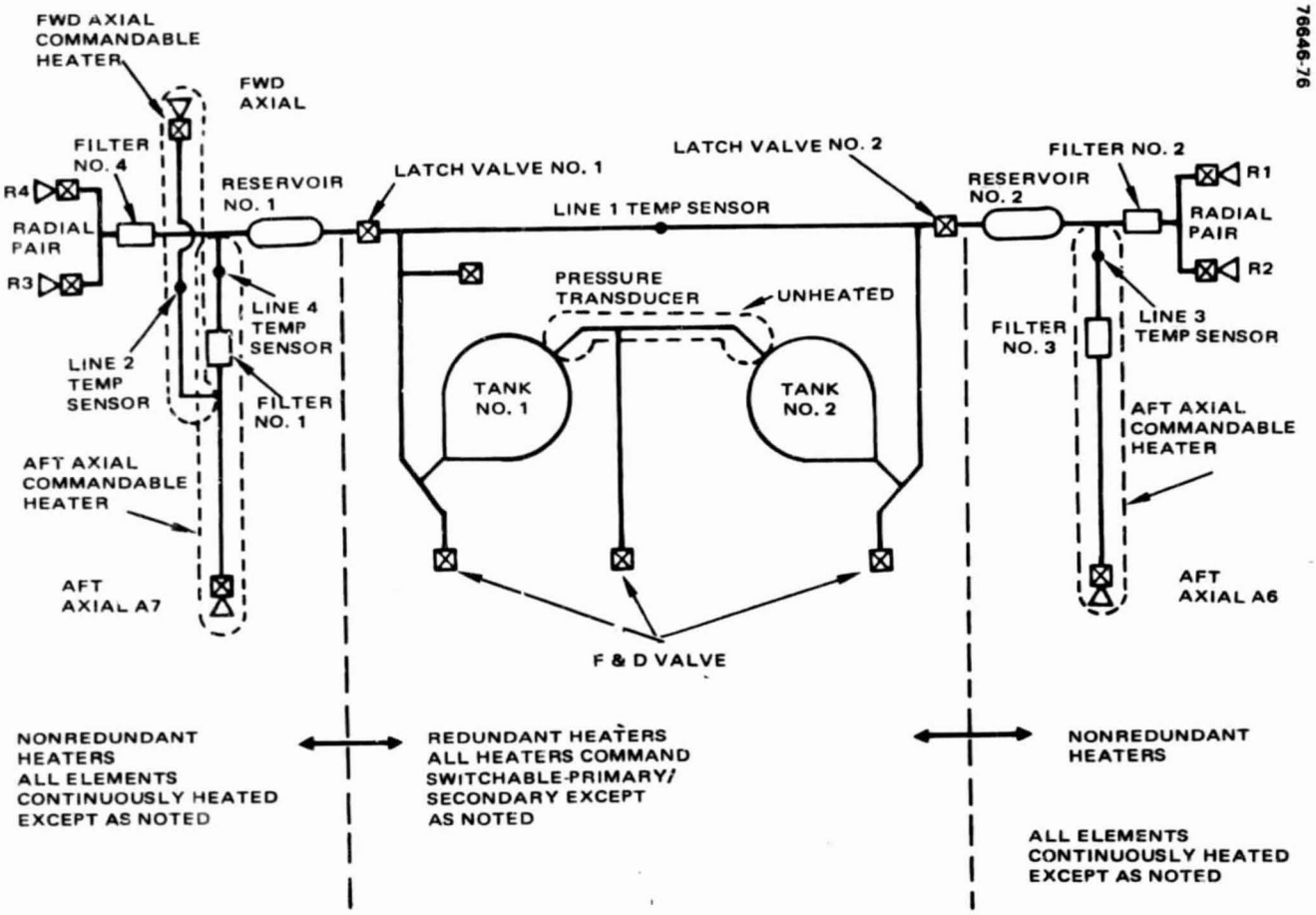


FIGURE 3.2.2.1.1-1. ORBITER PROPULSION SUBSYSTEM HEATER CONFIGURATION

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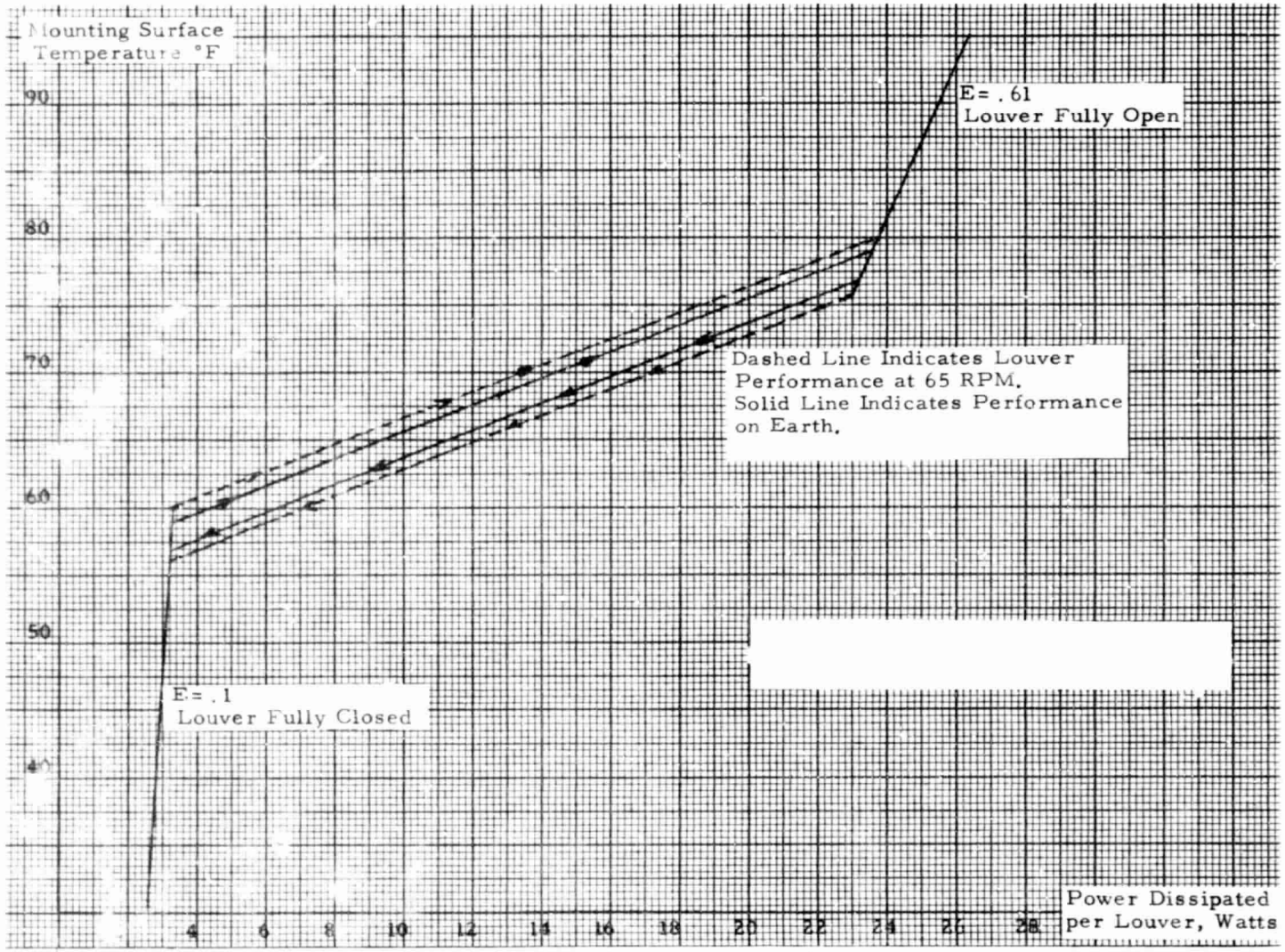


Figure 3.2.2.2-1. Louver Performance (General Type)

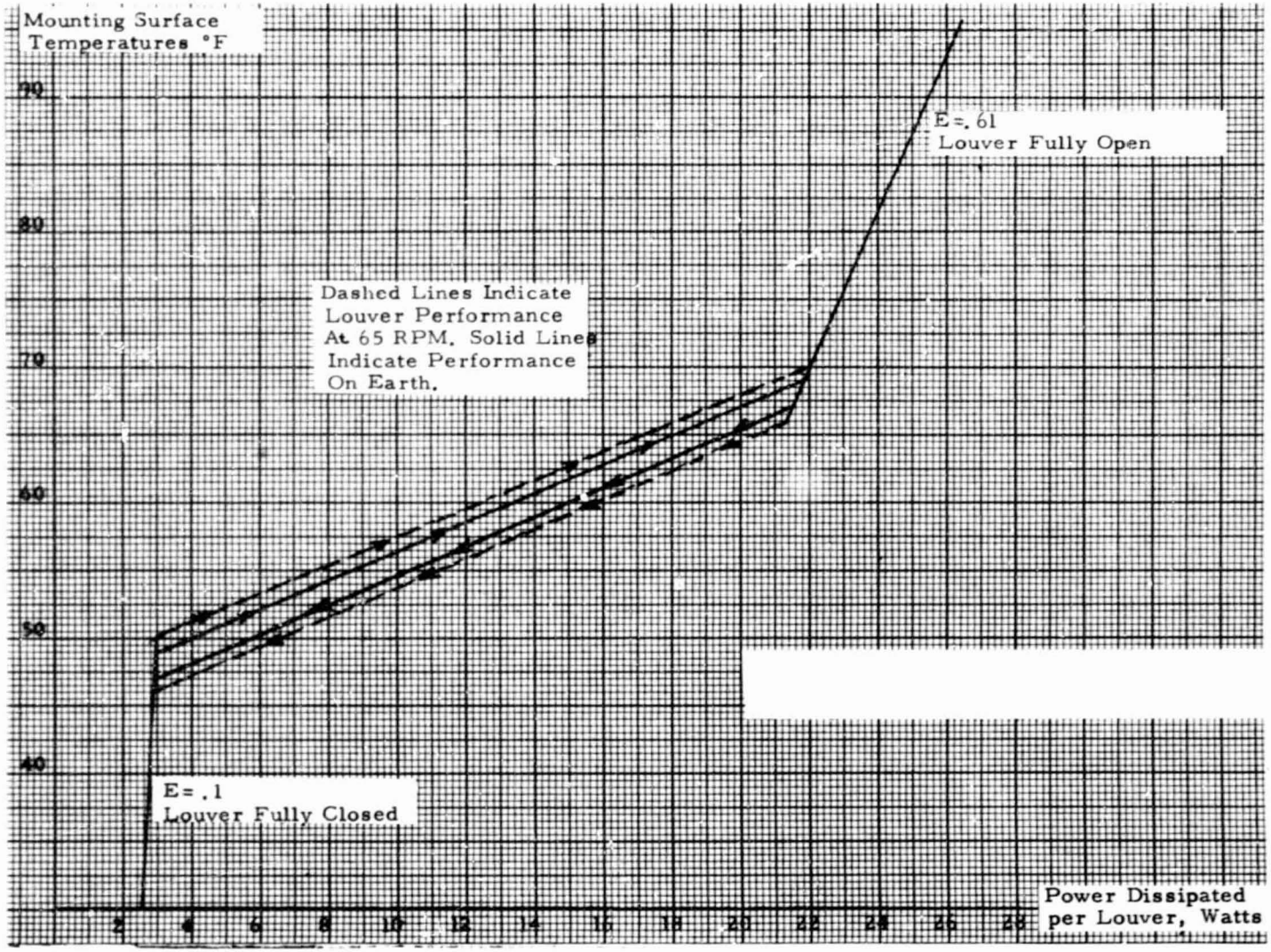


Figure 3.2.2.2-2. Louver Performance (Battery Type)

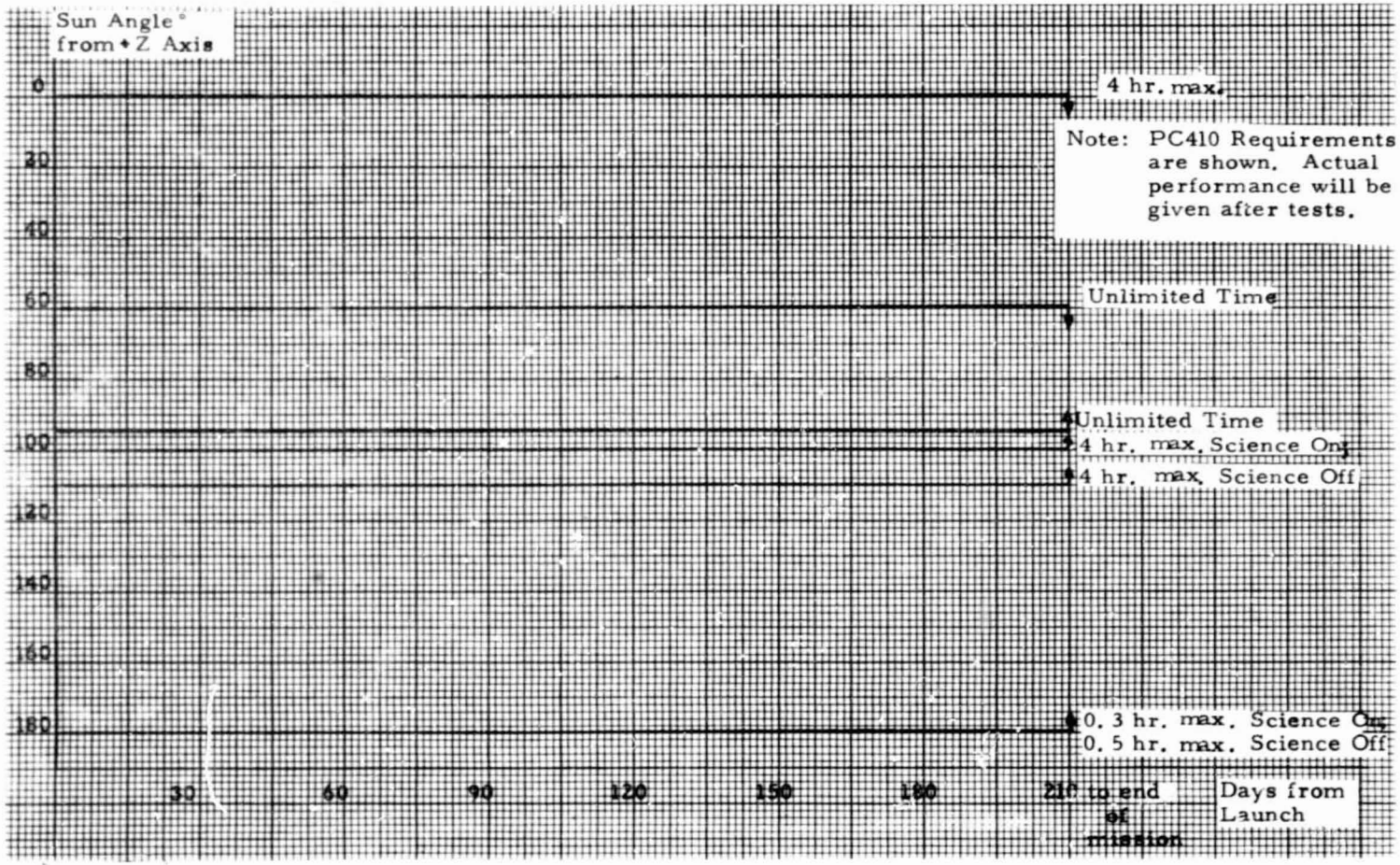


Figure 3.2.3-1. Allowable Time at Various Sun Angles

3.3 CONTROL SUBSYSTEM

3.3.1 Subsystem Description

3.3.1.1 Functional Description. The control subsystem provides the sensors and onboard logic to accomplish the following functions:

(a) Spin rate and spacecraft attitude determination
(b) Science roll reference signals
(c) Control of thrusters for spin axis precession maneuvers, spin speed control, and spacecraft velocity maneuvers
(d) Orbiter high gain antenna azimuth despin control to a desired Earth line-of-sight pointing
(e) Orbiter high gain antenna elevation positioning
(f) Magnetometer deployment
(g) Nutation damping

The simplified block diagram of Figure 3.3.1-1 illustrates the control subsystem elements that perform these functions.

The attitude determination design utilizes dual slit sun and star sensors to measure the sun and star positions relative to spacecraft coordinates. The sensors detect the selected celestial target crossing through their field-of-view (FOV) each spin period due to vehicle spin rotation. Both sensors use silicon detectors to develop an electrical pulse when the target crosses the FOV. The star sensor is capable of detecting 25 (≥ +1.5 silicon magnitude) stars. Time interval measurements are made between command selected sensor outputs and then telemetered to a ground station for computation of spacecraft attitude. A spin angle star gate, changeable in time occurrence in the spin period by command, is utilized in conjunction with a star detection threshold setting that is changeable in threshold level by command to provide discrimination of star references.

The inertial reference can be command selected to be either the sun or a star. This detected signal then becomes the onboard roll reference for attitude maneuvers, high gain antenna pointing, attitude measurements and the science reference signals. A "free-run" mode is also selectable which substitutes an electronically derived pulse for the inertial reference.

The Orbiter spacecraft utilizes seven thrusters to provide the capability for performing attitude maneuvers, spin speed control, and velocity change maneuvers. Possible thruster configurations and the selection required to perform a desired maneuver are discussed in Section 3.4. The control subsystem provides the capability of command selecting the thrusters and controlling the fire mode and firing duration. Thrusters may be fired in a pulsed or continuous mode. In the pulse mode the firing pulse can be command selected to occur at any azimuth angle. An additional option permits the generation of a firing pulse 180 degrees from the first pulse (alternate fire mode). Further options permit two selectable pulse widths (128 ms or 512 ms) and the pulsed firing duration to be controlled by time or pulse count. Details and application of this feature are given in Sections 3.3.2.3.5, 3.3.3, 3.3.4, and 4.3.1.

The High Gain Antenna (HGA) requires continuous control to maintain the antenna boresight pointing towards the Earth. Despin, position and slew control is provided to maintain the antenna boresight pointing at a ground commandable phase relationship relative to the selected inertial reference. The HGA elevation pointing for the dual RF frequency occultation experiment is provided by slewing the position of the antenna reflector relative to a fixed feed.

Magnetometer sensors that measure in three different direction are required to be deployed (on a single boom) after launch to a position where the spacecraft's magnetic field will not

interfere with the sensor's magnetic measurements.

The control subsystem also provides passive nutation damping of the spacecraft.

3.3.1.2 Hardware Implementation. The equipment required to perform the control subsystem functions consists of an attitude data processor, sun sensor assembly, star sensor assembly, despin control electronics, bearing and power transfer assembly, a high gain antenna elevation positioner, a magnetometer deployment mechanism, and nutation damper. The functions performed by these units are as follows:

(a) Attitude Data Processor (ADP). The ADP is a command controlled unit which processes the sun and star inputs for attitude determination and on-board roll reference, controls thruster firing modes for precession, velocity and spin maneuvers, and provides the roll index pulse (RIP), roll reference pulse (F_S) and sector pulses for the scientific instruments. Two units are provided for redundancy.

(b) Sun Sensor Assembly. The sun sensor assembly contains a mid-range sensor, an upper sensor and lower sensor to provide the required field-of-view. The output from a vertical slit (ψ pulse) provides an inertial reference and spin speed information. The output from a canted slit (ψ_2 pulse) provides indirectly sun pitch or aspect angle (sun elevation angle wrt the spacecraft +Z axis) information. Each sensor is redundant.

(c) Star Sensor Assembly. The star sensor is a command controlled unit which contains a vertical (ψ^*) slit and canted ($\psi_2{}^*$) slit either of which can be used to provide star pulses for an inertial reference or attitude determination.

(d) Despin Control Electronics (DCE). The DCE is a command controlled unit which provides the capability to despin and control the pointing of the high gain antenna (HGA) using the BAPTA motor, provides elevation control of the HGA by providing stepping pulses to the elevation positioner and provides two rate controlled pulses for scientific instruments.

(e) Bearing and Power Transfer Assembly (BAPTA). The BAPTA, as controlled by the DCE, provides capability to despin the HGA. It has the capability to transfer electrical signals and the S-band and X-band RF signals between the spun and despun sections.

(f) HGA Elevation Positioner. The HGA elevation positioner, as controlled by the DCE, provides capability of moving the HGA dish at command controlled slew rates and provides information of the dish angular orientation to telemetry.

(g) Magnetometer Deployment Mechanism (MDM). The MDM is a three segmented boom on which is mounted the three magnetometer sensors. The boom is stowed for launch and then deployed by real time ground commands to position the magnetometers at the desired orientation.

(h) Nutation Damper. The nutation damper is a passive, self contained device which uses the wave action of liquid freon within a tube to absorb the nutational energy of the spacecraft.

3.3.1.3 Spacecraft Coordinates Definition. The coordinate system for use with the controls subsystem is defined in Section 3.1.

3.3.1.4 Equipment Locations. The control subsystem units are mounted on the spacecraft as shown in Figure 3.3.1-2. The magnetometer deployment mechanism is shown in the stowed and deployed configuration in Figure 3.3.1-2.

3.3.1.5 Attitude Sensors Geometry

3.3.1.5.1 Sun Sensor Field of Views. The mid-range and extended range sun sensor field-of-views provide coverage (ψ output) from 10° to 170° from the +Z axis. The ψ_2 output provides coverage from 15° to 165° from the +Z axis. The field-of-view of each sensor is shown in Figure 3.3.1-3. For the direction of spacecraft rotation indicated in Figure 3.3.1-3, the ψ sensor will see the sun slightly ahead of the ψ_2 sensor in time.

3.3.1.5.2 Star Sensor Field of View. The star sensor field-of-view (FOV) is 24 degrees in elevation with the center of the FOV located at 56.182 degrees from the +Z axis as shown in Figure 3.3.1-4 for the direction of spacecraft rotation indicated in the Figure. The ψ^* sensor slit will see a selected star slightly ahead of the ψ_2^* sensor slit in time.

3.3.2 Units Description

3.3.2.1 Sun Sensors. Three redundant spin scan sun sensors mounted in each of two sensor assemblies are provided to meet the field-of-view requirements. Both sun sensor assemblies are located at the outer periphery of the shelf in order to have an unobstructed field-of-view. The midrange sun sensor assembly provides spin reference pulses (ψ) over a sun aspect angle range of 45 to 135 degrees from the vehicle spin axis and output pulses (ψ_2) for sun aspect angle measurements over a sun angle range of 55 to 125 degrees. The geometry of the midrange sun sensor is shown in Figure 3.3.2.1-1. The extended range sun sensor assembly contains two integral sensors. The upper sensor provides spin reference pulses over the range from 10 to 100 degrees and sun aspect angle pulses from 15 to 90 degrees. The lower sensor provides corresponding pulses over the range of 80 to 170 degrees and sun aspect angle pulses from 90 to 165 degrees, respectively. The geometry of the extended range sun sensors is shown in Figure 3.3.2.1-2.

One set of three sensors is connected to each ADP providing redundancy. The sensor assemblies are passive; there is no command or telemetry interface with them directly nor is any power provided. Electronic processing of the sensor outputs and the telemetered time occurrence of the pulses is performed by the ADP. This is more fully discussed in Section 3.3.2.3.

Only one of the three sensors can be selected by command at a time so that there is no ambiguity in pulse usage. The leading edge of the ADP detected sun pulse is the time reference (0.150 ma threshold level of sensed sun signal). The boresight location is nominally at 352.5 degrees in the spacecraft coordinate system. The relationship of the sun sensor boresight to the sun pulse as detected by the ADP is shown in Figures 3.3.2.1-3 and -4. The ψ_2 pulse is used solely for sun look (aspect) angle telemetry measurements.

The variation of the sun pulse at Earth and at Venus is also shown in Figure 3.3.2.1-3. Shifts in the leading edge of the sun pulse at Venus can be compensated for in the calibration data of the sun pulse.

The relationship of the PSI and PSI2 outputs over the field-of-view provided by the mid-range and extended range sun sensors is shown in Figure 3.3.2.1-5.

3.3.2.2 Star Sensor

3.3.2.2.1 General Description. The star sensor is a spin scan sensor having a dual slit solid-state detector with a fan-shaped FOV. One slit, ψ^*, is aligned in a plane containing the spin axis and the second slit, ψ_2^*, is canted 20 degrees relative to the ψ^* slit. The direction of cant causes the time interval between the ψ^* and ψ_2^* to be the shortest when the star is closest to the +Z axis. The electronics of the ψ^* and ψ_2^* channels are completely independent; providing redundancy. Both channels can operate with either ADP.

The boresight of the star sensor is located 56 degrees from the spacecraft spin axis. The elevation FOV is 24 degrees centered about the boresight axis. This is shown in Figure 3.3.1-4. The azimuth FOV of each slit is 0.5 degree. The star sensor is mounted on the outer periphery of the shelf with the boresight at an azimuth location of 15 degrees from the spacecraft +X axis, measured toward the +Y axis in the spacecraft X-Y plane. A star passing through the center of the FOV will nominally traverse the ψ^* slit at an azimuth angle of 21.22° and the ψ_2^* at 15°. It is provided an unobstructed field-of-view.

The sensor is an integral unit having a single conical sunshade, a common optical system and an electronics housing assembly containing the two redundant electronic modules.

The star sensor is not damaged by sunlight entering the optics. The sun shade design employs a two-stage baffle system to limit the input of stray light. This permits stars to be detected that are 55 degrees or more from the sun and 37 degrees or more from an illuminated Venus, Earth, or moon. These angles include an electronic recovery time of 66 milliseconds (worst case).

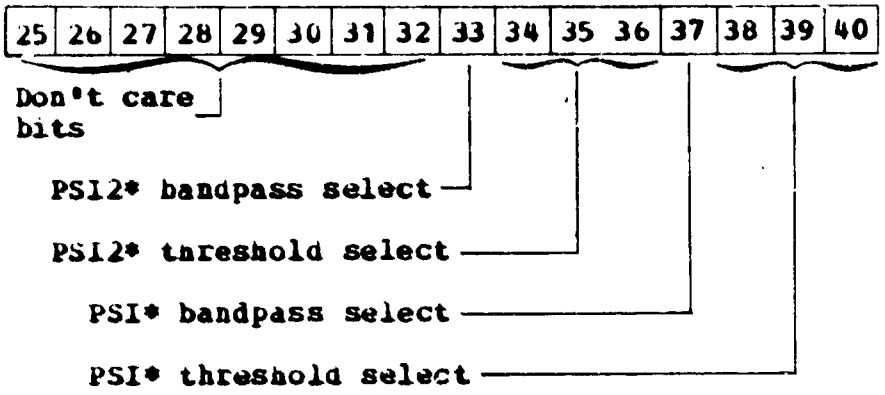
A common optical system provides energy to the two photodetectors of the ψ^* and ψ_2^* channels.

A block diagram of one of the two identical electronics channels is shown in Figure 3.3.2.2-1. The basic function of the sensor is to output a position pulse from each operating channel for use by the ADP whenever the detected amplitude level exceeds the commanded threshold value, thus providing amplitude discrimination against background noise and dim stars. Angle discrimination for the position pulse is provided by the ADP. A star pulse from either channel can be selected as the on-board inertial reference in the ADP or supplied to telemetry for attitude determination. This is more fully discussed in Section 3.3.2.3.

3.3.2.2.2 Command Functions. The two star channels are individually controlled by redundant power on and off pulse commands from COMs 1 and 2, (STR19 or STRA9 for PSI* ON; STR1Ø or STRAØ for PSI* OFF; STR29 or STRB9 for PSI2* ON; STR2Ø or STRBØ for PSI2* OFF). The two channels are not operationally mutually exclusive and thus can be used simultaneously.

The commands control a mechanical relay which connects a fused bus line to the unit regulated power supply. A single redundant quantitative command (STRQ1 or STRQA) controls the desired individual threshold level and bandpass selection in each sensor channel. The quantitative command bit structure is as follows:

Bit location in command word



Bandpass selection: 0 = 19 Hz;
1 = 2 Hz

The threshold selection coding is shown in Reference: Paragraph 1.5.1. Calibration is listed in Appendix A.

The bandpass selection controls the low frequency cut-off to be at 19 Hz or 2 Hz to enhance star detection over the broad spin range. A summary of the pulse commands, command assignments and associated mnemonics is shown in Section 3.3.4. It is recommended that the 19 Hz bandpass be used for all spin rates, based on vendor tests (Reference: Paragraph 1.5.21).

3.3.2.2.3 Telemetry Functions. The star brightness measurement is telemetered directly from each star sensor channel as an 8-bit analog measurement. The measurement is the peak amplitude of any detected light source appearing within a spin angle gate provided by the ADP, and is reset by the leading edge of that spin angle gate when it opens again one spacecraft spin period later. The spin angle gate is 11.25 degrees wide and can be positioned at any azimuth angle by command to the ADP.

The commanded threshold setting for each star sensor channel is also telemetered as an 8-bit analog measurement. The nominal expected voltage for each threshold setting is shown in Section 3.3.4.

The On/Off status and the bandpass selection status of each star channel are provided as bi-level status bits to the DIMs. A temperature sensor, located near the common star sensor detector chip is telemetered. This permits an assessment of the star sensor detection capability at a known telemetered temperature (effect of temperature on performance is discussed in Section 3.3.2.2-5 below). A summary of the star sensor telemetry is shown below.

STAR SENSOR TELEMETRY SUMMARY

TELEMETRY MEASUREMENT	TYPE	DESIGNATOR
PSI* Brightness	Analog	A*1BRH
PSI2* Brightness	Analog	A*2BRH
PSI* Threshold Setting	Analog	A*1THS
PSI2* Threshold Setting	Analog	A*2THS
Star Sensor Temperature	Analog	ASTART
PSI* ON/OFF Status	Bilevel	A*1ONS
PSI2* ON/OFF Status	Bilevel	A*2ONS
PSI* Bandpass State Status	Bilevel	A*1BPS
PSI2* Bandpass State Status	Bilevel	A*2BPS

3.3.2.2.4 Star Pulse Relationships. The position pulse is generated within the ADP when the peak amplitude

of the star sensor video pulse exceeds the commanded threshold level. The leading edge of the position pulse occurs when the trailing edge of the video pulse falls to 25 percent of the peak amplitude. The relationship of the position pulses, star sensor boresight axis and PSI* optical axis is shown in Figures 3.3.2.2-2, 3.3.2.2-3A through 3.3.2.2-3D, 3.3.2.2-4A and 3.3.2.2-4B. These curves were generated from information contained in References 1.5.21 and 1.5.22.

3.3.2.2.5 Detection Performance. Star detection capability, as affected by temperature, selected channel, and selected threshold level, is seen in Figures 3.3.2.2-5A, -5B, and -5C. Detected star brightness will vary as a function of the spacecraft spin rate. This effect is shown in Figure 3.3.2.2-6 and shall be compensated for in telemetry data processing using calibration curves provided with each flight unit. Likewise, there is a predictable and calibratable effect as a function of the command selectable bandpass. This effect is shown in Figures 3.3.2.2-7A and -7B. The variation of star brightness over the elevation field-of-view is shown in Figure 3.3.2.2-8. Reference: Paragraph 1.5.21 states that the maximum usable field-of-view of the star sensor is ±11.5°.

3.3.2.3 Attitude Data Processor (ADP)

3.3.2.3.1 General Description. Each of the two ADPs contains all the logic and signal processing to determine time intervals between sun and star sensor outputs for transmission to the ground station for computation of spacecraft attitude. For onboard functions, azimuth spin pulses are generated for spin period synchronization of star gates, reaction jet firing start signal, despin reference and science roll index pulse reference.

Each ADP interfaces with a dedicated set of three sun sensors. (The two ADPs and the two sets are not cross-strapped.) One of the three sun sensors is command selected based on the spacecraft spin axis to sun LOS pitch (aspect)

angle. The sun aspect angle relative to the spin axis is determined by measuring the time interval between pulses ($\psi - \psi_2$). With the spin period time measurement ($\psi - \bar{\psi}$), the aspect angle time measurement can then be quantized in terms of spin degrees. This then can be converted to obtain the resulting sun aspect angle.

Each ADP can operate with either channel of the star sensor. Normally, more than one star will appear in the FOV of each revolution, therefore angle and brightness discrimination is provided to obtain valid data from a known star. A commandable detection threshold setting in the star sensor will be used to provide amplitude discrimination against background noise and dim stars. A star gate, commandable in spin angle from the sun ψ pulse, will be used in the ADP to angle-discriminate against other bright stars. The star aspect angle relative to the spin axis is determined by the time difference of the ψ^* and ψ_2^* pulses. Time interval measurements from the sun ψ pulse to the star ψ^* pulse or ψ_2^* pulse are also made, thus permitting the use of only one star slit for attitude determination when favorable sun-star geometry exists.

Either the sun or star can be command selected for the onboard inertial reference. The selected input controls a phase lock loop (PLL) which then provides a quantization of the spacecraft spin period. The PLL is able to provide 2^{10} F_S sector pulses to the scientific instruments and 2^{12} F_S to the DCE. A commandable delay from the inertial reference is provided by the Roll Index Pulse (RIP). A roll reference pulse is also provided by the PLL which is phase synchronous with the selected inertial reference.

The ADP also controls all the firing modes of the thrusters via command, and the operation of the latch valves.

3.3.2.3.2 Discrete Command Functions. Redundant discrete commands using COMs 5 and 6 control power application to each of the two redundant ADP units. The ON command only affects the

designated unit and does not turn the other unit OFF. Thus both units can be on simultaneously. This feature is used for the initial spin up maneuver following launch vehicle separation to assure maximum success probability. It is imperative that the onboard command sequencer be programmed for this maneuver to only use COM 5 to turn on ADP No. 1 and only COM 6 to turn on ADP No. 2. These particular command lines are grounded through the separation switch and preclude the ADPs from being commanded on and executing the spin up maneuver due to any failure or spurious pulses that may affect the automatic sequence prior to spacecraft separation.

The ADP will initialize in a random manner at unit turn on except for the thruster control functions. These will initialize as follows:

CIRCUIT	STATUS
JCE Output Buffer	Disabled
JCE Magnitude Count	Zero
Jet Fire Command Latch	Reset
Jet Interlock Command Latch	Reset

A summary of the five ADP discrete commands and their mnemonics is as follows:

COMMAND DESCRIPTION	MNEMONIC	REMARKS
ADP 1 ON	ADP19 ADPA9	Not operative prior to separation
ADP 1 OFF	ADP1Ø ADPAØ	
ADP 2 ON	ADP29 ADPB9	Not operative prior to separation
ADP 2 OFF	ADP2Ø ADPBØ	
Latch Valve #1 Open	VAL19 VALA9	
Latch Valve #1 Closed	VAL1Ø VALAØ	
Latch Valve #2 Open	VAL29 VALB9	
Latch Valve #2 Closed	VAL2Ø VALBØ	

3.3.2.3.3 Quantitative Command Functions. A single but separate quantitative command "ADP configure," is provided to ADP #1 only via COM 5 (ATQØ1 through ATQ12), and ADP #2 only via COM 6 (ATQØA through ATQØL), i.e., there is NO command redundancy to each unit. The quantitative command is structured such that the first four LSBs of the available 16 bit quantity are dedicated for address identification to control 12 internal functions. An example of these functions and bit designations is shown in Table 3.3.2.3-1 for ATQØ1 (ADP#1) or ATQØA (ADP#2). Measurements A and B are equivalent to measurements 1 and 2 respectively, in the MNEMONIC assignments of PC-455. The remaining command functions may be found in PC-455 (Reference: Paragraph 1.5.1). The structure of the quantitative commands will be discussed in detail in the following sections to facilitate use of the ADP.

The command interface is shown in Figure 3.3.2.3-1 (Appendix C).

3.3.2.3.4 Mode Control

3.3.2.3.4.1 Circuit Description. Operational control of the PLL is provided by use of the ADP Mode Select quantitative command that uses Address #2 (ATQ#3 or ATQ#C of the ADP Configure Command). The logic processing diagram is shown in Figure 3.3.2.3-1.

The PLL is a variable frequency pulse generator which maintains both frequency and phase lock with the command selected input reference: sun, star, or simulated SRR. The prime element in the PLL is a voltage controlled oscillator (VCO) whose frequency is controlled by an analog compensation network. Under zero error conditions for the phase locked loop (i.e., phase and frequency lock with the input signals), the VCO will generate a frequency whose value counted down by either 2^{20}, 2^{21}, 2^{22}, or 2^{23} will provide an output pulse F_S essentially coincident with the input pulse. The different countdown chains are command selected ("PLL spin range select": ATQ#3; SR) to provide an operating range consistent with the spacecraft spin rate. For non-zero error conditions, the sensor input pulse and the output F_S are phase compared, and the resulting phase error forces the VCO frequency to be altered until zero phase error condition is achieved. In addition to generating the spin synchronous F_S output, the PLL also generates a synchronous high multiple of 2^{12} F_S and 2^{10} F_S (1024 sector pulses).

The output from the PLL is used to drive one fixed gate, gate-A, which is centered over the F_S pulse and two command variable (ATQ#5 or ATQ#E for ACS Angle Delay Magnitude; ATQ#6 or ATQ#F for Roll Index Delay Magnitude) delayed outputs. In addition, Gate A has two command selectable widths of 11.25 degrees and 45 degrees, both of which are centered over the F_S pulse. The 45 degrees width is only obtained when star acquisition is command selected ("Star

TABLE 3.3.2.3-1

ATQØ1 (ADP #1) OR ATQØA (ADP #2)
QUANTITATIVE COMMAND ASSIGNMENTS

Command Description	Magnitude Data															
	25	26	27	28	29	30	31	32	33	34	35	36	37	38	39	40
Attitude Data	LSB															MSB
Processor Configure	Address				Magnitude											
Measurement Select	0	0	0	0	X	X	X	X	X	X	A	A	A	B	B	B
A) Measurement A Select																
1) SRR - SRR											0	0	0			
2) SRR - Minor F Rate											1	0	0			
3) SRR - RIP											0	1	0			
4) SRR - MIP											1	1	0			
5) SRR - Gated Star B											0	0	1			
6) SRR - PSI 2											1	0	1			
7) SRR - Major F Rate											0	1	1			
B) Measurement B Select																
1) SRR - SRR														0	0	0
2) SRR - Minor F Rate														1	0	0
3) SRR - RIP														0	1	0
4) SRR - MIP														1	1	0
5) SRR - Gated Star B														0	0	1
6) SRR - PSI 2														1	0	1
7) SRR - Major F Rate														0	1	1

Acquisition/Normal"). The two variable delay generators provide output pulses at independently command controlled angle delays from the Fs reference. The desired delay is loaded into a 10-bit counter which is counted down by the 2^{10} Fs pulses.

One generator is used as the Roll Index Pulse (RIP) reference for use by the scientific instruments. The second generator is used to provide the desired roll angle delay for jet firing. This delay is also shared to locate a second gate, Gate B, which provides angle discrimination for star pulses.

Whenever the SRR pulse does not appear within Gate A, a "Missing SRR" pulse is generated at the end of Gate A. The generation of this pulse is not dependent upon the loss of lock enable command bit. The missing SRR pulse serves two functions outside the loss of lock logic; they are:

(a) To generate a "PLL Master Reset Pulse" in the PLL which is used to reset the phase detector, error integrator and PLL timing, so that the PLL frequency is not changed as a result of missing the SRR pulse.
(b) The pulses are accumulated for telemetry in a counter ("Missed SRR Count": AMISSC). This counter is reset each time it is read out to telemetry.

The SRR that controls the PLL can be any one of three inertial pulses (ψ, ψ^*, $\psi 2^*$) or the simulated SRR. The sun (ψ) is preferred because it provides an unambiguous reference. The initial sun acquisition is performed with the sun gate disabled by command, prior to attempting acquisition. This allows the PLL to lock onto the sun wherever it is in the spin cycle. Once locked, the gated sun mode can be used and the loss of lock detection enabled. In the gated mode, the Fs pulse must occur within Gate A. If the pulse is missing for two or more consecutive revolutions, the PLL automatically transfers to

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the simulated SRR. When the sun pulse appears within Gate A for two consecutive revolutions, the reference automatically transfers back to the sun. Any star pulse from either the PSI* or PSI*2 channel can be selected for a reference. All star references are constrained to be used in the gated mode.

The simulated SRR utilizes a commandable spin period (ATQØ7 or ATQØG for MSBs; ATQØ8 or ATQØH for LSBs of PLL spin period magnitude) which can be set from 0.25 msec to 16.384 seconds in 0.25 msec increments. This reference can be selected by either a command (ATQØ3, REF = SIM) or the loss of lock logic (if enabled by command ATQØ3, PL=E) if two consecutive missing SRR pulses occur. The period should be set within 0.78 percent of the actual spin period (at 5 rpm) when operating with the loss of lock enabled. The accuracy is required to maintain the inertial location of Gate A so that the reference pulse will reappear in the gate after the two revolution criteria is satisfied to transfer back to the inertial reference.

A star advance mode ("SRR Advance": SRA = ADV) is provided which permits switching the SRR from sun to star, from star to star, from star to sun, from simulated SRR to sun, or from simulated SRR to star. When transferring to sun, the sun gate must be disabled. When transferring to star, the star gate (Gate B) must be located over the desired star. Upon execution of the SRR advance command, the PLL will reset the phase so that the new reference will occur within Gate A.

A star acquisition mode (ST = N/A) is provided which allows locking to a star pulse in the absence of a sun reference.

Use of the mode control command is described in the following section.

3.3.2.3.4.2 Command Usage Mode control is accomplished using the ADP Mode Select command (ATQØ3 or ATQØC). The use of each bit in the command is described below:

(a) SRR Advance (SRA = SRR/ADV) - The SRR (Selected Roll Reference) Advance is used to switch from one reference to another. Typically reference switching from sun to star, from star to star, from star to sun, from simulated SRR to sun, or from simulated SRR to star (Star Acquisition) is accomplished by this command bit. The "SRR Advance" bit must be set to logical zero once proper transfer has been accomplished.

Since the "SRR Advance" can effect various modes of transfer, the interdependence of this bit with other logic bits of (ATQ03 or ATQ0C) requires careful elaboration. Normal transfer of reference, such as, sun to star or star to star is accomplished by positioning Gate B over the desired star that is to be the next primary reference. After receipt of the "SRR Advance" CMD bit, the phase of the PLL is reset so that the desired star NOW appears in Gate A and becomes the reference for the PLL. The other additional modes of transfer can best be described in subsequent paragraphs in conjunction with other CMD bit usage.

(b) Enable/Disable Sun Gate (SUG = E/D) - The Enable/Disable Sun Gate accomplishes a two-fold purpose: a) for a logical 1 (enable), it allows the loss of reference detector ("missing SRR" counter) to telemeter (AMISSC) the number of missed sun pulses during the sample interval; i.e., between leading edges of successive occurrences of ADP Status Word envelopes. (The interval time period depends on bit rate and selected TM format - refer to para. 1.5.2 for details). If two or more consecutive missing sun pulses (not within Gate A) do occur and assuming "PLL loss of lock" is enabled, it allows SRR transfer to simulated SRR;

b) for a logical 0 (disable), it allows normal sun acquisition.

(c) Star Acquisition/Normal (ST = N/A) - Star acquisition allows phase locking to a star pulse (likely used in the absence of a sun pulse). Normally, the simulated SRR signal will have been selected as a PLL reference and the commanded spin rate will be chosen as close to the actual spin rate as possible. However, some small frequency difference may exist which allows star gates A and B, which are now phase locked to the simulated SRR, to slip in phase with respect to the actual stars, making it impossible to keep a star within Gate B.

(d) SRR/Simulated SRR Select (part of REF = SUN/STR/SIM) - For a logical 1, the primary input to the phase lock loop will be the SRR (ψ, ψ" or $\psi 2$"), unless an automatic transfer is accomplished by two consecutive missing SRR pulses. In the latter case (PLL loss of lock enable = 1) the simulated SRR is selected. For a logical 0, the only input to the PLL is the simulated SRR irrespective of all other conditions. Note: Prelaunch tests on one Flight Model ADP have shown that the SRR to F_S delay at 4.7 rpm had changed from typically 2.6 milliseconds with use of a real SRR, to typically 4.85 milliseconds with use of the simulated SRR.

(e) Sun/Star Select (part of SUN/STR/SIM) - Either the selected ψ (logical "1") or the Gated Star A (logical "0") is selected by the "sun/star select" (serial command bit). The resulting signal is the Selected Roll Reference (SRR).

(f) Sun Sensor Select (SRR = N/U/L/S) - Signals ψ and ψ_2 are received, each on twisted and shielded pair, from each of three sun sensors, midrange, extended range upper and extended range lower.

These six signals are processed by six identical sun sensor electronics subassemblies. The outputs of these electronics are input to a digital multiplexer where one ψ and ψ_2 pair (both from the same sensor) are command selected for use by the ADP.

(g) Star Gate B; Channel 1/2 Select (SGB = 1/2) - One of the two star processor channels (ψ^* or ψ_2^*) within the ADP can be chosen by "star Gate B channel select" (logic 1 = ψ^*). The selected star sensor channel would then be for: (a) telemetered time interval measurement (SRR to Gated Star B ATTM) between the SRR and that star that is the first chronologically in Gate B that exceeds the commanded threshold; (b) telemetered star brightness measurement (A*1BBM or A*2BBM) of the brightest star appearing in Gate B before the telemetry is sampled; and (c) use in the SRR Advance Logic for transferring the SRR to the first star appearing in Gate B that exceeds the CMDED threshold level.

(h) Star Gate A; Channel 1/2 Select (SGA = 1/2) - One output (ψ^* or ψ_2^*) from the two-channel star processor is chosen by "star Gate A channel select" (logic 1 = ψ^*). The selected channel is gated by Gate A from the PLL to become "gated star A." Gated Star A is sent to the sun select/star select logic and the command and TM subunit for attitude measurements ("SRR to SRR," when the SRR is that star in Gate A).

(i) PLL Loss of Lock Enable/Inhibit (PL = E/D) - The "loss of lock" logic should more properly be labeled "loss of SRR" logic, as that is exactly what it detects.

The last commanded state of the "loss of lock" logic is telemetered as the bilevel signal" PLL Loss of Lock Enable/Inhibit" (ALOLES).

The actual state of the PLL w.r.t. lock on the SRR is telemetered as the bilevel signal "PLL Loss of Lock" (ALOCKS). After being enabled (logical "1") by command, this logic will detect and output a signal to the reference select logic, the JCE, and telemetry (ALOCKS will = 0) whenever two or more successive SRR pulses are found to be outside of Gate A. This loss of lock will cause automatic Roll Reference transfer to the simulated SRR, and inhibit the jets from being fired. If, while the PLL loss of lock logic is still enabled, and two or more successive SRR pulses were to reappear within Gate A, the "PLL loss of lock" signal would return automatically to its normal (in-lock) state. (The requirement for two or more successive reappearances prevents a single spurious noise pulse on the SRR line from reselecting the SRR as the PLL reference.

If inhibited (logical "0") by command, the "PLL Loss of Lock" signal (ALOLES) will indicate an apparent "in-lock" state (logical "1"), whether or not the PLL is truly locked on the SRR.

(j) PLL Spin Range Select (SR = 4-71) - The PLL VCO output (nominal operating frequency) is divided down by a counter with variable length. The PLL Spin Range Select command bits select the appropriate variable divider output to give a total countdown chain length versus spin range as shown below:

COUNTDOWN CHAIN LENGTH	PLL OPERATING RANGE
$+ 2^{23}$	4.0 - 8.85 rpm
$+ 2^{22}$	8.0 - 17.7 rpm
$+ 2^{21}$	16.0 - 35.4 rpm
$+ 2^{20}$	32.0 - 70.8 rpm

These operating ranges have overlap designed to avoid dead zone regions and also to allow some flexibility with respect to the transmission of a new PLL spin range select command.

Nominal range switching would be done only associated with substantial spin speed changes during which the PLL loss of lock would be inhibited, as it cannot track continuous burn spin speed changes. After such a change, the PLL would require approximately 70 revolutions minimum, as a rule of thumb, before it has settled and is ready for lock-on to an SRR.

3.3.2.3.5 Maneuver Control

3.3.2.3.5.1 Circuit Description. Attitude, velocity and spin speed control are provided through the use of a combination of axial and radial thrusters. Four radial, one forward and one aft axial thrusters are common to the Orbiter spacecraft and Multiprobe spacecraft. The Orbiter has an additional aft axial thruster, providing redundancy and additional flexibility.

The jet controlling logic functions are generated within each of the ADPs. However, only one driver module is provided for each thruster or latch valve. Each ADP contains half of the required drivers. Either ADP can operate any thruster even though the driver is physically located in the other ADP. Each driver is connected to the spacecraft essential bus and separately protected using 5 amp fuses. There is no need to activate the drivers by command; they are permanently activated. The logic diagram of the thruster control functions is shown in Figure 3.3.2.3-1.

The drivers, thrusters and latch valves are interconnected so that the ADP that controls the latch valve also controls all the thrusters that are supplied hydrazine by that latch valve. A complete loss of an ADP will still permit maneuvers to be performed.

In the continuous firing mode the firing duration is controlled by way of the JET countdown quantitative command that uses Address #3 (ATQ04 or ATQ0D of the ADP configure command) which permits up to 2096.64 seconds in 0.512 second increments.

The pulsed firing mode has two commandable pulse widths, 128 ms and 512 ms. The roll oriented firing location is selected by using the ACS variable delay output from the roll reference generator (via ACS Angle Delay quantitative command that uses Address #4 (ATQ05 or ATQ0E of the ADP configure command)). The occurrence of the delayed output initiates the start of the fire pulse. In the command selected normal mode, a fire pulse will be generated every revolution. In the alternate fire mode (2 pulse firings per spacecraft revolution), a second fire pulse will be generated 180 degrees azimuthal with respect to the first pulse. The duration of the pulsed firing mode can be controlled by pulse count or time duration. With pulse count selected, the magnitude command converts to a pulse counter which then permits up to 4095 pulses in one pulse increments.

After the firing logic has been selected, the actual firing is initiated by commanding the interlock function (via JET Fire Interlock quantitative command that uses Address #10 (ATQ11 or ATQ0K of the ADP configure command)) followed by the fire command (via JET Fire quantitative command that uses Address #11 (ATQ12 or ATQ0L of the ADP configure command)). There is no time limitation after the interlock command but the command sequence should group these two commands together. The firing sequence will terminate when the magnitude decrements to zero count. The JCE output buffers are automatically commanded off and the magnitude counter will underflow, terminating with a count of 4095 (indicated in telemetry by "JCE Countdown" AJHAGC - as twelve "ones").

A duty cycle detector is activated during all pulsed firing modes which monitors the JCE buffer

output to the solenoid drivers. If the duty cycle exceeds the limits shown in Figure 3.3.2.3-2, firing will be terminated. The detection point is a function of spin speed and will exclude the normal operation of the wide pulse mode above 20 rpm. This feature cannot be disabled by command.

One other protective feature is provided for use with the pulse mode. Since the roll angle delay requires the PLL to be locked for accurate firing, a PLL loss of lock detection can be enabled which will terminate any maneuver. To use this capability, the PLL loss of lock is enabled and the PLL reference should be operated in the narrow gate mode. If two successive SRR pulses occur outside Gate A indicating the PLL is not locked, the out of lock detector will terminate the sequence. This feature should be disabled for spin speed maneuvers since the PLL cannot track large spin speed changes and will terminate the spin sequence if loss of lock is detected.

3.3.2.3.5.2 Command Usage. Operational use of the thrusters is accomplished by use of the commands described below:

(a) Jet Control (Quantitative command that uses Address #1 (ATQ#2 or ATQ#B of ADP configure command)).
(1) Jet Select (R1, R2, R3, R4, A5, A6 or A7; each = E/D) - seven bits of the magnitude data are dedicated to selecting any set of four radial and three axial thrusters, by commanding a logical "1" state for each thruster intended to be active. Any or all thrusters can be selected; however, during normal mission operations, a maximum of two thrusters will normally be selected. PC-455 (Refer to Paragraph 1.5.1) shows the specific magnitude data bit associated with a given thruster. Possible thruster options per maneuvering function is specified below:

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FUNCTION	SELECTED FIRING MODE	THRUSTER SELECTION
Spin Speed Change	Continuous or Pulse	2R+4R (Spin-down) 1R+3R (Spin-up)
Velocity Change	Continuous Pulse Pulse (Twice Spin Frequency)	5A+6A+7A+(6A•7A) (1R•2R)+(3R•4R) 5A+6A+7A
Attitude Change	Pulse	5A+6A+(5A•6A)+7A (1R•4R)+(2R•3R)

(2) Spin Rate Detector Enable/Inhibit (SRD = E/D) - Onboard detection is provided which will terminate any firing sequence when specific limits are detected and a logical 1 is commanded for this magnitude bit. A spin rate detector measures the spin period and will terminate the maneuver if the spin frequency is less than 3.7 rpm or over 67 rpm. This capability is provided by a special counter which measures the time interval between two successive SRR pulses using a 15.625 Hz clock rate. A failure is indicated if the detected count is less than 14 or greater than 254. If the gated reference mode (i.e., "loss of lock" logic is enabled) of the PLL is being used during the first firing sequence, firing will be terminated if the SRR moves outside Gate A. The loss of SRR pulses will also be detected as a low rpm condition. This detection capability can be disabled by command (logical "0") but this should only be done for the initial spinup maneuver so that the spin rate can be set above the lower threshold.

(3) Continuous/Pulse Fire Select (F = P/T) - The thrusters will be fired in two modes of operation, namely: a) continuous (logical "1") and, b) pulse. In the continuous mode

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the thrusters is fired continuously until terminated by a proper completion of the count sequence or automatic override features previously discussed.

(4) Pulse/Time Count (C = P/T) - The jet countdown magnitude (contained in quantitative command that uses Address #3 (ATQ#4 or ATQ#D of ADP configure command)) initializes a 12 bit counter (AJMAGC), that is subsequently counted down to zero based upon the logical state of this bit. The countdown counter will count elapsed time when either (or both) the continuous fire mode or the time count mode is selected, and counts pulses when both the pulse fire mode and the pulse count mode are selected. (If the alternate mode is selected and pulses are to be counted, each of the two pulses per revolution is counted). Thus, if both the pulse fire and time count modes are commanded the pulse duration of the firings is still controlled by the pulse width selection signal, but the countdown counter counts elapsed time from the start of the first pulsed firing. In pulse mode, the firing pulse can be commanded to occur at any spin azimuth point using the ACS angle delay command.

(5) Pulse Width Select: (512/128 ms; P = 2/8 respectively) - The firing signal to the solenoid drivers in the pulse mode of operation will either be 512 milliseconds (logical 1) or 128 milliseconds (logical 0).

(6) Normal/Alternate Fire Mode (FM = N/A) - In the normal fire mode (logical 1) a single fire pulse will be generated every spin period at the commanded angle delay. In the alternate fire mode (logical 0) a second firing pulse is generated 180 degrees after the first pulse in each revolution. The alternate fire mode provides the capability to translate

the spacecraft axially in a vernier increment, i.e., pulsed, while keeping nutation at a small level. The first pulse firing to occur after the Alternate Fire Mode is selected will correspond to the first occurrence of the SRR pulse (0° azimuth angle), even if Mode selection occurs on-board before occurrence of the 180° azimuth angle event.

(b) Jet Countdown (Quantitative command that uses Address #3 (ATQØ4 or ATQØD of ADP configure command)). Twelve bits of magnitude represent 4095 pulses or 2096.64 seconds maximum count. In the time count mode, a clock frequency of 1.953 Hz or 0.512 seconds per bit is used to countdown the elapsed time as represented by the contents of the 12-bit jet countdown register, that has been loaded by command. In the pulse mode, the same register is used, and decrements for each fire pulse generated.

(c) ACS Angle Delay Magnitude (Quantitative command that uses Address #4 (ATQØ5 or ATQØE of ADP configure command)). The JCE jet fire angle is determined by the ACS delay generator. The ACS delay pulse is generated by a 10-bit (360°) delay counter. The command is the binary magnitude of the desired delay with a quantization of 0.351 degree per bit. This counter is initialized to the commanded "ACS delay" angle by the PLL output signal F_s. The counter is then counted down by the "2^{19} F_s" clock pulse until it underflows. The counter underflow is the "ACS delayed pulse," and initiates thruster firing in the pulse mode. The ACS delay generator is shared with the star Gate B function. The angle delay locates the starting edge of Gate B (Gate A is centered on F_s). The ACS delay quantity has a one count bias, i.e., the actual delay is one count greater than the commanded

value. Therefore, to accomplish a zero degree angle delay, a delay of 360° (ten "ones") must be commanded into the countdown register.

(d) JCE Buffer Output Enable (Quantitative command that uses Address #8 (ATQØ9 or ATQØI of ADP configure command)). The +5 volt and the +15 volt to the JCE logic circuitry is controlled by the ADP power on/off discrete command. However, an additional line switch provides command control of the +15 volt bus to the JCE output buffers. The line switch is latched on (and indicated by AJCE1S or AJCE2S, corresponding respectively to the use of ADP1 or ADP 2), whenever the "JCE buffer output enable" command is sent. The latch is reset to logical "0" and the switch turned off whenever any of the six disable conditions listed below occurs:

(1) Countdown counter (AJHAGC) reaches zero count and underflows (normal operation)
(2) ADP initialization (insures that the JCE buffer output is off at ADP power turn-on)
(3) Receipt of the "JCE buffer output disable" cmd (see item (e) ahead)
(4) "PLL loss of lock" condition when enabled
(5) "Over/under spin rate" condition when spin rate detector is enabled
(6) "Duty cycle failure" condition when operating in pulse firing mode

(e) JCE Buffer Output Disable (quantitative command that uses Address #9 (ATQ1Ø or ATQØJ of ADP configure command)). This command will terminate any thruster firing in progress irrespective of the state of the countdown counter or the onboard JCE failure protection. The command disables the JCE output buffers and will reset the fire and interlock

logic. The counter will indicate the incompleted count of the maneuver.

(f) Jet Fire Interlock (Quantitative command that uses Address #10 (ATQ11 or ATQ#K of ADP configure command)). The jet fire interlock CMD controls the state of the JCE Fire Enable/Disable logic. This Enable/Disable logic does not allow the jets to fire (as indicated by AJCEFS - JCE Fire Enable Status remaining a logic "0") until both the jet fire interlock command and the jet fire command have been received in that order without an intervening JCE Buffer Output Enable command. The interlock command sets a flip flop which then permits another flip flop to be set when the fire command is received. Should the JCE Buffer Output Enable command be received after the jet fire interlock command but before the jet fire command, both flip flops will be reset and the jet fire interlock command must be transmitted again followed by the jet fire command before the jets will begin to fire. There is no time lapse requirement between the jet fire and jet fire interlock command.

(g) Jet Fire (Quantitative command that uses Address #11 (ATQ12 or ATQ#L of ADP configure command)). This command starts the thruster firing sequence if proper initialization described above is true.

3.3.2.3.6 Attitude Measurements

3.3.2.3.6.1 Circuit Description. A logic diagram of the attitude measurement function is shown in Figure 3.3.2.3-1.

A total of seven different time separation measurements may be computed, however, only two may be selected at a time, in any combination from the table below:

SELECTABLE MEASUREMENT (ATTM)	ADDRESS
SRR - SRR	0 0 0
SRR - Minor Frame Rate Signal	1 0 0
SRR - RIP (Roll Index Pulse)	0 1 0
SRR - MIP (Master Index Pulse)	1 1 0
SRR - Gated Star B	0 0 1
SRR - PSI 2	1 1 1
SRR - Major Frame Rate Signal	0 1 1

The signal SRR (selected roll reference) may be command selected to be: SUM (ψ), Star Channel (ψ^*), Star Channel 2 (ψ_2^*), or Simulated SRR, while operating in the so-called "free-run" mode. Note that for the SRR-Gated Star B measurement, Gated Star B may be either ψ^* or ψ_2^*.

The actual measurements are computed in binary counters using a clock frequency of 4 kHz, thus giving a measurement resolution of 0.25 milliseconds. After the measurement is completed, the counter is frozen and cannot be reset by the SRR until the 16-bit magnitude along with the measurement identification, A or B, (equivalent to 1 or 2 respectively in the mnemonic assignments of Reference: Paragraph 1.5.1) and the command selected addresses are parallel transferred into the serial TM registers at the leading edge of the first of the three readout envelopes. This clears the counter and enables the start of a new measurement. The 24 bits of data in the TM register is then shifted out serially least significant bit first in three groups of 8 bits. The readout format is shown in Table 3.3.2.3-2. This same measurement will be continued to be read out until the on-going measurement is complete and the transfer and readout process is repeated.

A summary of the attitude measurement processing is as follows:

(a) The selected roll reference pulse generates a "start" pulse which:
 (1) Resets the ATTM counter to zero
 (2) Enables the 4 kHz clock pulse to the ATTM counter

TABLE 3.3.2.3-2

ATTM TM FORMAT

Word No.	Ref. Desig.	LOC Code	Telemetry Description	TM Word 0	1	2	3	4	5	6	7
1	ATTM1Z (Redun.)	6SD06 7SD05 (For Minor Frame)	Attitude Measurement	LSB							
			Measurement Data (8 LSB's of 16 bit word)	0							7
			Measurement Data (8 MSB's of 16 bit word)	8							15
			Range: 0 sec to 16.38 sec (0 to 65535 counts)								MSB
			Resolution: 0.25 msec								
2	ATTM2Z	6SD07 7SD06 (For Sub-com)									
				0	1	2					
3	AM1ADS		Measurement A Address								
			SRR - SRR	0	0	0					
			SRR - Minor Frame Sync	1	0	0					
			SRR - RIP	0	1	0					
			SRR - MIP	1	1	0					
			SRR - Gated Star B	0	0	1					
			SRR - PSI 2	1	0	1					
			SRR - Major Frame Sync	0	1	1					
							3		5		
	AM2ADS		Measurement B Address								
			SRR - SRR				0	0	0		
			SRR - Minor Frame Sync				1	0	0		
			SRR - RIP				0	1	0		
			SRR - MIP				1	1	0		
			SRR - Gated Star B				0	0	1		
			SRR - PSI 2				1	0	1		
			SRR - Major Frame Sync				0	1	1		
										6	7
	ATTMSS		Data Measurement A = 1, B = 0								
			Unused Bit								X

(b) The selected "stop" pulse (SRR, Minor Frame Rate, Major Frame Rate, HIP, HIP, Gated Star B, or Ψ_2):
 (1) Disables the 4 kHz clock pulse to the ATTM counter
 (2) Sets a "measurement start/stop control" flip-flop which freezes the ATTM counter until the telemetry subsystem is ready to read the measurement

(c) The first of the three ATTM read envelopes will:
 (1) Transfer the measurement, if complete, to the TM register
 (2) Enable the start of new measurement at the next SRR
 (3) Start the TM readout process

The synchronization of the attitude measurements with telemetry will insure that information will only be loaded into the TM register when the previous measurement has been completely read out at least once. If no stop pulse is detected before a second selected roll reference pulse is generated, the counter will be reset and the same measurement will be started again.

The serial TM register shall be shifted out during 3 TM read envelopes; 3 subcommutation envelopes 3 subcommutation envelopes in the subcommutation mode (S MODE) or 3 minor frame envelopes in the minor frame mode (M MODE). In each mode, the envelopes are counted modulo 3 and the data are output to two separate DIMs providing redundancy.

If a new measurement is not ready when the first envelope of the next group arrives, the old measurement is shifted out again (recirculated). The repeated measurement word can be identified by Sel A/B (equivalent to 1/2 respectively), bit; Sel A/B is toggled as each new measurement is loaded and the previous measurement is read out in TM. The state of this bit remains unchanged if a measurement is repeated.

3.3.2.3.6.2 Data Valid Determination- The attitude measurements can be telemetered in either the subcom(s) mode or in a minor frame format (M mode). If data is being telemetered in a minor frame format, the subcom data is invalid. The onboard processing is as follows:

S mode is arbitrarily assumed at the start of each major frame. M mode is then entered and S mode is inhibited if a single minor frame envelope is detected. If M mode is envoked, subcom envelopes will be ignored for the remainder of the major frame. If M mode is not entered, the subcom data will remain valid. A summary of this process is as follows:

(a) If no minor frame envelopes occur, the data in the subcom words will be valid.

(b) If subcom envelopes and minor frame envelopes both occur within the same minor frame, the minor frame data will be valid and the subcom data will be all logical zero's except for the first subcom word in each major frame.

(c) When switching from the subcom only sampling format to the minor frame sampling format, the minor frame data may be out of sequence until the next measurement is completed.

(d) When switching from the minor frame sampling format to the subcom only format, the subcom data will be all zero's until the start of the next major frame, or until the next measurement is completed, whichever occurs last. If the new measurement occurs last, the subcom words following the first major frame pulse may be randomly permutated until the new measurement is completed.

(e) If any of the selected measurement pulses disappear, the ATTS will "hang up" and the previous measurement will be repeated.

3.3.2.3.6.3 Unique Attitude Measurements. Several types of measurements are somewhat unique and bear mentioning.

(a) The SRR-Minor Frame Rate or Major Frame Rate measurement. Whenever the spacecraft rotor rate is higher than the minor frame rate, two or more SRR pulses may be expected between two minor frame rate pulses. The ATTM logic is implemented to calculate the smallest time separation between these two pulses. This is accomplished by the logic automatically resetting the ATTM counter whenever an SRR pulse occurs, regardless if a measurement is in progress or not.

(b) Whenever an inertial reference is lost and the Simulated SRR becomes the SRR, all measurements will be computed with the Simulated SRR acting as the "start" pulse. This insures that such vital science data as SRR-RIP signal, or SRR-Major Frame Rate signal will not be lost. The drift rate between the Simulated SRR and the actual rotor rate can be checked by SRR-Ψ_2 or SRR-Gated Star B telemetry measurement.

3.3.2.3.7 Roll Index Delay Magnitude (Quantitative Command that Uses Address #5 (AT0Ø6 or AT0ØF of ADP Configure Command)). The Roll Index Pulse (RIP) is generated by a 10-bit (0° to 359.65°) counter in exactly the same manner as is the ACS delayed pulse. The counter is initialized to the commanded roll index delay magnitude by P_s. The command is the binary magnitude of the desired delay with a quantization of 0.351 degree per bit. The RIP is transmitted to science, to the command and telemetry subunit for attitude measurements and to the DCE.

Unlike the ACS delay, the actual RIP delay is equal to the number of magnitude counts commanded.

3.3.2.3.8 PLL Spin Period Magnitude - MSB (Quantitative Command that Uses Address #6 (ATQ07 or ATQ0G of ADP Configure Command)). Eight bits of this magnitude command are used to control the eight most significant bits of a 16-bit counter that control the simulated SRR generator. The simulated SRR generator outputs pulses at a period proportional to the PLL Spin Period Magnitude serial commands. The 16-bit command words are loaded into a counter which is counted down at 4 kHz. Upon reaching zero, the counter outputs a pulse and reloads itself (in the "free-run" mode) with the 16-bit command word and continues to count down).

The output frequency of this generator can be made nearly equal to the actual rotor spin rate within 1/2 of the 0.25 msec resolution. The phase of the output signal is made equal to the PLL output, F_S, by using F_S to load the counter prior to manual or automatic switching to the sim SRR as the PLL reference. Thus, negligible phase transients are introduced to the PLL when switching to Simulated SRR as a PLL reference.

The generator is allowed to "free run" (not being loaded by F_S but by the Simulated SRR pulse itself) whenever the Simulated SRR is used as the PLL reference, which occurs whenever a "Loss of Lock" condition exists or whenever so selected by the SRR/Simulated SRR select command bit.

3.3.2.3.9 PLL Spin Period Magnitude - LSB (Quantitative Command that Uses Address #7 (ATQ08 or ATQ0H of ADP Configure Command)). Eight bits of this magnitude command are used to control the eight least significant bits of a 16-bit counter described in the preceding paragraph.

3.3.2.3.10 ADP Status Measurements

3.3.2.3.10.1 Circuit Description. There are 79 bits of status information within the ADP that is formatted into ten serial digital TM words, consisting of 8 bits per word. Most of the ADP configure commands (ATQ01 through ATQ12) status bits are telemetered as part of these ten words,

allowing for command verification. In addition, the following data is contained within these ten TM words:

(a) Missed SRR Count (4 bits)
(b) PLL Loss of Lock Status (1 bit)

The data format for each of these words is contained in PC-454 (Reference: Paragraph 1.5.2).

The ADP Status TM can be read out in either a subcom mode or in a minor frame mode. In the subcom mode, the ten words are read out one word per minor frame for ten consecutive minor frames. In the ACS format, or minor frame mode, the words are read out in ten consecutive word slots, all ten words being read twice per minor frame.

The interface with the telemetry subsystem is done by way of a three signal interface. The ADP receives a TM read envelope and TM read clock and outputs ADP status data serially, one bit per TM read clock, 8 bits per read envelopes. The ADP has redundant output buffers to each DIM which precludes a single failure in one DIM from destroying the data read into the redundant DIM.

3.3.2.3.10.2 ADP Status Data Valid Determination. The ADP status data can be telemetered in either the subcom mode or in the minor frame format. If data is being telemetered in the minor frame format the subcom data is invalid. The onboard process is as follows:

A unique read envelope is sent to the ADP for each of the two formats. This allows the ADP to determine which format has been selected and output the data on a single serial interface. A summary of this process is as follows:

(a) If no minor frame envelopes occur, the data in the subcom words will be valid.
(b) If subcom envelopes and minor frame envelopes both occur within the same minor frame:

(1) The data appearing in the subcom words will be word 1 of the ten word sequence.
(2) The data in the minor frame words will be valid.

(c) When switching from the subcom only sampling format to the minor frame sampling format, the minor frame data will be valid within the first complete minor frame.

(d) When switching from the minor frame sampling format to the subcom only format, the data will be valid within the first complete major frame.

3.3.2.3.11 Bilevel Telemetry. Each ADP provides two bilevel bits directly to separate DIM channels as follows:

TM DESCRIPTION	MNEMONIC
ADP No. 1 ON/OFF Status	AADP1S
ADP No. 2 ON/OFF Status	AADP2S
JCE No. 1 Buffer Output Status	AJCE1S
JCE No. 2 Buffer Output Status	AJCE2S

3.3.2.4 Despin Control Electronics (DCE)

3.3.2.4.1 General Description. The DCE contains the processing electronics which uses inertial and platform rate information to generate motor drive signals for closed loop azimuth pointing control of the HGA. The despin electronics responds to position and rate commands to control the antenna pointing.

The DCE also provides elevation open loop control of the HGA by sending rate pulses to the stepper motor of the HGA positioner. Command capability exists for controlling the rate and polarity of the stepping. There is no direct position control capability.

Two independently rate command controlled reference pulses (RAM VPD and NADIR VPD) are

provided to selected scientific instruments. A variable phase delay generator associated with each provides an output pulse once every rotor spin period that is phase referenced to the Roll Index Pulse (RIP) received from the ADP. The initialization and phase change of each signal is independently command controlled. The NAM VPD pulse will be programmed to occur when the velocity vector crosses the spacecraft X-Z plane such that the projection of the velocity vector on the X-Y plane is coincident with the +X axis. The NADIR VPD pulse will be programmed to occur when a vector pointing in the NADIR direction crosses the spacecraft's X-Z plane such that the projection of this vector on the X-Y plane is coincident with the +X axis.

3.3.2.4.2 Discrete Command Functions. Redundant discrete commands from COMS 1 and 2 control power application to each of the two redundant DCE units. The ON command to a designated DCE will automatically turn the other unit off, thus it is not possible to have both units on simultaneously. A "DCE 1 and 2 OFF" command is the only command that sends an "OFF" signal to both units simultaneously. The BAPTA (Bearing Power Thruster Assembly) motor drive control within each DCE is individually enabled by separate and redundant discrete commands. However, the motor drive power is interlocked with the processing electronics of the associated DCE such that the drive power cannot be on without that corresponding DCE being on. Likewise, commanding one DCE OFF will also turn the corresponding BAPTA motor drive off via the relative rate failure detection logic. However, the preferred sequence, in order to monitor load change, would be to command the BAPTA drive off first with the "MOTOR DRIVE 1 and 2 OFF" command then to turn the DCE OFF.

The DCE will initialize in a random manner at turn on. Quantitative commands as described in the next section should then be commanded to initialize the unit in a known configuration.

A summary of the DCE discrete commands and mnemonics are as follows:

COMMAND DESCRIPTION	MNEMONIC
DCE 1 ON/2 OFF	DCE 19 DCE A9
DCE 2 ON/1 OFF	DCE 29 DCE B9
DCE 1 and 2 OFF	DCE 1Ø DCE AØ
Motor Drive 1 ON	MTR 19 MTR A9
Motor Drive 2 ON	MTR 29 MTR B9
Motor Drive 1 and 2 OFF	MTR 1Ø MTR AØ

3.3.2.4.3 Quantitative Command Functions. A single but separate quantitative command, "DCE CONFIGURE," is provided to DCE #1 only via COM 1, and to DCE #2 only via COM 2, i.e., there is no command redundancy to each unit. The quantitative command, using 3-bits of the available 16-bit quantity for address identification, controls six internal functions. These functions and bit designations are shown in Table 3.3.2.4-1.

Since the DCE is dependent upon the inertial reference from the ADP, it should not be commanded ON until after the ADP has achieved phase lock. After the DCE has been turned on, the following mode command should be sent to initialize it in a known configuration (HGA positioner OFF, phase reset enabled, and VPD's reset):

DCE MODE COMMAND: (DCEQ6 for DCE #1; DCEQF for DCE #2)

1	0	1	0	0	0	0	0	0	0	0	0	1	1	1	0
ADDRESS															

Unless the other functions controlled by the quantitative commands are to be immediately configured, they also should be set to a known value, e.g., all zeros.

The command interface plus internal functions are shown in Figure 3.3.2.4-1 (Appendix C).

3.3.2.4.4 Despin Control. Despin control is accomplished by closed loop control of the BAPTA motor with signals from the DCE.

The ADP supplies the DCE with the inertial reference pulse (F_S) and a phase locked binary multiple frequency 2^{12} F_S that quantizes the spacecraft circumference into 0.088 degree increments. A shaft angle encoder (SAE) in the BAPTA produces the master index pulse (MIP or F_Z pulse) and a binary multiple of the MIP (2^7 F_Z) by sensing the relative position between the spinning rotor and the despun platform.

The DCE processes the F_S, MIP and 2^7 F_Z and generates a despin position error which then provides continuous torque commands to the BAPTA torque motor. This process is illustrated in Figure 3.3.2.4-2 for one complete revolution of the spacecraft spinning section. The three position drawing depicts a steady-state null condition with the HGA pointing towards the earth line of sight. The DCE produces a position motor torque voltage that maintains the MIP in phase lock with a delayed filtered inertial reference signal (F_S'). The delay of F_S' from the ADP F_S signal is generated within the DCE from the commanded azimuth offset value (DCEQ5 or DCEQE - Azimuth Position). The commanded offset value is relative to the selected roll reference (SRR) in the ADP. The offset command capability permits the HGA to be pointed to any azimuth position with a resolution of 0.0879 degree, or more precisely $360^0 \div 4096$ (2^{12}), but with a deadband of $360^0 \div 2^7$.

Prior to changing to a new inertial reference in the ADP, it is necessary to command "Phase Reset Disable" (DCEQ6 or DCEQF: Mode CMD) to prevent antenna pointing transients. Upon completion of the transfer, a new azimuth offset value which compensates for the new inertial reference location must be commanded. "Phase Reset Enable" can then be commanded to return pointing control

TABLE 3.3.2.4-1

DCE CONFIGURE QUANTITATIVE COMMAND FORMAT

Mnemonic	Command Description	Magnitude Data 25 26 27 28 29 30 31 32 33 34 35 36 37 38 39 40
	Despun Control Electronics Configure:	LSB MSB Address Magnitude
DCEQ1 or DCEQA	Azimuth Pointing Slew Sign/Magnitude Notation MSB = Sign; 0 = Rotation in direction of Rotor Spin. 1 = Despin Direction. Quantization = 0.04024 deg. min.	0 0 0 LSB MSB Magnitude
DCEQ2 or DCEQB	RAM Variable Phase Delay Rate Sign/Magnitude Notation MSB = Sign; 0 = positive (advances RAM pulse in direction of spin) 1 = negative. Quantization = 0.04024 deg. min.	1 0 0 LSB MSB Magnitude
DCEQ3 or DCEQC	NADIR Variable Phase Delay Rate Sign/Magnitude Notation MSB = Sign; 0 = positive (advances NADIR Pulse in direction of spin) 1 = negative Quantization = 0.04024 deg. min.	0 1 0 LSB MSB Magnitude
DCEQ4 or DCEQD	HGA Elevation Positioner Slew Sign/Magnitude Notation MSB = Sign; 0 = Rotates Dish towards +Z axis. 1 = Rotates Dish towards -Z axis. Quantization = 0.00975 deg. min.	1 1 0 LSB MSB Magnitude
DCEQ5 or DCEQE	Azimuth Position 2's Complement Notation MSB Sign; 0 = Rotation in direction of Rotor Spin 1 = Despin Direction Quantization = 0.08793 Deg.	0 0 1 LSB MSB Magnitude X
DCEQ6 or DCEQF	Mode Command HGA Positioner ON/OFF No Change ON OFF NADIR VPD Reset (1 = Reset) NADIR VPD Reset (1 = Reset) Phase Reset Enable/Disable No Change Enable Disable	1 0 1 X X X X X X X LSB MSB Magnitude 0 0 0 1 1 0 0 0 0 1 1 0

to the normal mode. The azimuth offset value must take into account all the biases between the HGA boresight and the selected reference sensor to properly position the despun platform.

The biases associated with the use of the sun sensor are presented in Table 3.3.2.4-2 and with the star sensor in Table 3.3.2.4-3. The geometry associated with these biases is shown in Figure 3.3.2.4-3. To compensate for the bias value, it must be added to the desired offset angle which is referenced to the selected inertial reference, sun or star. In-flight calibration should be used to further refine the offset bias value. An example of the offset command bit structure using the sun sensor as the SRR reference is as follows:

Desired pointing direction: 32.2 degrees from sun line, measured in direction of spacecraft spin

Required offset command is: 12.55° (from Table 3.3.2.4-2) +32.2° ≈ 44.8°

Binary equivalent of angle (44.8 ÷ 0.0879 ≈ 510): 000 111 111 101

Magnitude in 2's complement notation: 111 000 000 011

Command format with address (LSB first): 001 110 000 000 111

SRR to MIP telemetry will translate in angular degrees to read ≈ +1.41° more than the offset command value (i.e., 44.8° +1.41° = 46.21°).

Figure 3.3.2.4-4 defines the spacecraft polarity conventions and relationship with the DCE control.

TABLE 3.3.2.4-2

HGA BORESIGHT/SUN SENSOR BORESIGHT BIAS

(1)	Sun Sensor Leading Edge Detection Level (150 μamp Level) to Leading Edge of ADP Detected Pulse (SRR Pulse When Sun is Selected)	-0.02°	(Note 1)
(2)	Sun Sensor Boresight to Sun Sensor Leading Edge Detection Level (150 μamp level)	$+0.75^{\circ}$	(Note 2)
(3)	+X Axis to Sun Sensor Boresight	-7.44°	
(4)	SRR to F_S Bias (PLL) (Maximum)	$\pm 0.25^{\circ}$	(Note 3)
(5)	Antenna Boresight to +X-Axis	-0.33°	
(6)	MIP to +X-Axis	$+21.1^{\circ}$	
(7)	MIP to 128th SAE Pulse	-1.41°	(Note 4)
(8)	128th SAE Mechanical to Electrical Location	-0.12°	
		$+12.53^{\circ} \pm 0.25^{\circ}$	

NOTE 1: A positive time delay reflects itself as a negative angle (Positive angle is in the positive spin direction).

Varies essentially with spin rate. A value may be estimated via use of Figure 3.3.2.1-4.

NOTE 2: Varies with solar intensity. The absolute value may be estimated via use of Figure 3.3.2.1-3 and interpolating geometrically as necessary. The change in this value due to solar intensity may be measured by comparing ATTM measurements to the same star, using the sun as the SRR for all measurements. The absolute value may be determined by measurements to two or more stars, and determining star sensor biases concurrently via iterative estimation.

NOTE 3: This is measurable by commanding the SRR to RIP delay to zero degrees and reading "SRR to RIP" ATTM. (See Figure 3.3.2.1-3 and Paragraph 3.3.3.1).

NOTE 4: This is measurable by commanding the HGA Azimuth position to zero degrees, reading "SRR to MIP" ATTM, and subtracting the value found via Note 3.

TABLE 3.3.2.4-3

HGA BORESIGHT/STAR SENSOR BORESIGHT BIAS

(A)	Delay between PSI* optical axis (Star Sensor Trailing edge detection level, i.e., 25% of peak) and PSI* position pulse (SRR pulse when PSI* is selected)	-0.25° (Note 1)
(B)	Star Sensor Boresight to PSI* Optical Axis	$+6.21^{\circ}$ (Note 2)
(C)	+X-Axis to Star Sensor Boresight	$+15.018^{\circ}$
(D)	SRR to F_s Bias (PLL) (Maximum)	$\pm 0.25^{\circ}$ (Note 3)
(E)	Antenna Boresight to +X-Axis	-0.33°
(F)	MIP to +X-Axis	$+21.1^{\circ}$
(G)	MIP to 128th SAE Pulse	-1.41° (Note 4)
(H)	128th SAE Mechanical to Electrical Location	-0.12°
		$+40.22^{\circ} \pm 0.25^{\circ}$

NOTE 1: A positive time delay reflects itself as a negative angle (Positive angle is in the positive spin direction).

The value shown here is equivalent to a nominal 2 millisecond delay at 15 rpm. Actual delay is dependent on the star brightness, star elevation, star channel, bandpass selection, and spin rate. A value may be determined iteratively with determination of star elevation, via use of Figure 3.3.2.2-3A and -3B if the PSI* channel is being used for the SRR, or via use of Figure 3.3.2.2-3C and -3D if the PSI2* channel is being used for the SRR.

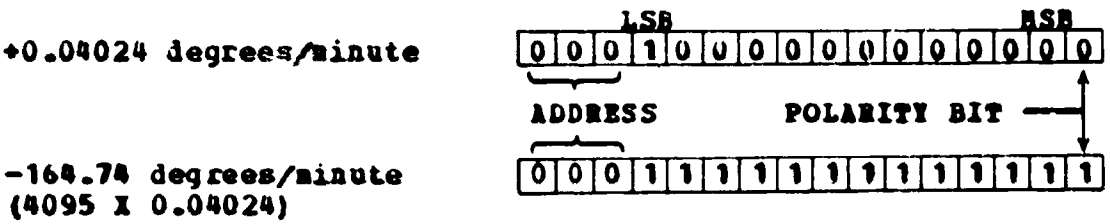
NOTE 2: If PSI2* slit is selected for star reference, this value would be deleted from the list.

NOTE 3: This is measurable by commanding the SRR to RIP delay to zero degrees and reading the "SRR to RIP" ATTM.

NOTE 4: This is measurable by commanding the HGA Azimuth position to zero degrees, reading the "SRR to MIP" ATTM, and subtracting the value found via Note 3.

Azimuth slew rates of up to ± 164.74 degrees per minute in increments of 0.04024 degree per minute can also be commanded. The slew input modifies the stored position information causing the desired slew rate.

Examples of the command bit structure are as follows:



A summary of the internal response for the despin control is presented in Table 3.3.2.4-4.

3.3.2.4.5 HGA Elevation Control. Elevation control of the HGA is provided by open loop control of rate pulses to the stepper motor of the HGA positioner. The HGA elevation positioner slew command (DCEQ4 or DCEQD) provides rate and polarity information which is translated into a four-phase pulse sequence, the order of which establishes the stepping direction. The capability is as follows:

ELEVATION POINTING	NOMINAL VALUES
Step Quantization:	0.0213 degree
Slew Rate Range:	0 to ±39.94 deg/min
Rate Quantization:	0.00975 deg/min

The HGA positioner will move in a direction to point the antenna down toward the rotor (+Y rotation) when the most significant bit (Bit 40) in the command word is set to logic one. The other eleven magnitude bits in the command word produce a rate proportional to their count, where the least significant bit produces the lowest rate, 0.007625 pulses per second causing the antenna dish to move at a nominal rate of 0.00975 degree per minute.

TABLE 3.3.2.4-4

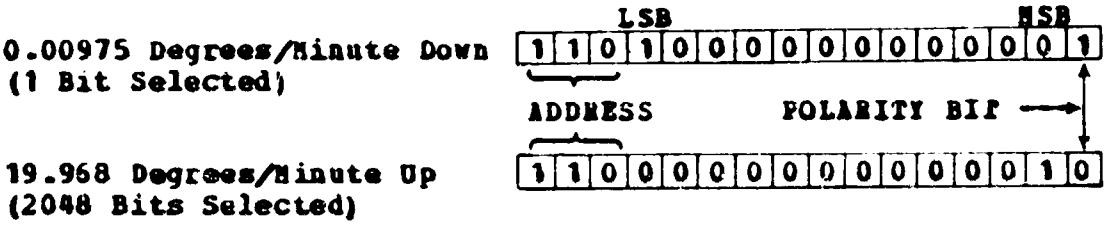
INTERNAL RESPONSE AND POLARITY SUMMARY

Command or Condition	Resulting Incremental Torque Generated on Antenna Platform	Resulting Direction Of Platform Rotation About S/C Spin Axis (+Z)	Telemetry Indicators	Reference Designation
			TM Title and Remarks	
Azimuth Pointing Slew Command (DCEQ1 or DCEQA) with Sign Bit = 1 (Bit 40 = 1)	Despin Torque	-Z (Clockwise)	SRR to MIP gets smaller.	AM1ADS or AM2ADS
BAPTA Torque Bias	Despin Torque	-Z	SRR to MIP gets smaller.	AM1ADS or AM2ADS
Azimuth position command (DCEQ5 or DCEQE) to a larger offset value.	Spin-up Torque	+Z (Counterclockwise)	SRR to MIP gets larger. Position torque momentarily decreases.	AM1ADS or AM2ADS APOSTQ
Despin Motor Drive Current increasing from previous nominal.	Despin Torque	-Z	Despin motor current gets larger.	AMOTRI
Rate Torque increases plus.	Despin Torque	-Z	DCE Rate Torque increases above nominal 2.75 V	ARATEQ
Position Torque Increases plus.	Spin-up Torque	+Z	DCE position Torque increases above nominal 2.5 V.	APOSTQ

The maximum output voltage for the elevation position TM (AELEVD) occurs when the actuator is fully extended causing the antenna to be pointed away from the rotor (-Y rotation).

The antenna will move in elevation indefinitely in response to any slew command until the antenna reaches the mechanical stops, or the slew is terminated by a zero elevation slew command or by commanding the positioner electronics OFF (DCEQ6 or DCEQF). The elevation position TM of the dish location (AELEVD) will read a maximum value of -21.47 degrees beam position (antenna pointed toward spacecraft +Z) and +17.26 degrees beam position (antenna pointed toward spacecraft -Z) at the two mechanical stops. Actual mechanical travel range is ±10.1° between mechanical stops.

Examples of the command bit structure of the Elevation Slew Command is as follows:



3.3.2.4.6 Scientific Reference Signals. The two variable phase delay generators provide a pulse once every rotor spin period to selected scientific instruments. The rate of change of the pulse location is controlled by command and has the following characteristics:

Angle Quantization:	0.088 degree
Rate Quantization:	0.04023 deg/min
Rate Range:	0 to 164.74 deg/min

Individual reset commands (part of DCEQ6 or DCEQF) control the initialization of each pulse. Upon being reset, the VPD output will occur ≈ 360 degrees [actual is 360 -1/2 (P_s/4096) or 359.96 degrees] with reference to the previous RIP.

This is equivalent to setting the phase delay from the previous RIP to [360° minus the VPD pulse width (width is ≈0.004° spin angle)]. The commanded rate will then cause the pulse location to steadily increase or decrease. The delay between the RIP and VPD output will decrease whenever the variable phase delay MSB (Bit 40) is set to logic one and will increase when set to logic zero. The zero state of the polarity bit moves the VDP pulses in the direction of spin.

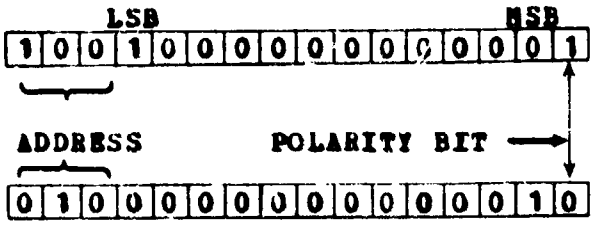
The phase delay will continue to increase or decrease indefinitely at the commanded rate unless changed or set to a zero rate by a subsequent rate command. The computed angle delay of each output is telemetered (AVPDRD and AVPDWD). The electronics processing of the VPD signals is shown in Figure 3.3.2.4-2.

The phase delay profile should be established such that either pulse will not transition through 0 or 360 degrees once initiated. This will prevent double pulses or loss of an output pulse during the spin revolution where the transition occurs. A commanded negative rate will cause a missed pulse each time 0° is transitioned. A commanded positive rate will cause a double pulse each time 360° is transitioned. For example if a positive rate is initiated from the reset condition, a VPD output will occur 359.96 degrees with respect to one RIP and a second pulse will occur during the same spin revolution which is referenced to the second RIP. Unless a phase reset is again commanded, a phase error will result.

Examples of the command bit structure is as follows:

RAM
VPD: 0.04024 Degrees/Minute
Decreasing Rate
(1 Bit Selected)

NADIR
VPD: 2.391 Degrees/Minute
Increasing Rate
(2048 Bits Selected)



3.3.2.4.7 Telemetry Interface. A 32-bit serial digital output is provided to telemetry as a four 8-bit word containing the following information:

TM DESCRIPTION	MNEMONIC	NUMBER OF BITS
• RAM Variable Phase Delay	AVPDRD	12
• NADIR Variable Phase Delay	AVPDND	12
• DCE Mode Status		
• Phase Reset Enable/Disable	ADSPNS	1
• Spare bits		7

The 12 bit variable phase delay outputs use two's complement notation. The straight binary count should be subtracted from 360 degrees for the correct angle representation. For example, an all "1" output represents a 1 bit (0.088 degree) delay from RIP. An output pulse is generated on overflow or when one 4096 F_s pulse is counted.

The data is shifted out most significant bit first. The data is updated during the last half of every major frame and then held at the start of the major frame for the once per major frame read rate.

This timing relationship is illustrated in Figure 3.3.2.4-5. The VPD generator is updated every 4096 F_s time interval. The computed angle is transferred to a counter at the RIP event, and is clocked out to the on-board instruments at overflow. The computed angle is transferred to the TM register at the start of the major frame,

and is telemetered during the first half of the major frame. Although the actual telemetered angle may differ from the on-board angle, the computed rate profile will be the same as the on-board profile in time. Knowledge of RIP location in time will then relate the actual on board VPD angle location.

The serial output from each DCE is wired OR'd and provided to a common DIM channel, thus there is no TM redundancy for the two units.

The following analog channels are sent directly to a DIM which provides 8-bit quantization for each.

TM DESCRIPTION	MNEMONIC	OPERATING RANGE MAX.	OPERATING RANGE MIN.
a. Position Torque*	APOSTQ		
b. Rate Torque*	ARATEQ		
c. Despin Motor Drive Current*	AMOTRI		
d. DCE #1 Power Supply Voltage	ADCE1V		
e. DCE #2 Power Supply Voltage	ADCE2V		
f. DCE #1 Temperature	ADCE1T	20°F	120°F
g. DCE #2 Temperature	ADCE2T	20°F	120°F

*NOTE: The output from each DCE is wire OR'd and provided to a common DIM channel; thus there is no TM redundancy for the two units.

The following bi-level outputs are provided by the designated unit to separate DIM channels.

TM DESCRIPTION	MNEMONIC
Despin Motor Drive #1 ON/OFF	AMTR1S
Despin Motor Drive #2 ON/OFF	AMTR2S
HGA Positioner Electronics #1 ON/OFF	APOS1S
HGA Positioner Electronics #2 ON/OFF	APOS2S

3.3.2.5 Bearing and Power Transfer Assembly (BAPTA). The BAPTA is located on the spin axis of the Orbiter spacecraft and provides all the mechanical, electrical and RF interfaces between the spun and despun sections. It supports, despins and controls azimuth pointing of the HGA with signals from the DCE. The BAPTA contains an RF rotary joint which permits the transfer of X-band and S-band frequencies between the spun and despun sides.

There is no launch lock to secure the BAPTA despun section for the launch phase, so the despun section is free to rotate at all times. Also, radial loads cannot cause rotating and non-rotating parts from contacting one another through the use of internal mechanical stops. Bearing lubrication is provided by two foam ball retainers impregnated with oil. Sufficient oil is provided for the bearings which is 17 times more than required for the nominal mission.

Two temperature sensors are located to sense temperature in the vicinity of the bearings. The inner race temperature of the forward bearing is designated by ABAPIT. The outer race temperature of the aft bearing is designated by ABAPOT. The expected bearing temperature during the mission will range from 47° to 75°F. The operating range is 30° to 90°F for each of the two telemetered temperatures.

The unit contains two brushless DC torque motors sharing common laminations and a permanent magnet assembly and two resolvers. A motor and a resolver are married to a DCE (no cross-strapping).

The DCE supplies a sinusoidal voltage to the primary (modulator) coil of the resolver. The amplitude of this signal is proportional to the despin error signal. The relative angular relationship of the spinning section to the despun section provides the internal phasing relationship within the BAPTA. The resolver sine and cosine outputs then provide the phase relationship and a proportional amplitude error

signal to the DCE. These signals are demodulated in the DCE, power amplified and feedback in the proper phase to the BAPTA motor sine and cosine windings.

The output torque capability of one motor winding set is illustrated in Figure 3.3.2.5-1 for the minimum switched loads bus voltage condition (24 V available to the motor). Also shown is the motor capability with a nominal 27.8 volts supplied to the motor. In addition, the figure illustrates the torque margins of the motor for the conditions of minimum operating voltage at both minimum and maximum spin speeds. The margin of available motor torque above BAPTA friction reduces as spin speed increases. For example, at a spin speed of 35 rpm and at 40°F operation, the available torque is 0.71 ft-lb whereas 0.20 ft-lb maximum is required to operate the BAPTA.

The BAPTA provides the two DCEs relative position information for despin control by detecting coincidence of reference marks between the spun and despun sections. This once per revolution pulse is the master index pulse (MIP). In a similar fashion, the relative rate of the two sections is provided to each DCE by detecting 128 (SAE) pulses per revolution. The DCE compares this pulse train with the 2^{12} F_S pulse train from the ADE to derive the rate information. These pulses are generated by two completely passive magnetic pickups as they pass over soft iron teeth accurately positioned on the despun BAPTA flange. Each of the two magnetic pickups are bifilar wound. This provides separate outputs to each DCE while maintaining the same phase relationship to each unit.

The signal interface across the BAPTA is shown in Figure 3.3.2.4-1 for the HGA positioner and DCE interface, and in Figure 3.7.1-4 for the communication subsystem.

3.3.2.6 Nutation Damper. The nutation damper on each spacecraft is a passive, sealed straight tube of liquid freon B3.

Wave action in the peripheral surface of the liquid absorbes the nutational energy of the spacecraft, assuring positive stability of the vehicle. The damper is required to reduce vehicle nutation to 1/3 (≈ 37 percent) of the initial value in a time interval of ≤60 minutes for all phases of the mission.

A fill fraction of 25 percent of the total cavity volume is employed, based upon results of optimization studies.

The damper longitudinal axis is mounted 46.5 ±1.0 inches radially outboard from and parallel to the vehicle spin axis. On the Orbiter spacecraft, an azimuth location constraint is added to obtain effective damping while the magnetometer boom is stowed. To meet this constraint, the damper is physically located at an angle of 143.6 degrees, as measured from the +X axis.

Liquid Freon contained within the damper is free to move at all times. Its boiling point of +306°F and freeze point of -160°F are well outside the qualification temperature range of 138°F and 0°F. Because of favorable inertia ratios during pre- and post-apogee motor firing, it will not produce unfavorable dedamping or amplification of nutational motion. After spinup, centripetal force acts on the liquid Freon forcing it to spread out along the tube longitudinal axis. During spacecraft nutational motions, free surface waves are propagated in the liquid Freon. Energy of nutation is dissipated by means of viscosity friction effects of these surface waves relative to the damper tube wall.

The performance of the damper is dependent on spin rate, temperature and the mass properties of the vehicle. Parametric performance curves of damper performance for nutation angles less than 1 degree are presented in Figure 3.3.2.6-1. The nutation angle will be in excess of 1 degree after boom deployment. The performance of the damper at this phase and other specific phases of the mission is summarized in Table 3.3.2.6-1.

TABLE 3.3.2.6-1

PREDICTED DAMPER PERFORMANCE SUMMARY FOR ORBITER
(ONE DAMPER PER VEHICLE)

Mission Phase	Temp (°F)	Inertia Ratio	Spin Speed (RPM)	τ (Min.)	
Separated; boom in	21	1.18	6.5	17.7	
Separated; boom in	77	1.18	6.5	32.8	
Separated; boom in	140	1.18	6.5	55.0	
Boom out	21	1.26	15	35*1	3.6*2
Boom out	77	1.26	15	35*1	6.1*2
Boom out	140	1.26	15	35*1	9.3*2
Orbit Insertion	21	1.26	30	2.9	
Orbit Insertion	77	1.26	30	4.9	
Orbit Insertion	140	1.26	30	7.5	
Venus Orbit	21	1.38	5	6.4	
Venus Orbit	77	1.38	5	3.6	
Venus Orbit	140	1.38	5	2.4	

*1 Based on wide angle nutation tests, having an initial nutation angle of 12°.
*2 Steady state (near zero degrees nutation) values.
*3 τ is time constant (to produce 1/e of starting value).

3.3.2.7 Magnetometer Deployment Mechanism. The Magnetometer Deployment Mechanism (MDM) is a three segmented deployable boom on which three magnetometer sensors are mounted. After launch, the boom will be deployed, positioning the magnetometers away from the Orbiter's magnetic field so that it will not interfere with the sensor's magnetic measurements. The MDM is mounted on the Orbiter shelf's thermal barrier such that in the deployed configuration the base of the outboard magnetometer sensor is 19.5 feet from the spacecraft Z axis and 15.7 feet from the inboard root joint.

The boom is deployed via real time ground command after the spin rate has been set to 6.5 ± 0.3 rpm. The boom is held in the stowed position on a fixed base by two hinged beams which are released by pyrotechnic actuated pin pullers. Each hinge beam is held by two pin pullers providing redundancy. Either one or both pin pullers on each beam will permit deployment of the boom. Each pin puller contains a standard single bridgewire electric squib. The interface with the squib driver is shown in Figure 3.3.2.7-1. Both squib drivers can be armed and fired by either COM 1 or COM 2. A single fire command will cause all four squibs to fire simultaneously. The redundant set of commands will fire the same set of squibs through different command line inputs to the PCU.

Upon command, the pin pullers release the caged boom assembly, and centrifugal force developed by the Orbiter's spin rate drives the hinged boom segments outward. In addition, a limited travel deployment spring on the outboard stowage support provides an initial impulse to overcome static friction. Centrifugal force causes the three segments to deploy, and detent pins at each hinge automatically lock the deployed assembly.

The deployment sequence depends on the magnitude of the spin speed, hinge friction, detent pin friction, and cable bending torques. Computer simulation indicates the most likely sequence is as follows: The center hinge locks first, then

the outboard hinge, and finally the inboard hinge locks, completing the deployment.

A switch positioned on the inboard hinge support triggers a deployment indication signal when the boom has deployed and the inboard hinge has locked. The signal is provided to DIM 2 (SBOOMS) and becomes a logic "1" upon locking.

In the extended and locked configuration, all three magnetometer sensors are positioned and maintained within 1 degree of their spacecraft reference axis, through the transit phase, and after orbit insertion.

3.3.2.8 High Gain Antenna (HGA) Elevation Positioner. The HGA positioner provides the capability for in-flight elevation orientation of the high gain antenna parabolic antenna. The elevation control provided by the HGA positioner and azimuth control provided by the DCE permit the antenna to be pointed to Earth for communication and the occultation experiment. The elevation control capability permits the HGA beam to be moved -21.47° up from horizontal, toward the +Z axis, and down +17.26°; corresponding to a total 20.2 degrees mechanical movement of the dish.

Movement of the antenna is achieved by the rotation of a jackscrew by a stepper motor. The stepper motor receives pulses at a controlled rate from the DCE. Each pulse causes the motor to step 90 degrees, which in turn results in a 15 degrees rotation of the jackscrew shaft. The 15 degrees rotation of the jackscrew results in an antenna motion of 0.0213 degree per motor step nominal. The variation in step size with gimbal position is shown in Figure 3.3.2.8-1.

The DCE converts a quantized rate command (DCEQ4 or DCEQD of "DCE CONFIGURE" quantitative command) into a sequential pulse output to the stepper motor. The commandable frequency range is zero to 31.25 pulses per second in 0.007625 pulses per second increments. With a step size of 0.0213 degree per pulse, this produces a nominal antenna movement control of up to 39.9 degrees per minute

with rate increments of 0.009744 degree per minute. The DCE controls the sequential order of the pulses providing polarity control. The pulse widths are fixed duration for a given slew rate and are at 100 percent duty cycle, i.e., no time separation between pulses and thus no current transients should be observed between pulses (Pulse power is 11 volts at 28 v.) The electrical interface with the DCE is shown in Figure 3.3.2.4-1.

The elevation position of the HGA is telemetered using a linear potentiometer. The potentiometer is provided with +4.85 volts dc across the slip rings by the DCE that is on (0 v to + 4.85 v range for the potentiometer wiper arm pickoff). As the antenna moves through its travel range, it moves the wiper shaft of the potentiometer, providing a proportional voltage which relates the antenna or beam position. The potentiometer output and its return is provided across the slip rings as a differential signal to DIM 3 and DIM 5 (AELEVD).

The temperature of the stepper motor is telemetered (APOSMT). The operating range is -60° to +180°F, although the turn-ON (non-operating) range extends down to -200°F and up to +130°F.

3.3.2.9 Engineering Instrumentation

3.3.2.9.1 Temperature Monitoring. Temperature sensors for the control subsystem provide thermal information directly to DIMs as shown in the table below. The measurements are only sampled in the subcom format and there is no redundancy.

DESCRIPTION	LOCATION	MNEMONIC
DCE #1 Temperature	Near BAPTA Motor Driver #1	ADCE1T
DCE #2 Temperature	Near BAPTA Motor Driver #2	ADCE2T
HGA Position Motor Temperature	On the backside of the stepper motor	APOSMT
Star Sensor Temperature	Near the detector chip	ASTART
BAPTA Inner Race Temperature	Near the forward BAPTA bearing	ABAPIT
BAPTA Outer Race Temperature	Near the aft BAPTA bearing	ABAPOT

3.3.2.9.2 Unique Telemetry Measurements. The deployment status of the magnetometer boom is provided directly to DIH 2 as a bi-level status bit (SBOOMS). There is no redundancy. A microswitch is located at the inboard hinge support and is actuated by a locking detent pin when the boom assembly rotates to the fully deployed position. In the "not deployed" state, the switch is shorted to signal ground providing a logical zero state. In the "deployed state", the switch is open circuit providing a logical one state.

3.3.3 Operational Description

3.3.3.1 Sun Acquisition. Sun acquisition occurs when the PLL in the ADP obtains a frequency and phase such that the F_s output pulse is in lock with the detected sun pulse. Since the sun can be detected unambiguously, acquisition can be accomplished using the ungated (360 degree gate) sun mode. However, certain command information must be supplied to the ADP to achieve and verify a successful acquisition. After the ADP "ON" command is generated (ADP19 or ADPA9 for ADP No. 1; ADP29 or ADPB9 for ADP No. 2), then the ADP must be configured via the quantitative commands in the following manner (Measurements A and B correspond to measurements 1 and 2 respectively in the Mnemonic listing contained in Reference: Paragraph 1.5.1):

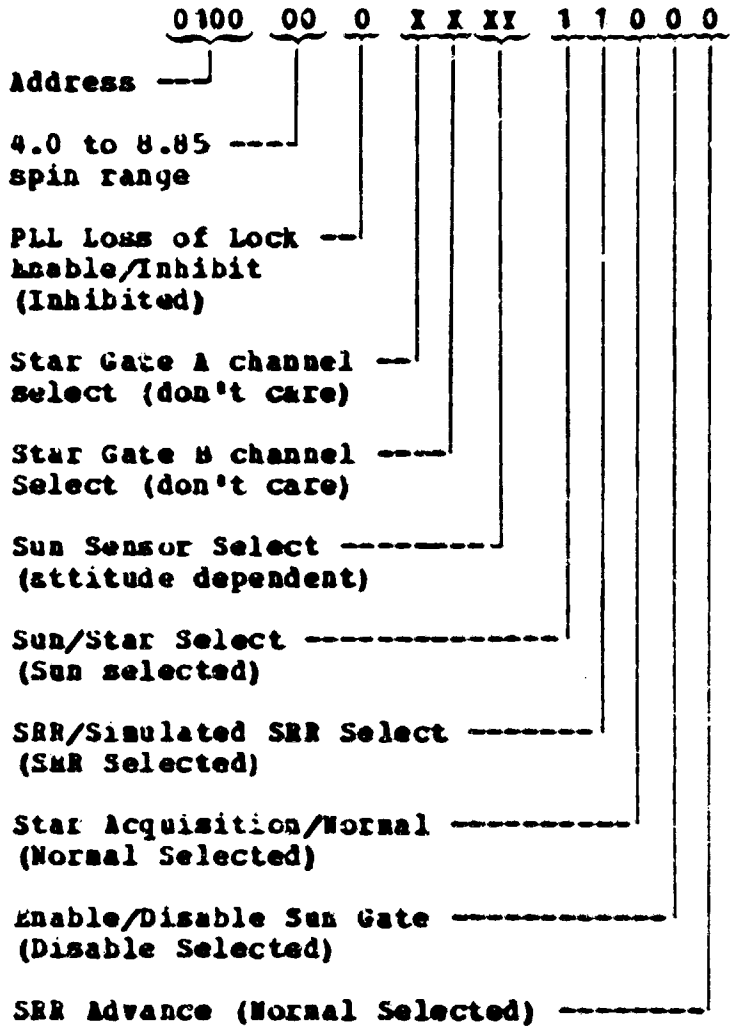
(a) Measurement Select (ATQØ1 or ATQØA)

COMMAND bIP #:	25 thru 28	29 thru 34	35 thru 37	38 thru 40
	0000	XXXXXX	010	010
	Address	Don't care	Measurement A; SRR to RIP	Measurement D; SRR to RIP

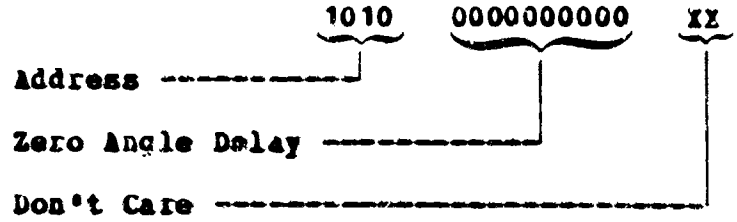
(b) Jet Control (ATQØ2 or ATQØB)

1000	0000000	1	0	0	0	1
Address	No jets selected	Normal fire mode selected	128 ms pulse width selected	Time count selected	Pulse mode selected	Spin rate detector enabled (assumes spin rate >3.7 rpm)

(c) ADP Mode Select (ATQ03 or ATQ0C)



(d) Roll Index Delay Magnitude (ATQ06 or ATQ0F)



The "SRR to RIP" measurement is selected to verify normal Phase Lock Loop Acquisition. At ADP power turn-on, the PLL is initialized to the center of one of the randomly selected spin ranges; i.e., 6.425 rpm, 12.85 rpm, 25.7 rpm or 51.4 rpm. Proper initialization by the ADP Mode Select Command above is necessary so that the PLL can slew in the correct frequency range using the sun as a reference until the PLL rate is identical with the spacecraft rotor rate. The only time the PLL will not slew in frequency is if the PLL is in phase and frequency lock with the SRR. Therefore, by selecting the RIP angle to be zero, the attitude measurement SRR to RIP will provide a measurement of the PLL phase bias from ADP input leading edge detection (SRR pulse) to P_S (PLL overflow pulse). This is identified as SRR to P_S bias in Figure 3.3.2.1-3. (Any roll index delay can be used if this phase bias is not to be measured.) During sun acquisition, the measurement of SRR to P_S will behave like a damped sine wave (see Figure 3.3.3.1-1). Once the PLL is in lock, the measurement provides the steady state phase bias between the SRR and the PLL output for attitude determination update. The steady state bias is not expected to exceed ±0.25°.

A time history of representative phase lock loop acquisition for 5 rpm and 15 rpm (Figures 3.3.3.1-1 and 3.3.3.1-2) shows total acquisition time of 450 seconds and 76 seconds, respectively. These figures were derived from a computer simulation that excluded any effect of nutation and wobble.

The following time interval measurements are obtained upstream from the phase lock loop, and their validity is therefore unaffected by the state of the phase lock loop: (ψ-ψ, ψ-ψ_2, ψ-Major Frame Rate, and ψ-Minor Frame Rate).

Proper telemetry indication after Sun Acquisition is as listed in Table 3.3.3.2-1.

3.3.3.2 ADP Normal Operating Mode. The normal operating mode will be considered to be sun reference while

TABLE 3.3.3.2-1

TELEMETRY INDICATIONS FOR ADP STATUS

Mnemonic for Pertinent TM	Status After Sun Acquisition	Status After Commanded to Normal Operating Mode
ATTM1Z and ATTM2Z	• SRR to SRR equal to expected spin period • SRR to RIP = 0.0° ± 0.25° (0° delay cmd) • SRR to PSI2 equal to (62° ±10°) to (80° ±10°) in the absence of nutation damping.	• SRR to SRR equal to expected spin period • SRR to RIP equal to commanded delay ±0.25° • SRR to PSI 2 equal to 90.0° ±2°. • SRR to MIP (corresponds to commanded azimuth position command after bias compensation)
AJMAGC	• JCE countdown at maximum count 4095.	• JCE countdown at a count of zero.
APULLS	• Pulse width selection = 128 ms	• Pulse width selection = 128 ms.
AMAGCS	• Pulse time count selection = time count	• Pulse time count selection = time count.
AJETMS	• Pulse fire mode selected	• Pulse fire mode selected.
ASDETS	• Spin rate detector enabled.	• Spin rate detector enabled.
AJETMS	• Normal fire mode sel.	• Normal fire mode selected.
AJETSS	• No jets selected.	• No jets selected.
AJCEFS	• JCE fire status disable	• JCE fire status disabled.
A*GASS	• Star Gate A channel select as commanded.	• Star Gate A channel select as commanded.
A*GBSS	• Star Gate B channel select as commanded.	• Star Gate B channel select as commanded.
ASUNSS	• Sun sensor selection as commanded	• Sun sensor selection as commanded

TABLE 3.3.3.2-1 (Continued)

Mnemonic for Pertinent TM	Status After Sun Acquisition	Status After Commanded to Normal Operating Mode
ASRRMS	• Sun selected as SRR	• Sun selected as SRR
A*ACQS	• Star normal mode selected.	• Star normal mode selected.
ASPINS	• Spin range selection = 4.0 to 8.85 rpm.	• Spin range selection as commanded
ALOLES	• PLL Loss of Lock inhibited.	• PLL Loss of Lock enabled.
ASUNGS	• Sun gate disabled.	• Sun gate enabled.
ADVANS	• SRR advance set to normal.	• SRR advance set to normal.
ARIPAD	• RIP delay equal to zero degrees.	• RIP delay as commanded.
ALOCKS	• In lock status	• In lock status.
AMISSC	• Normally = 0 counts	• Normally = 0 counts.
AACSAD	• ACS angle delay is a random selection	• ACS angle delay as commanded.
ASIMSZ	• PLL spin period as commanded	• PLL spin period as commanded.

Status after Sun Acquisition applies for magnetometer boom stowed.
Status for Normal Operating Mode applies for 15 rpm and cruise configuration, or 5 rpm and on-orbit configuration.

spinning at 15 rpm for the transit phase or 5 rpm for the on-orbit phase. The two different spin speeds only affect the selected spin range for the PLL (SMXX of ATQ∅3 or ATQ∅C) and the magnitude of the commanded simulated spin period (ATQ∅7 and ATQ∅8) or (ATQ∅G and ATQ∅H). The sun acquisition operation discussed in the preceding paragraph is considered to have been accomplished at this point.

The initial conditions for transferring to the normal operating mode are assumed to be as follows:

(a) Spacecraft spinning at initial spin-up value (5 to 6.5 rpm) or at 15 rpm.
(b) ADP is ON and the PLL is in lock with the sun as reference.

The jet control logic should have been commanded to the safest configuration; i.e., no jets selected with the spin rate detector enabled during sun acquisition (see 3.3.3.1).

Telemetry indications for these initial conditions should indicate as shown in Table 3.3.3.2-1.

The jet countdown magnitude (ATQ∅4 or ATQ∅D) should be commanded to a zero count. (The initial spin-up maneuver would have terminated with a maximum count in the count register.) Verify the JCE buffer output power (AJCE1S or AJCE2S) is off, or disable the output by command (ATQ1∅ or ATQ∅J).

The ATTM measurements (ATQ∅1 or ATQ∅A) should be command selected to be:

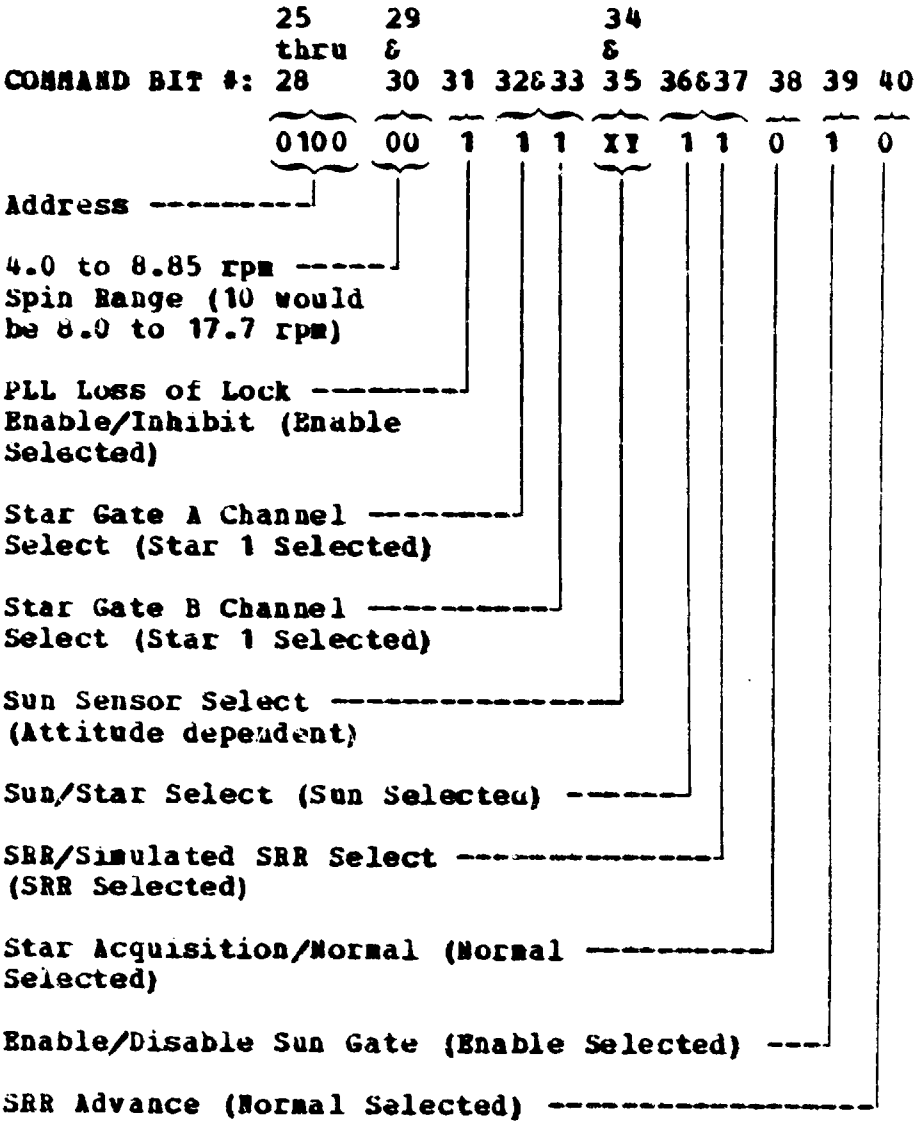
(a) SRR to SMR
(b) SRR to PSI 2.

This will provide a measurement of the actual spin period and the sun aspect angle. These two measurements provide a quick look of the spacecraft inertial relationship. After the average spin period has been determined, the

simulated spin period of the same magnitude should be commanded ((ATQ#7 and ATQ#8) or (ATQ#G and (ATQ#H)). It is recommended that the spin period be continuously telemetered throughout the mission if other attitude measurements are not being made. The simulated spin period then should be upuated to the nearest integral value (25 millisec, resolution) when command opportunities exist. A one-bit error will permit operation in the free-run mode for 16-minutes at 15 rpm without the inertial reference moving outside of Gate A.

After the correct simulated spin period has been commanded, the ADP should be commanded to operate in the gated sun mode with the PLL loss of lock enabled. This will permit transfer to the simulated roll reference if two consecutive sun pulses are missed for any reason. The ADP mode command to select this configuration is as follows:

AT063 or AT06C:



3.3.3.3 Star Sensor Control Including Star Acquisition

3.3.3.3.1 Channel Control. Each channel of the star sensor is individually controlled by ON and OFF discrete commands. Both channels will normally be commanded ON for attitude determination measurements. During the eclipse period, one of the two channels will be selected for the roll

reference (SRR) in the ADP. At other times in the mission, one or both channels may be commanded off depending on power availability.

Typical telemetry response for the star sensor with both channels operating is shown in Table 3.3.3.3-1.

3.3.3.3.2 Star Detection Capability. The star sensor has capability of detecting (one at a time) any one of 25 stars or the planet Saturn within the celestial sphere, dependent upon the spacecraft attitude. The detection capability at any point in time will vary with spacecraft spin rate, star elevation within the FOV, sensor temperature, and the selected bandwidth within the sensor. These effects are presented in 3.3.2.2 and are to be used to correct the telemetered star brightness value. The sequences for star brightness calibration are listed in Table 3.3.3.3-2. Temperature affects the probability of detection for a given star, but does not affect the brightness to TM volts relationship significantly.

The brightest stars are listed in Table 3.3.3.3-3 in decreasing order of brightness index ("silicon magnitude" as determined by the star sensor silicon detector spectral response and tabulated in Reference: Paragraph 1.5.21) with their positions expressed in terms of Earth Centered Ecliptic coordinates. In-flight calibration shall be used to correct the list to remove intensity errors created by a non-perfect star standard during test.

The minimum star brightness levels (as telemetered) required for time interval measurements versus selected parameters, are listed in Table 3.3.3.3-5.

TABLE 3.3.3.3-1

TELEMETRY INDICATIONS FOR STAR SENSOR RESPONSE

MNEMONIC FOR PERTINENT TM	TM CHANNEL	TYPICAL STATUS OR EXPECTED VALUE
ASTART	Star Sensor Temperature	Dependent on spacecraft attitude and solar intensity; TBD °F at sun aspect of 90±5°
A*1BRM	Star 1 Brightness	Dependent on location of star gate B and star magnitude within the gate.
A*1BPS	Star 1 Bandpass State	Consistent with command selected value (S1BWX selection of STRQ1 or STRQA).
A*1THS	Star 1 Threshold Setting	See Section 3.3.4 for expected value associated with commanded threshold setting.
A*2BPS	Star 2 Bandpass State	Consistent with command selected value (S2BWX selection of STRQ1 or STRQA).
A*2BRM	Star 2 Brightness	Dependent on location of star gate B and the star magnitude within the gate.
A*2THS	Star 2 Threshold Setting	See Section 3.3.4 for expected value associated with commanded threshold setting.
A*1ONS	Star 1 ON/OFF	Logic Level "1" if ON.
A*2ONS	Star 2 ON/OFF	Logic Level "1" if ON.

TABLE 3.3.3.3-2

STAR BRIGHTNESS CALIBRATION METHOD

CASE A: Starting with telemetered data, determine silicon magnitude of the star:

(A.1) Calculate the mean and 3σ deviation from the mean of 30 successive TM samples of A*1BRM or A*2BRM (corresponding to that star channel that was assigned to Star Gate B at the time of data collection). Reject all samples outside the 3σ deviation, and calculate the mean of the remaining samples ($\equiv \mu_1$).

(A.2) For worst case conditions (to account for end-of-life, possible errors in system calibration, and Johnson Reference), increase μ_1 by 14%:

$$\mu_2 = (1.14)\ \mu_1 .$$

(A.3) To account for star sensor FOV effect on the sensor input signal level, calculate

$\mu_3 = (1 - C)\ \mu_2$, where C = the magnitude change obtained from Figure 3.3.2.2-8.

(A.4) Calculate the equivalent silicon magnitude for the star:

$$M_{si} = \text{silicon magnitude} = -2.512 \log \left(\frac{\mu_3 - A}{B\, I_o}\right)$$

where:

A = Offset
B = Gain
I_o = DC video for "0" magnitude star
μ_3 = AC video for the subject star.

Parameters A and B are obtained from Table 3.3.3.3-4, corresponding to the prevailing spin rate and selected star channel bandpass.

Parameter I_o = 286.9 mv for the PSI* channel; I_o = 294.2 mv for the PSI2* channel.

CASE B: Starting with the silicon magnitude of a star, determine the expected telemetered value for star brightness (A*1BRM or A*2BRM):

(B.1) Calculate the uncorrected star sensor signal input level:

$$\mu_3 = \text{uncorrected signal} = A + BI_o\,(2.512)^{-M_{si}}$$

where the terms are as defined in Step (A.4) of this table.

(B.2) To account for star sensor FOV effect on the sensor input signal level, calculate:

$\mu_2 = (1 + C)\ \mu_3$, where C = the magnitude change obtained from Figure 3.3.2.2-8.

(B.3) For worst case conditions (to account for end-of-life, possible errors in system calibration, and Johnson Reference), decrease μ_2 by 12%:

$$\mu_1 = (0.88)\ \mu_2 .$$

μ_1 is the expected telemetered value in mv for A*1BRM or A*2BRM, as applicable.

TABLE 3.3.3.3-3

DETECTABLE STAR LIST

Bright Star Number**	PSI* Channel	PSI2* Channel	Star Astronomical Nomenclature	Location Celestial Latitude	Location Celestial Longitude
2491	-1.35	-1.35	Sirius (A CMA)	-39.599	103.723
2061	-0.83	-0.83	Betelgeuse (A ORI)	-16.031	88.391
2326	-0.77	-0.77	Canopus (A CAR)	-75.827	104.602
5340	-0.55	-0.55	Arcturus (A BOO)	30.754	203.871
5459	-0.46	-0.46	Rigil Kentaurus (A CEN)	-42.587	239.152
6134	-0.23	-0.23	Antares (A SCO)	- 4.567	249.399
1708	-0.17	-0.16	Capella (A AUR)	22.864	81.494
7001	+0.10	0.10	Vega (A LYR)	61.734	284.951
1713	0.20	0.20	Rigel (B ORI)	-31.126	76.466
1457	0.11	0.11	Aldebaran (A TAU)	- 5.469	69.42
2943	0.26	0.26	Procyon (A CMI)	-16.015	115.427
4763	0.41	0.41	Gacrux (G CRU)	-47.828	216.377
472	0.58	0.57	Achernar (A ERI)	-59.377	344.941
5267	0.71	0.71	Hadar (B CEN)	-44.134	233.431
8636	0.56	0.56	Beta Grus (B GRU)	-35.429	321.624
7557	0.75	0.75	Altair (A AQU)	29.304	301.409
4730	0.93	0.93	Acrux (A CRU)	-52.876	221.510
2990	0.83	0.83	Pollux (B GEM)	6.682	112.857
5056	1.08	1.08	Spica (A VIR)	- 2.053	203.478
8728	1.22	1.22	Fomalhaut (A PAU)	-21.132	333.494
4853	1.40	1.38	*Mimosa (B CRU)	-48.636	221.286
7924	1.28	1.28	Deneb (A CYG)	59.907	334.971
337	1.21	1.22	Mirach (B AND)	25.944	29.706
3634	1.34	1.34	Al Suhail (L VEL)	-55.871	160.495
3982	1.47	1.47	*Regulus (A LEO)	0.464	149.468
-	Ref. to Fig. 3.3.3.3-1		Saturn		

Star positions from 1974 American Ephemeris.
Listed in approximate descending order of brightness for silicon detector response as documented in Reference 1.5.21.

* Not reliable for spin rates above 30 rpm.
**Yale University Observatory catalog edited by D. Hoffleik.

TABLE 3.3.3.3-4

OFFSET (=A) AND GAIN (=B) VERSUS SPIN RATE,
BANDPASS AND SELECTED STAR CHANNEL

CHANNEL BANDWIDTH (Hz)	PSI* 2	PSI* 19	PSI2* 2	PSI2* 19
Rate: 4.7 (RPM)	A = 82.213 B = 3.592	45.545 2.586	A = 50.434 B = 3.841	39.239 2.765
10	64.397 3.953	42.640 3.086	26.567 4.192	52.868 3.223
15	41.979 4.168	37.938 3.298	15.612 4.308	27.733 3.473
20	30.829 4.199	35.221 3.407	18.659 4.279	23.366 3.525
25	40.030 4.052	34.372 3.402	43.854 4.163	21.670 3.496
30	23.648 3.841	32.558 3.317	47.734 3.948	25.108 3.385
35	12.346 3.591	37.932 3.110	35.788 3.716	39.612 3.197
40	14.458 3.315	39.468 2.924	48.114 3.433	22.754 3.036
45	7.370 3.090	20.980 2.752	52.238 3.182	36.176 2.824
50	5.150 2.860	36.508 2.532	56.739 2.956	45.753 2.622
55	- 8.778 2.673	40.923 2.367	57.205 2.749	37.527 2.470
60	- 5.208 2.486	36.558 2.234	58.709 2.560	39.649 2.308

TABLE 3.3.3.3-5

MINIMUM STAR BRIGHTNESS LEVEL (AS TELEMETERED) FOR TIME INTERVAL MEASUREMENTS VERSUS SELECTED THRESHOLD LEVEL, BANDPASS, AND STAR CHANNEL

LEVEL	Pd (Note 1)	PSI* 2Hz 0.993	PSI* 19Hz 0.993	PSI2* 2Hz 0.993	PSI2* 19Hz 0.993	PSI* 2Hz 0.999	PSI* 19Hz 0.999	PSI2* 2Hz 0.999	PSI2* 19Hz 0.999
8	V(mv)	184	184	196	194	198	200	211	208
7		218	220	238	237	233	236	253	248
6		252	253	276	273	267	269	290	288
5		296	300	325	324	310	316	340	338
4		460	460	490	508	470	475	500	520
3		705	710	775	780	720	720	785	795
2		1085	1095	1215	1220	1100	1110	1230	1235
1		1695	1700	1920	1920	1710	1715	1930	1940

NOTE 1: Pd = Probability of detection.

3.3.3.3.3 Star Brightness Measurements. Star brightness measurements are telemetered directly from each channel of the star sensor, A*1BRM for the PSI* channel and A*2BRM for the PSI2* channel. To measure the brightness of a given star, the azimuth position of Gate B must be commanded via the ADP (ATQ#5 for ADP No. 1; ATQ#E for ADP No. 2) to encompass the star location within the 11.25 degree gate. Formulation of the command bit structure to position Gate B properly is discussed in 3.3.2.3.5.2. The commanded angle delay is the leading edge of Gate B with respect to the occurrence of the SRR pulse. This same Gate B is used within the ADP to make a time interval measurement to the star pulse occurring within Gate B.

Important distinctions between the brightness measurement and the time interval at measurement should be understood.

For the sake of understanding the brightness measurement, the opening of Gate B can be interpreted in terms of time occurrence as well as azimuthal location. The commanded azimuthal position for Gate B is translated by the Controls Subsystem into a time delay w.r.t. the SRR occurrence, for the opening of Gate B. At the instant that Gate B begins to open, the star brightness detector resets, and begins to look for the peak signal that occurs while Gate B remains open (for the time period equivalent to 11.25° azimuthal angle). At the instant that Gate B closes, the brightness detector stops looking for any signal, and instead, holds the peak signal in storage for nearly one spin period, when Gate B begins to open again, and the brightness detector is reset again.

The telemetered value will be the peak star brightness detected during the last occurrence of Gate B if the telemetry system samples the measurement between gate occurrences. If the telemetry system samples the measurement during the gate occurrence, the telemetered value will be the peak value obtained up to the time of telemetry sampling occurrence. For example, if

the telemetry sampling time occurs instantly after the opening of Gate B, but before a star is brightness detected azimuthly later in Gate B, the telemetered value for brightness would be zero. As a second example, if the TM sampling time occurred between two stars that are within Gate B, the telemetered brightness value will correspond to that star that has already been brightness detected, even though the star yet to be detected may be brighter.

The brightness measurement in the star sensor is independent of the commanded threshold value. For instance, the commanded threshold value may be insensitive to a star within Gate B, thus precluding a time interval measurement w.r.t. the SAK occurance, but the star's brightness will be telemetered (if the TM sampling interval occurs after the star has been brightness detected).

Since only one star channel may be assigned to Gate B at any instant, only that star channel's brightness detector is usable to produce valid telemetry. The other star channel's brightness detector will produce meaningless telemetry.

It follows also that, in the ADP, only one star channel can be selected at a time for the Gated B star time interval measurement. Operationally, it may be desirable to move the location of Gate B, which is 11.25 degrees wide, around a given star to determine if there are any interference sources near that star. Similarly, the threshold setting of each channel can be varied above and below a given setting to determine the optimum setting as well as possible interference effects from dimmer stars.

3.3.3.3.4 Star Mapping. A star map can be attained by moving the location of Gate B to successive angular positions. The peak brightness reading of any detected star will be telemetered directly by the star sensor. The attitude measurement capability of the ADP is used to obtain the azimuth location of a star by selecting the Star 1 input and the elevation of a star by selecting the Star 2 input. Since these measurements are

made within respect to the SRR pulse, it may take up to three spacecraft revolutions to acquire a single star measurement depending upon the selected alternate attitude measurement. Furthermore, if a detected star is not bright enough to exceed the commanded threshold level, the star sensor will not output a star pulse. Consequently, if no pulse is generated at the location of Gate B, the ADP will "hang up" attempting to make the gated star measurement and will no longer alternate between measurement A and B. The telemetered star brightness would then be used to confirm that a detectable star is not present at the gate location, and the gate can be moved to a new azimuth location.

The use of the star mapping capability would be dependent upon the current knowledge of the spacecraft attitude and the probable number of stars that can be detected. If the approximate attitude is known, Gate B can be moved to the probable star locations. If the attitude is totally uncertain, Gate B (11.25 degrees wide) may be moved in 10-degree increments to attain a 360-degree sweep of the sensor field-of-view or until the approximate attitude can be derived. This would minimize the number of gate locations that would have to be commanded and the time involved since only three to five stars can be expected to be detected in any single spacecraft sweep, i.e., spin revolution. In all conditions, set the star detectable threshold to level 7 (to level 8 only if no stars are detected with level 7), select the ATTM measurements to be the same (SRR to PSI*, or SRR to PSI 2*), and allow telemetry to gather at least 10 revolutions of data (5 samples of each of two measurements). A suggested flow chart to perform a star map is shown in Figure 3.3.3.3-1.

3.3.3.3.5 Star Acquisition. Star acquisition is a process by which the PLL will lock onto the first detectable star anywhere in the spin period irrespective of the Location of Gate B. It is intended to be used to lock-up the PLL in the absence of a sun pulse. Conceivably, this would only be necessary if the spacecraft spin axis is

within 10 degrees of the sunline (non-standard attitude) or during an eclipse condition when either the inertial reference is lost due to an incorrect transfer to a star reference or it becomes necessary to transfer to the redundant ADP.

The sequence of events to lock to a star pulse is as follows:

(a) Transmit the ADP mode command that contains the following information:
 (1) Star acquisition bit = logical 1
 (2) SRR advance bit = logical 1
 (3) Star Gate A and Star Gate B channel select are both equal to the desired star channel (Star 1 or Star 2). NOTE: Star Gate B is selected in anticipation of making a star brightness measurement after the PLL is phase-locked to the star pulse.

(b) The star acquisition bit (logical 1) performs two functions:
 (1) Widens star Gate A in the PLL to 45 degrees.
 (2) Enables the SRR advance logic to accept the first star pulse appearing in star Gate A channel select.

(c) The first star pulse that appears on the selected star Gate A channel is used to phase reset the PLL counters and the Hold Zero Phase Error flip flop, thus forcing that star to be in the middle of the 45-degree Gate A.

(d) The PLL then proceeds to produce the correct phase and frequency to lock on this star pulse. The wide Gate A can accommodate the phase transients in this process, which can be in excess of the normal 11.25 degree gate.

Formulation of the command bit structure and the telemetry response is presented in 3.3.4.

3.3.3.4 DCE Normal Operating Mode (HGA Despin Control). Despin control of the HGA can be accomplished by

operating either or two DCEs. After unit turn-on, the unit should be commanded to a known configuration as described in 3.3.2.4.3. This sequence will allow the despin processing electronics to achieve phase synchronization with the F_S pulse from the ADP. If the DCE is turned on after the PLL in the ADP has attained phase lock, the unit can be immediately operated in any of its modes. If the PLL is not in lock, the DCE can still be commanded on, provided that the despin motor drive is not commanded on until the PLL has attained lock. This will prevent the HGA azimuth control from tracking the phase error transients of the PLL. Typical telemetry response for the DCE is shown in Table 3.3.3.4-1.

Azimuth control of the antenna is then accomplished by the azimuth position command (DCEQ5 or DCEQE), or by the azimuth pointing slew command (DCEQ1 or DCEQ4) as described in 3.3.2.4.4.

Since the despin control uses the F_S pulse from the ADP for the pointing reference, it is necessary to command "Phase Reset Disable" (DCEQ6 or DCEQF) to prevent antenna pointing transients, prior to changing to a new inertial reference in the ADP.

Upon completion of the transfer, a new azimuth offset value which compensates for the new inertial reference location must be commanded. "Phase Reset Enable" ((DCEB6-ENAQ) or (DCEQF-ENAB)) can then be commanded to return pointing control to the normal mode. This azimuth offset value must take into account all the biases between the HGA boresight and the selected reference sensor to properly position the despun platform, as discussed in 3.3.2.4.4.

If a spin change or a spin trim maneuver is to be performed, the PLL will also generate phase transients, thus affecting antenna pointing. By commanding "Phase Reset Disable" until the maneuver is completed and the PLL again acquires lock, the antenna will remain pointed in the

TABLE 3.3.3.4-1

TELEMETRY INDICATIONS FOR DCE RESPONSE

Mnemonic for Pertinent TM	TM Channel	Typical Status or Expected Value
ADCE1T	DCE1 Temperature	_____ ± _____ °F if ON _____ ± _____ °F if OFF
ADCE1V	DCE1 Power Supply Voltage	2.5 V if ON 0.0V if OFF
APOS1S	HGA Positioner Electronics 1 ON/OFF	Logic "1" if commanded on (DCEQ6 selection)
AMTR1S	Despin Motor Drive 1 ON/OFF	Logic "1" if commanded on (MTR19 or MTRA9 selection with DCE1 on)
ADCE2T	DCE2 Temperature	_____ ± _____ °F if ON _____ ± _____ °F if OFF
ADCE2V	DCE2 Power Supply Voltage	2.5V if ON 0.0V if OFF
APOS2S	HGA Positioner Electronics 2 ON/OFF	Logic "1" if commanded on (DCEQF selection)
AMTR2S	Despin motor Drive 2 ON/OFF	Logic "1" if commanded on (MTR29 or MTRB9 with DCE2 ON)
AMORTI	Despin Motor Drive Current	_____ ± _____ amps
ARATEQ	Rate Torque	_____ ± _____
APOSTQ	Position Torque	_____ ± _____
AVPDRD	RAM Variable Phase Delay	Dependent on commanded RAM VPD rate (DCEQ2 or DCEQB) and time occurrence of RAM VPD reset command (DCEQ6 or DCEQF).
AVPDND	NADIR Variable Phase Delay	Dependent on commanded NADIR VPD rate (DCEQ3 or DCEQC) and time occurrence of NADIR VPD reset command (DCEQ6 or DCEQF).
ADSPNS	Despin Phase Reset Enable/Disable Status	Logic "1" if enabled (DCEQ6 or DCEQF selection)

desired direction. This is more fully discussed in 3.3.3.5.

3.3.3.5 JCE Control. All command sequences for a thrust maneuver including the latch valve control should be carefully checked before the commands are transmitted. The proper receipt of each command should be carefully verified by telemetry prior to initiating the thrusting sequence. (Status should be as shown in Table 3.3.3.5-1). It is highly recommended that all maneuvers be performed within view of a ground station so that prompt corrective action can be taken, if necessary. For any maneuver sequence, the attitude measurements should be selected to measure spin period (SRR to SRR) and sun aspect angle (SRR to ϕ2) to provide a direct indication of expected performance as well as any anomalous thruster behavior and its effect on spin rate or spacecraft attitude. The command sequence should include a back-up command, "JCE Buffer Output Disable," (AT0 1ϕ or AT0ϕJ) following normal expected completion of the jet firing.

During periods when there is no ground station coverage, it is recommended that both latch valves be closed to protect against any possible failure.

If a maneuver sequence has to be performed in a manner where the actual magnitude of each maneuver is not verified prior to the start of a maneuver, then the magnitude should be transmitted at least two times. This will preclude the possibility of a command rejection causing an improper magnitude execution. The repeated transmission is to insure that the countdown register is changed to the desired value since it would have terminated at the maximum count after completion of the previous maneuver.

A JCE quantitative command or mode command should not be transmitted while a desired maneuver is in progress. The commands will be accepted and will immediately reconfigure the JCE accordingly.

TABLE 3.3.3.5-1 TELEMETRY INDICATIONS FOR ADP STATUS FOR THRUST MANEUVERS

Mnemonic for Pertinent TM	Status Prior to Executing Maneuver Sequence	Status After Completion of Maneuver Sequence	Remarks
ATTM1Z and ATTM2Z	o SRR to SRR equal to expected spin period	o SRR to SRR equal to expected spin period	o Change dependent on magnitude of spin maneuver
	o SRR to PSI 2 equal to $90^{\circ} \pm 2^{\circ}$	o SRR to PSI 2 equal to expected sun aspect angle $\pm 2^{\circ}$	o Change dependent on magnitude of precession maneuver
AJMAGC	o JCE countdown at commanded count	o JCE countdown at maximum count of 4095	
APULLS	o Pulse width selection as commanded	o Pulse width selection as commanded	
AMAGCS	o Pulse time count selection as commanded	o Pulse time count selection as commanded	
AJETMS	o Pulse fire mode selection as commanded	o Pulse fire mode selection as commanded	
ASDETS	o Spin rate detector enabled	o Spin rate detector enabled	
AJETSS	o Jets selection as commanded	o Jets selection as commanded	
AJCEFS	o JCE fire status disabled	o JCE fire status disabled	o JCE fire status becomes enabled only for actual firing duration
A*GASS	o Star Gate A channel select as commanded	o Star Gate A channel select as commanded	
A*GBSS	o Star Gate B channel select as commanded	o Star Gate B channel select as commanded	

desired direction. This is more fully discussed in 3.3.3.5.

3.3.3.5 JCE Control. All command sequences for a thrust maneuver including the latch valve control should be carefully checked before the commands are transmitted. The proper receipt of each command should be carefully verified by telemetry prior to initiating the thrusting sequence. (Status should be as shown in Table 3.3.3.5-1). It is highly recommended that all maneuvers be performed within view of a ground station so that prompt corrective action can be taken, if necessary. For any maneuver sequence, the attitude measurements should be selected to measure spin period (SRR to SRR) and sun aspect angle (SRR to $\phi 2$) to provide a direct indication of expected performance as well as any anomalous thruster behavior and its effect on spin rate or spacecraft attitude. The command sequence should include a back-up command, "JCE Buffer Output Disable," (ATQ1Ø or ATQØJ) following normal expected completion of the jet firing.

During periods when there is no ground station coverage, it is recommended that both latch valves be closed to protect against any possible failure.

If a maneuver sequence has to be performed in a manner where the actual magnitude of each maneuver is not verified prior to the start of a maneuver, then the magnitude should be transmitted at least two times. This will preclude the possibility of a command rejection causing an improper magnitude execution. The repeated transmission is to insure that the countdown register is changed to the desired value since it would have terminated at the maximum count after completion of the previous maneuver.

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ATTM1Z and ATTM2Z	o SRR to SRR equal to expected spin period	o SRR to SRR equal to expected spin period	o Change dependent on magnitude of spin maneuver
	o SRR to PSI 2 equal to 90° ± 2°	o SRR to PSI 2 equal to expected sun aspect angle ± 2°	o Change dependent on magnitude of precession maneuver
AJMAGC	o JCE countdown at commanded count	o JCE countdown at maximum count of 4095	
APULLS	o Pulse width selection as commanded	o Pulse width selection as commanded	
AMAGCS	o Pulse time count selection as commanded	o Pulse time count selection as commanded	
AJETMS	o Pulse fire mode selection as commanded	o Pulse fire mode selection as commanded	
ASDETS	o Spin rate detector enabled	o Spin rate detector enabled	
AJETSS	o Jets selection as commanded	o Jets selection as commanded	
AJCEFS	o JCE fire status disabled	o JCE fire status disabled	o JCE fire status becomes enabled only for actual firing duration
A*GASS	o Star Gate A channel select as commanded	o Star Gate A channel select as commanded	
A*GBSS	o Star Gate B channel select as commanded	o Star Gate B channel select as commanded	

TABLE 3.3.3.5-1 TELEMETRY INDICATIONS FOR ADP STATUS FOR THRUST MANEUVERS (cont'd)

Mnemonic for Pertinent TM	Status Prior to Executing Maneuver Sequence	Status After Completion of Maneuver Sequence	Remarks
ASUNSS	o Sun sensor selection as commanded	o Sun sensor selection as commanded	
ASRRMS	o Sun selected as SRR	o Sun selected as SRR	
A*ACQS	o Star normal mode selected	o Star normal mode selected	
ASPINS	o Spin range selection as commanded	o Spin range selection as commanded	
ALOLES	o PLL loss of lock (per Table 3.3.3.5-2)	o PLL loss of lock selection remains unchanged	
ASUNCS	o Sun gate enable/disable status (per Table 3.3.3.5-2)	o Sun gate enable/disable selection remains unchanged	
ADVANS	o SRR advance set to normal	o SRR advance set to normal	
ARIPAD	o RIP delay as commanded	o RIP delay as commanded	
ALOCKS	o In lock status	o In lock status	With PLL loss of lock detection inhibited, status will still show "in lock", although SRR may be outside Gate A.
AMESSC	o Normally = 0 counts	o Dependent on maneuver effect on PLL operation	
AACSAD	o ACS angle delay as commanded	o ACS angle delay as commanded	

TABLE 3.3.3.5-1 TELEMETRY INDICATIONS FOR ADP STATUS FOR THRUST MANEUVERS (cont'd)

Mnemonic for Pertinent TM	Status Prior to Executing Maneuver Sequence	Status After Completion of Maneuver Sequence	Remarks
ASIMSZ	o PLL spin period as commanded	o PLL spin period as commanded	
AJCE1S	o JCE 1 buffer output status = 1 (ADP 1 ON); = 0 (ADP 2 ON)	o JCE 1 buffer output = 0	
AJCE2S	o JCE 2 buffer output status = 0 (ADP 1 ON); = 1 (ADP 2 ON)	o JCE 2 buffer output = 0	

Equally important to selecting the DCE configuration is establishing the proper configuration of the PLL for a particular maneuver. One of the basic determinations is whether it is necessary for the PLL to remain in lock for successful execution of the maneuver, in which case the PLL loss of lock function and the sun gate must be enabled for the maneuver. This would apply for all pulse firing modes where the PLL is used to generate the angle delay for the azimuth phase angle, but would exclude spin speed changes using the pulse mode. In operating axial thrusters in the alternate fire mode, the PLL phase accuracy is not necessary for the start fire angle, but is necessary to generate the 180-degree pulse separation (pulse firings 180° apart) to provide cancellation of precession inputs in order to preclude an unwanted attitude disturbance. Again, in this case, the PLL loss of lock function and the sun gate must be enabled.

All continuous firing initiations and terminations are accomplished by time alone, and are independent of the PLL operation. However, when using the axial thrusters in the continuous fire mode, the spin rate should remain essentially constant, and additional failure protection can be obtained by using the loss of lock detection capability, since the PLL should stay in lock during this maneuver. Under such circumstances, any unwanted radial thruster firing (due to a logic failure or improper command selection) would produce a spin change, causing the PLL to lose lock and terminate the firing sequence.

A summary of the use of PLL loss of logic for the various maneuver sequences is presented in Table 3.3.3.5-2.

Associated with the PLL performance during any maneuver sequence is the azimuth pointing control of the HGA. In the normal operating mode, the DCE uses the 2^{32} F_S and the F_S from the PLL as well as the shaft angle encoder output (2^7 F_S) to control inertial pointing of the HGA. Transients

TABLE 3.3.3.5-2

USE OF PLL LOSS OF LOCK FOR THRUST MANEUVERS

Maneuver Type	PLL Loss of Lock Selection	Sun Gate Selection	Remarks
Continuous Radial ($\Delta W \equiv$ Spin rate change)	Disabled	Disabled	PLL response unable to maintain SRR within Gate A during large spin changes. Removal of gate allows reference pulse to be detected once per revolution for proper operation of spin rate detector and preventing erroneous termination.
Continuous Axial ($\Delta V \equiv$ Velocity change)	Enabled	Enabled	Loss of lock detection not required for maneuver accuracy, but protects against a radial thruster failure.
Pulsed Axial, 1 per Rev. ($\Delta P \equiv$ precession)	Enabled	Enabled	Loss of lock detection required for phase angle accuracy; protects against a radial thruster failure.
Pulsed Axial, 2 pulses per Rev. (ΔV) (180° intervals)	Enabled	Enabled	Loss of lock detection required to maintain 180-degree phase separation between pulses.
Pulsed Radial Pair (ΔV)	Enabled	Enabled	Loss of Lock detection required for phase angle accuracy; protects against a radial thruster failure.
Pulsed Radial (512 ms) (ΔW)	Disabled	Disabled	Loss of lock detection disabled to prevent PLL loss of lock termination.
Pulsed Radial (128 ms) (ΔW)	Enabled	Enabled	PLL can track phase angle change in either normal or alternate fire mode.

in the PLL outputs caused by small spin speed changes due to axial thruster misalignment or commanded spin trim maneuvers using the pulsed mode will be propagated into the DCE, causing antenna pointing transients. This effect can be minimized if "Phase Reset Disable," (DCEQ6 or DCEQF) is commanded for the duration of the maneuver.

The PLL has four discrete operating ranges as discussed in 3.3.2.3.4. When a spin change is commanded that causes transition of the spacecraft into a new spin range, the command sequence must include an ADP mode select command to the new spin range. When the spin rate is increased to 15 rpm or higher, the PLL will reacquire lock (based on unit test data) within 120 seconds maximum after the maneuver is completed and the proper spin range has been selected. When the spin rate is decreased to 5 rpm, it may take up to 8 minutes for PLL to reacquire lock.

Normally, the midrange sun sensor will be selected to provide the sun reference pulses to the ADP. Since its field-of-view (FOV) is limited to ±45 degrees, any precession maneuver outside this range will require that the appropriate extended range sun sensor be selected to complete the maneuver. However, the extended range sun sensor can be used for the entire sequence, since its FOV extends past the 90-degree sunline angle (see Figure 3.3.2.1-5).

The disadvantage of this last approach is the loss of resolution in the sun aspect angle in the initial part of the maneuver. The disadvantage of the first approach is that it would require a two-step sequence. This assumes that the bias angle between the two sensors would have to be compensated for to achieve the desired accuracy. If the bias error was minimal and could be accommodated, the transfer could be done while the maneuver was in progress provided it was controlled so as not to occur while a sun pulse is being generated. This could result in double pulses being provided to the PLL, and cause it to

slew off frequency, thereby generating a loss of lock condition. Therefore, it is recommended that while a sun sensor is acting as the SRB, it should NOT be replaced by a different sun sensor during a maneuver.

After the spacecraft has been spun up to 6.5 rpm (just prior to magnetometer boom deployment), spin rate detection should be enabled (SEDX selection of ATQ#2 or ATQ#B) to provide that protective capability for the remainder of the mission.

The detector uses a 15.625 Hz clock rate to measure the time interval between successive PLL reference pulses. When the PLL is operated in the gated mode, the gate will exclude the reference pulse, if the PLL phase error is greater than half the gate's width, (11.25° or 45°). The loss of the reference pulse for more than 16.256 seconds will appear as a low rpm condition (<3.7 rpm) and terminate any maneuver when this detection is enabled. To preclude premature termination of a spin change maneuver for this reason, large spin change maneuvers, which will create large PLL phase errors, should be performed with the sun gate disabled. The narrow pulse firing mode for spin changes will not cause large phase errors and thus, can be used in the gated mode.

The P_s phase error generated during spin maneuvers using the pulsed firing mode is shown in Figures 3.3.3.5-1 and 3.3.3.5-2 for the narrow (128 ms) pulse mode and in Figures 3.3.3.5-3 and 3.3.3.5-4 for the wide (512 ms) pulse mode.

The typical command sequence for any maneuver is shown in Table 3.3.3.5-3. Use of the PLL loss of lock detection and sun gate mode should be controlled as previously shown in Table 3.3.3.5-1. The table illustrates only a single maneuver. If the sequence is to be followed by another maneuver, the listed commands following the completion of the first maneuver can be revised to configure the JCE accordingly, rather than

configuring to an intermediate "SAFE" configuration.

3.3.3.6 Attitude Determination. Attitude determination in its simplest form requires sun aspect measurements and star aspect measurements or Multiple Star aspects measurements.

The electrical signals from the sensors are processed by a common attitude data processor (ADP) unit. The time interval between pulses are measured and transmitted to the Ground Station for computation of spacecraft attitude. The understanding of how these time interval measurements are made will facilitate a better overview of the attitude determination problem.

A total of seven different time separation measurements may be computed; however, only two may be selected at a time, in any combination. What is actually measured is shown in Table 3.3.3.6-1.

Several types of measurement are unique and are affected by the relation of the TM sample rate with the spin rate. The SRR - Minor Frame Start is one such measurement. Whenever the spacecraft rotor rate is higher than the minor frame rate (Figure 3.3.3.6-1), two or more SRR pulses may be expected between two minor frame rate pulses. The ATTM logic is implemented to calculate the smallest time separation between these pulses. This is accomplished by always resetting the ATTM counter with the SRR pulse, if the measurement has not been terminated by the detection of the selected reference pulse. Another interesting point illustrated by Figure 3.3.3.6-1 is that the completion of the measurement does not automatically allow the next measurement to start. The initiation of the readout sequence, i.e., the leading edge of the first of three read envelopes, will parallel transfer the measured data to a TM register for readout during that word time and the next two word times (24 bits total). This same event also allows the alternate measurement to be started at the occurrence of the next SRR pulse. This alternate

TABLE 3.3.3.5-3

TYPICAL MANEUVER SEQUENCE

Step #	Command Mnemonic	Command Title	Remarks
1.	ATQ02 or ATQ0B	Jet Control	Select jets to be fired, firing mode, and spin rate detection.
2.	ATQ05 or ATQ0E	ACS Angle Delay Magnitude	Only necessary when using pulse fire mode.
3.	ATQ04 or ATQ0D	Jet Countdown	Select desired pulse count or time count.
4.	ATQ03 or ATQ0C	ADP Mode Select	Only necessary to disable sun gate and PLL loss of logic detection.
5.	DCEQ6 or DCEQF	DCE Mode Command	Disable Phase reset to prevent PLL outputs from affecting HGA pointing.
6.	VAL19 or VALA9	Latch Valve 1 OPEN	Open latch valve No. 1 to fire 3R, 4R, A5 or A7.
	AND/OR:		
	VAL29 or VALB9	Latch Valve 2 OPEN	Open latch valve No. 2 to fire 1R, 2R, or A6.
7.	ATQ09 or ATQ0I	JCE Buffer Output ENABLE	Enables JCE output buffers.
8.	ATQ11 or ATQ0K	Jet Fire Interlock	Enables JCE firing logic.
9.	ATQ12 or ATQ0L	Jet Fire	Initiates thruster firing.
10.	ATQ10 or ATQ0J	JCE Power Buffer Output Disable	Command to be sent after a delay compatible with commanded magnitude.
11.	VAL10 or VALA0	Latch Valve 1 CLOSE	As deemed necessary to provide adequate safeguard after completion of maneuver.
	AND/OR		
	VAL20 or VALB0	Latch Valve 2 CLOSE	

TABLE 3.3.3.5-3 (Continued)
TYPICAL MANEUVER SEQUENCE

Step #	Command Mnemonic	Command Title	Remarks
12.	ATQØ3 or ATQØC	ADP Mode Select	Only necessary at this time if selection of different spin range is required.
13.	ATQØ4 or ATQØD	Jet Countdown	Set to zero count, or to account for the next maneuver if maneuver is to be performed immediately.
14.	ATQØ2 or ATQØB	Jet Control	Set all jet select bits to zero or to the jet combination for the next maneuver if a subsequent maneuver is to be performed immediately.
15.	ATQØ3 or ATQØC	ADP Mode Select	Only necessary to re-establish sun gate and PLL loss of lock detection if disabled in Step No. 4. Command to be sent after PLL has obtained lock.
16.	DCEQ6 or DCEQF	DCE Mode Command	Enable phase reset if it had been disabled in Step No. 5.

TABLE 3.3.3.6-1

MEASURED PARAMETERS
VIA ALL COMBINATIONS OF ATTM MEASUREMENTS

Commanded Measurement	Measurement is W.R.T. the Following Selected Reference:			
	Sun (ψ Only)	Star Gate A Channel 1 (ψ*)	Star Gate A Channel 2 (ψ_2*)	Sim. SRR
1. SRR to SRR	Actual Spin Period	Actual Spin Period	Actual Spin Period	Simulated Spin Period
2. SRR - Minor Frame Start	Determination of Azimuth Reference (In TM Clock Time)			
3. SRR - RIP	Science Ref.	Science Ref.	Science Ref.	Science Ref.
4. SRR - MIP	HGA Internal Azimuth Despin Position Location			Simulated Drift Rate
5A. SRR - Gated Star B (Channel 1)	Separation of Star in Gate B w.r.t. the Sun.	Separation of Star in Gate B w.r.t. star in Gate A.	Aspect angle of Star in Gate B w.r.t. star in Gate A, measured in plane parallel to ψ_2* slit.	Simulated Drift Rate
5B. SRR - Gated Star B (Channel 2)	Aspect angle of Star in Gate B w.r.t. the sun.	Aspect angle of Star in Gate B w.r.t. star in Gate A.	Biased aspect angle of star in Gate B w.r.t. star in Gate A.	Simulated Drift Rate
6. SRR - ψ_2	Sun Aspect Angle	Aspect angle of sun w.r.t. star in Gate A	Biased aspect angle of sun w.r.t. star in Gate A.	Simulated Drift Rate
7. SRR - Major Frame Start	TM Determination of Azimuth Reference (Primarily used when in the Subcom mode)			

measurement, if completed, cannot be transferred to the TM register until initiation of the next group of three read envelopes. This insures that the previous measurement is read out at least once. If measurement A is not completed at the initiation of the read out sequence, measurement B will be read out again. However, measurement B (SRR to minor frame start) was measured to the start of the minor frame and the data is only valid in the first minor frame it is read out.

Figure 3.3.3.6-2 on the other hand, represents a condition of high TM sample rate, i.e., many minor frames between successive SRR pulses. The resultant shown here, is that a lot of repeated measurements are sampled. All repeated measurements of SRR to SRR (measurement A) are valid, whereas, only the readout of measurement B in the first minor frame is valid.

3.3.3.7 Roll Reference Transfer. The following is an overview description of transfers. Detailed command sequences are shown in Section 4.3.3.

3.3.3.7.1 Sun Sensor to Sun Sensor. Typically, the sun sensor will be used to provide the roll reference pulse for the PLL. When the attitude of the spacecraft is changed, which requires the use of another sun sensor, e.g., extended range sensor, only the appropriate sun sensor need be selected by command (ATQ03 or ATQ0C). If there is a bias difference between the pulse outputs, there will be a momentary phase transient in the PLL outputs, the magnitude of which is equal to the sensor bias. The change in bias can be measured by making an attitude measurement to a known star before and after the transfer process. The RIP location should be corrected if the bias is greater than a one-bit change (0.351 degree) in the RIP angle delay capability.

3.3.3.7.2 Sun Sensor to Star Sensor. The transfer from a sun reference to a star reference requires that Gate B be commanded to an angular location containing the desired star. The star sensor threshold setting should then be set commensurate with the predicted star intensity to achieve an

acceptable probability of detection and false alarm rate. The transfer is accomplished by using the SRR Advance bit and selecting the desired star channel (ATQ$3 or ATQ$C).

The transfer will change the phasing of all the PLL outputs. The RIP location, to which the RAM and NADIR are referenced, the HGA pointing direction and the location of Gate B are affected and require additional commands to restore operation to normal.

3.3.3.7.3 Star Sensor to Sun Sensor. The transfer from a star reference to a sun reference is made easier because the sun is an unambiguous reference requiring no spin gate for initial capture. Consideration to the phase changes in the PLL outputs as discussed in the preceeding paragraph is still necessary.

3.3.3.7.4 Star Reference to Star Reference. When transferring from one star reference to another star reference, the new star reference must be located within Gate B. Configuring the ADP mode command (ATQ$3 or ATQ$C) to perform an SRR advance will cause the SRR to transfer to the new star. If the transfer involves transferring to the alternate star channel, the ADP mode command (ATQ$3 or ATQ$C) must include selection of the desired channel for Gate A as well as the SRR advance command bit.

3.3.3.7.5 Use of the Free Run (Simulated SRR) Mode During Reference Transfer. In orbits 24 to 118 (Early launch), the sun becomes eclipsed during the periapsis pass and cannot be used as a roll reference. Furthermore, there are approximately 10 orbits when the desired star reference, Rigel Kentaurus is obscured for up to six minutes by the planet Venus at the onset of the solar eclipse phase of the trajectory. Since it is not possible to transfer directly from a sun reference to this star reference, it is necessary to transfer to the free-run mode until the star becomes visible. It is essential that the simulated spin period (ATQ$7 and ATQ$8) or (ATQ$G and ATQ$H) be commanded to an accurate value of

the actual spin period for this sequence. Any errors will cause the inertial location of star Gate B to change each revolution. If the spin period error is greater than 39 ms (for a nominal spin period of 12 sec), the accumulated error after 6 minutes will have moved the nominal gate location 5.5 degrees. This will cause the desired star to be outside Gate B. The command sequence which is controlled by the SCL will transfer the ADP to the star mode, but since no star is visible in the gate, no transfer will take place. Consequently, the ADP will remain in the free-run mode until commanded to the sun mode following the pass. This requires then that the actual spin period be accurately measured when an inertial reference is available. Furthermore, any spin speed changes, due to rotor contraction resulting from a temperature change during the eclipse or interaction with the atmosphere during the pass, should be measured and used to correct the commanded spin period accordingly.

The command sequence to use the free run mode as an intermediate mode consists of selecting the simulated SRR reference, and then inhibiting PLL loss of lock (ATQ$3 or ATQ$C). When the star becomes visible, the appropriate sequence in Section 4.3.3 should be performed.

3.3.3.8 HGA Elevation Control. After power has been applied to the HGA drive electronics (DCEQ6 or DCEQF), elevation positioning of the HGA is controlled by the use of elevation rate commands to the DCE (DCEQ4 or DCEQD), the command quantization permits the dish to be moved at rates from zero to 40 degrees per minute in increments of 0.00975 degree per minute. Time control can be executed with 125 millisecond resolution using the stored command capability of the Command Processor. Either DCE can operate the HGA elevation positioner. When the DCE is initially turned on, the phasing of the elevation drive signals with respect to the four phase elevation positioner motor can be out of synchronization by up to two pulses in either direction. When the HGA is commanded on with zero slew rate commanded, the dish can step in

either direction a maximum of two steps to achieve synchronization. However, once the signals are in synchronization with the motor, they will remain in synchronization unless a mechanical end stop is contacted or the DCE is turned off, which would again produce up to a two-step ambiguity. Synchronization can be maintained when turning off the HGA elevation power output (DCEQ6 or DCEQF) by first commanding a zero elevation slew rate. After power is turned back on again, a slew rate can then be commanded without incurring any step ambiguity. If the positioner is commanded off while a slew is in progress or a slew is commanded while the positioner is OFF, the two-step ambiguity will occur when power is commanded back on.

Positioning of the HGA to a desired angle may be accomplished by timing the duration of a rate command and then commanding a zero slew rate. Direction of slew is controlled by use of the polarity bit in the HGA elevation positioner slew command (DCEQ4 or DCEQD). Formulation of the proper bit structure of this command is presented in 3.3.2.4.4. Elevation control of the antenna for the occultation experiment is discussed in 4.3.4.

The command to move the antenna in elevation is executed immediately on the spacecraft with no significant delay in mechanical response. The telemetered HGA elevation positon (AELEVD) provides eight bits resolution.

3.3.3.9 Science Output Control

3.3.3.9.1 General. All of the reference pulses provided by the controls subsystem for use by the scientific instruments are dependent upon the operation of the PLL in the ADP. This implies that the PLL must be in lock with the selected roll reference for proper performance of any of these signals. When the simulated SRR is used as a reference, the phase accuracy of any output is directly dependent on how accurately the commanded simulated spin period matches the actual spin period.

3.3.3.9.2 Roll Reference Signal (F_S). The roll reference signal (F_S) is an output of the PLL that occurs once per spin revolution synchronous with the occurrence of the selected roll reference (a small bias can exist between the SRR and the F_S, the magnitude and polarity of which can be calibrated). The time occurence of this pulse in inertial coordinates is dependent upon the selected roll reference, such as the sun or a particular star. A discussion of these phase changes is presented in 3.3.2.4.4.

The occurrence of the F_S pulse can be related to telemetry by making the measurement of SRR to Minor Frame Start or to Major Frame Start (ATTM1Z and ATTM2Z). The phase difference between the SRR and the F_S event time can be calibrated at each spin speed by setting the Roll Index delay magnitude to zero which makes the occurrence of the RIP coincident with F_S. By making the measurement SRR to RIP, the phase delay through the PLL is then measured.

3.3.3.9.3 Spin Period Section Pulses (2^{10} F_S). The spin period sector pulses are generated by the PLL as a continuous pulse train having 1024 near-equally spaced pulses per spin revolution.

Phase Lock of the F_S pulse to the SRR is controlled in the PLL by correcting the VCO frequency as a function of the magnitude and polarity of the phase difference between the F_S and the SRR pulses. The phase error correction is completed within 1.8 seconds following receipt of both pulses (SRR and F_S). This means that the pulse to pulse jitter of the sector pulses caused by the VCO frequency correction will be at a maximum for the first 1.8 seconds following receipt of the later pulse. The maximum jitter is expected to be less than 0.03 degree. Following the 1.8 second correction interval, the pulse to pulse jitter is essentially unmeasurable.

3.3.3.9.4 Roll Index Pulse (RIP). The RIP is a pulse that occurs once per spin revolution at a commanded angular delay from the occurrence of the F_S

pulse. The angular delay is counted using the 1024 sector pulses (2^{10} F_s), providing an angular increment of 360°/1024 or 0.351 degree. The RIP pulse occurrence is coincident with the sector pulse at the commanded angular delay. The actual location can be measured with respect to the input reference by making the measurement of SRR to RIP. Whenever the selected roll reference is changed or a different star is selected when using the star reference, the inertial location of the F_s pulse will change. Since the RIP will shift in phase accordingly, a new angular delay must be commanded at the time of transfer to maintain the RIP at the same inertial location. However, unless the transfer command and the new RIP delay are timed to occur between the occurrence of the old reference pulse and the new reference pulse, the RIP may occur at the wrong angle for one revolution during the transfer process. The fact that the commanded RIP delay is loaded into a working register, each revolution at the occurrence of the F_s pulse can cause the one revolution ambiguity. The timing relationship further clarifying this point can be seen in Figure 3.3.3.9-1.

3.3.3.9.5 Variable Phase Delay Pulses (NAM and NADIR). The angular delay value of the NAM and NADIR pulses with respect to RIP is updated once every 16 milliseconds (4 milliseconds per operation) according to the commanded rate profile for each. The computed value for each at the time of RIP occurrence is loaded into a register and then counted out during the next spacecraft revolution using 2^{12} F_s pulses from the PLL in the ADP. Any erroneous occurrence of the RIP, as discussed previously, or transitory behavior of the 2^{12} F_s will cause errors in the generation of the NAM and NADIR pulses during that particular revolution. Furthermore, in normal operation, the angular error associated with each output increases in proportion to the pulse location with respect to RIP. The magnitude of the error is dependent on the incremental angular change each revolution. Conceivably, if the pulse was to occur just prior to a new RIP occurrence, the angular error would be approximately the total

angular change that would occur during the next revolution. Without consideration to programming compensating biases, the most accurate pulse location then will occur when it is closest to the RIP. The error caused by this mechanization can be reduced by generating a rate profile which will cause the correct phase angle for the next revolution to be generated at the time occurrence of the RIP. This angle would then be loaded into the working register at RIP occurrence and would then be clocked out at the desired angle during the next revolution.

The telemetered values of RAM VPD (AVPDRD) and NADIR VPD (AVPDND) are the prevailing values that apply at the instant of telemetry readout.

Information on controlling these outputs by command is presented in 3.3.2.4.5. Operational constraints regarding these outputs are provided in 3.3.3.11.3.

3.3.3.10 Unit Redundancy. Redundancy within the control subsystem is achieved by using the following complement of units:

(a) Two ADPs
(b) Two DCEs
(c) One star sensor having two channels which are electrically redundant.
(d) Two sets of sun sensors with each set containing a mid-range and a lower and upper extended range sun sensor.
(e) One RAPTA with a motor assembly that has dual control.

The interconnection of these units and the interface with other units to achieve redundancy is presented in Table 3.3.3.10-1. It is emphasized that each ADP configure command interfaces with only one COH, and each DCE configure command interfaces with only one COH, whereas each discrete command interfaces with either of two COHs.

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TABLE 3.3.3.10-1.
CONTROL UNIT REDUNDANCY MECHANIZATION

Unit	Interfacing Units	Remarks
ADP	COMs	
ADP	DIMs	
ADP	TM Processors	See Section 3.5 for telemetry clock signals.
ADP	Sun Sensors	Each ADP is handwired to a separate set of sun sensors (mid-range and extended range) (no crosstrapping).
ADP	Star Sensor	Each ADP can operate with either star sensor channel (full crosstrapping).
ADP	Scientific Instruments	Each ADP provides separate outputs to each scientific instrument (full crosstrapping).
ADP	DCE	Either ADP can operate with either DCE (full crosstrapping).
Star Sensor	COMs	Any discrete command or quantitative command can be sent to either star sensor channel from either COM 1 or COM 2 (full crosstrapping).
Star Sensor	DIMs	• PSI* and PSI2* brightness measurements are telemetered on redundant subcom channels using DIM 6 and DIM 7. PSI* brightness is telemetered in the minor frame mode only using DIM 6 (not redundant). PSI2* brightness is telemetered in the minor frame mode only using DIM 7 (not redundant). • All other measurements are processed in a non-redundant manner with each measurement assigned to a single TM channel.
DCE	COMs	• Any discrete command to a DCE can be sent by either COM 1 or COM 2. • A quantitative command can only be sent to DCE 1 from COM 1 and to DCE 2 from COM 2.
DCE	DIMs	
DCE	HGA	Control signals from each DCE are handwired together so that either DCE can operate the HGA.
DCE	BAPTA Motor	Each DCE controls one set of the dual motor windings (no crosstrapping).
DCE	TM Processors	See Section 3.5 for telemetry clock signals.
HGA	DIMs	• HGA Elevation position data (AELEVD) is telemetered via both DIM 3 and DIM 5. • HGA position motor temperature (APOSMT) is telemetered only via DIM 5 (non-redundant).
BAPTA	DIMs	BAPTA temperature measurements are processed in a non-redundant manner. BAPTA inner race temperature (ABAPIT) is provided to a single TM channel in DIM 5. BAPTA outer race temperature (ABAPOT) is provided to a single TM channel in DIM 3.
Magnetometer Deployment Mechanism	DIMs	Magnetometer boom deployment status (SBOOMS) is non-redundant and is provided to a single TM channel in DIM 2.

3.3.3.11 Operational Restrictions

3.3.3.11.1 Operation of two ADPs Simultaneously. The command capability permits both ADPs to be operating simultaneously. This feature was intended to be used solely for the purpose of performing the initial spin-up maneuver which occurs out of view of any ground station. At all other times, only one ADP should be on at a time, because the telemetry channel for attitude measurements and status data are shared between the two ADPs. These data channels would become meaningless with both units operating simultaneously. Additionally, the interfaces with the star sensor and the DCE would be controlled by both ADPs, producing non-standard operation.

3.3.3.11.2 ADP Attitude Measurements. Attitude measurements made in the ADP alternate between the two command selected measurements. If either of the two measurements is not completed due to that event not occurring, the ADP will "hang up" waiting for this event to occur, i.e., there is no default condition to cycle to the alternate measurement. Telemetry will, however, continually readout the previously measured data. This possibility is likely to occur when transferring to a star reference for the SRR. Typically, the desired star is located within Gate B, and the attitude measurement selected would be "SRR to Gated Star B". When the transfer is made to this star, it will no longer occur in Gate B, causing the attitude measurement to "hang up". After the reference transfer, the gate should be moved or a new attitude measurement selected to allow attitude measurements to continue.

The attitude measurement "SRR to Minor Frame Start" shall only be selected when ACS data is being telemetered at minor frame rate in order to be able to identify the specific minor frame the measurement is referenced to. Furthermore the data is valid only in the first minor frame that is read-out, since that is the frame to which the measurement was made. The same data will

however, be continuously read out until the alternate (on-going) measurement is complete.

The attitude measurement "SRR to Major Frame Start" shall only be selected when ACS data is being telemetered at a subcom rate, so as to provide an unambiguous reference to telemetry time. The measurement can be made in the minor frame mode, but could take an entire major frame to complete. In the meantime, the alternate measurement would be read out each minor frame, thus defeating the purpose of the high sample minor frame rate.

3.3.3.11.3 MAN and NADIR Outputs. The rate control program for both MAN and NADIR must be established so that the pulse will not transition through 360 degrees or cross back through 0 degree once started. A phase error will be incurred if this occurs. Proper operational control should preclude operating beyond a 360-degree range but should it be necessary to do so, the phase reset command can be transmitted at the transition of 0 degrees or 360 degrees to achieve proper phasing.

At the start of each major frame, the current computed angular delay for both MAN and NADIR is loaded into a TM register for read-out during that major frame. (The telemetered value is not the angle delay loaded at BIP occurrence). The most accurate time tag that can be assigned to that telemetered quantity is the start time of the major frame. Using the TM dwell mode (programmable format) will not provide better resolution of the angular increment since only more samples of the same value would be obtained.

3.3.3.11.4 JCE Restrictions. The JCE countdown command (ATCØ4 or ATCØD) or the jet control command (ATCØ2 or ATCØB) should not be sent to the ADP while a desired maneuver is in progress. The commands will be accepted and will configure the JCE accordingly.

If a maneuver appears to have terminated prematurely, possibly due to on-board detection logic, the maneuver should not be restarted until

the contents of the countdown register is verified. If the register had counted down completely, it would have terminated at full count. Restarting the sequence would then initiate firing for maximum duration.

3.3.3.11.5 Programmed Initial Spin-Up Sequence. It is imperative that the commands to turn the ADPs on for the initial spin-up maneuver be programmed through the proper COMs. The ADP No. 1 ON command via COM 5 (ADP19) and ADP No. 2 ON command via COM 6 (ADPB9) are the only ones to be loaded into the stored command sequencer for this maneuver. These commands are inter-locked through the separation switch and are not operative until spacecraft separation has occurred. This prevents any failure or an anomalous event from initiating the stored sequence and firing thrusters while in the launch configuration.

3.3.3.11.6 DCE Thermal Dissipation. The DCE is to be turned on between 8 and 12 hours after liftoff (required for thermal dissipation). The BAPTA drive need not be on. (Reference: 1.5.17)

3.3.3.12 Non-Standard Modes

3.3.3.12.1 Loss of Roll Reference - (Outside Gate A). When the ADP is operated with the PLL loss of lock enable, it will transfer to the simulated SRR reference, "free run mode" if the reference pulse is missing for two consecutive revolutions. If the correct simulated spin period is commanded, the PLL will automatically transfer back to the selected roll reference pulse when the pulse reappears for two consecutive revolutions. If the commanded spin period is not sufficiently accurate, causing Gate A to move more than 5.5 degrees inertially, the reference pulse will be outside Gate A and the transfer will not occur (recognized by ALOCKS and ANISSC, indicating sustained loss of lock and missed SRR counts accumulating). To recover from this condition, two possibilities are presented. The first, which is preferred and the easiest, assumes that the sun is visible or the operation is delayed

until it is visible. The second method assumes that the sun is not visible and a star reference must be used.

(a) Method One (Sun Reference Available): If the sun is visible, a single ADP mode command will allow the ADP to perform a sun acquisition and re-establish lock. The command (ATQ#3 or ATQ#C) would be configured as follows:

PLO:	Loss of lock inhibited selection (X = 0)
SSRX:	X Selected as required for spacecraft attitude.
REF SUN:	To select sun reference.
SUGD:	Disable sun gate for 360 degree sun detection.
SRRADV:	Transfer phase reference to sun when detected.

Restoration of PLL lock can be verified by measuring SMR to RIP as discussed in 3.3.3.1, and observing the decay of the phase transient.

(b) Method Two (Sun Reference Not Available): If the sun is not visible, a star acquisition would have to be performed to obtain a reference. Consideration would first have to be given to the commanded star sensor threshold setting to insure a high probability of detecting a star. If a star reference has been reliably detected previously, and has merely slipped out of Gate A, the threshold setting should be left the same. It could possibly be lowered one setting to enhance the detection capability, provided that there are no interference sources within 22.5 degrees (1/2 Gate width) of the desired star. When star acquisition is commanded, the PLL will lock onto the first detectable star. The command sequence is presented in 3.3.3.3.5. By locating Gate B over Gate A (ATQ#5 or ATQ#E), the star

intensity can be measured to ascertain if this star is the desired reference. If it is not, Gate B can be moved to find the desired star reference. When found, transfer to it may be performed as previously described in 3.3.3.7.4.

3.3.3.12.2 Loss of Roll Reference - (Outside Sensor FOV - Maneuver Induced). The roll reference may be lost during the execution of a precession maneuver if the sun moves out of the sun sensor FOV. If the maneuver was being performed on the midrange sun sensor, the selection of the appropriate extended range sun sensor (ATQ#3 or ATQ#C) will provide a sun reference pulse. If phase reference has been lost, a sun acquisition must be performed as described in Method One, above.

If the FOV of an extended range sun sensor is exceeded as a result of the maneuver (sun angle less than 10 degrees), the SRR would have transferred to the "free run mode". This mode still provides a reference pulse which can be used to perform a maneuver in the reverse direction. Even if the phase of the PLL has changed due to an incorrect simulated spin period, the maneuver should be performed to reacquire the sun. After the attitude has been established, a maneuver to the correct attitude can be performed. To perform a precession maneuver using the free run mode, the ADP mode command (ATQ#3 or ATQ#C) would be configured as follows:

PLO:	Select PLL Inhibit (X = 0) to prevent loss of lock from terminating the maneuver.
SSRX:	X Selected for the required extended range sun sensor.
REF SIM:	Simulated reference selected for maneuver reference. (SUN is not selected since it may introduce PPL transient when sun is detected).

3.3.4 Control Subsystem Command Response. Table 3.3.4-1 presents the command response for the Control Subsystem. The table lists every command that directly affects the subsystem, and the telemtry indication that verifies the proper execution of the command. The mnemonics for the command and telemetry parameters are also included.

When a unit is in the unpowered OFF state, all telemetry from that unit will be reading zero scale (uncalibrated), except temperature telemetry which is powered by DIMS.

TABLE 3.3.4-1. COMMAND RESPONSES FOR CONTROL SUBSYSTEM

CMD	Command Title	Command Description & Subsystem Internal Response	TM Mnemonic	TM Title and/or Remarks
Star Sensor				
STR19 STRA9	PSI* ON	Provides 28 Vdc switched loads bus pwr to the PSI* channel Electronic Control Unit (ECU Regulator) (Figure 3.3.2.3-1) (vertical slit)	A*1ONS	PSI* ON/OFF Status (Bilevel)= Logical "1": indicates an "ON" condition
			A*1THS	PSI* threshold setting indicates 0 Vdc prior to PSI* ON CMD and 0.28 Vdc minimum thereafter. (There is no preferred threshold state due to PSI* ON cmd.)
			ASTART	Star sensor temperature - Rises
			PBUSLI	Spacecraft Loads Current increases 35 milliamperes. Since the minimum resolution for PBUSLI is 72 milliamperes per count, no TM change for PBUSLI is anticipated.
STR1Ø STRAØ	PSI* OFF	Removes 28 Vdc switched loads bus pwr to the PSI* Channel ECU regulator (Figure 3.3.2.3-1)	A*1ONS	PSI* ON/OFF Status (Bilevel)= Logical "0": indicates an "OFF" condition
			A*1THS	PSI* threshold setting indicates 0 Vdc.
			ASTART	Star sensor temperature - decreases.
			PBUSLI	Spacecraft loads current decreases 35 milliamperes. No TM change anticipated.

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Table 3.3.4-1 (continued)

CMD	Command Title	Command Description & Subsystem Internal Response	TM Mnemonic	TM Title and/or Remarks
STR 29 STRB9	PSI2* ON	Provides 28 Vdc switched loads bus power to the PSI2* Channel ECU Regulator (Figure 3.3.2.3-1) (canted slit)	A*2ONS	PSI2* ON/OFF Status (bilevel) = Logical "1": indicates an "ON" condition.
			A*2THS	PSI2* threshold setting indicates a minimum of 0.28 Vdc (there is no preferred threshold state due to PSI2* ON cmd).
			ASTART	Star sensor temperature - rises
			PBUSLI	Spacecraft loads current increases 35 milliamperes. No TM change is anticipated.
STR2Ø STRBØ	PSI2*OFF	Removes 28 Vdc switched loads bus power to the PSI2* channel ECU regulator (Figure 3.3.2.3-1)	A*2ONS	PSI2* ON/OFF status (bilevel) = Logical "0": indicates an "OFF" condition.
			A*2THS	PSI*2 threshold setting indicates 0 Vdc
			ASTART	Star sensor temperature - decreases
			PBUSLI	Spacecraft loads current decreases 35 mA. No TM change is anticipated.

Table 3.3.4-1 (continued)

CMD	Command Title	Command Description & Subsystem Internal Response	TM Mnemonic	TM Title and/or Remarks
STRQ1, STRQA	Star Sensor Threshold and Bandwidth Select (Quantitative Command)			
S2GX	Star 2 threshold = 1-8	"X" determines the threshold setting, 1 through 8, inserted in a register in (Figure 3.3.2.3-2) for detection of stars from -1.0 mag. to +1.2 mag., respectively, for the PSI2* channel. Bits 34, 35, and 36 are the binary states used as follows: 34 35 36 X 0 0 0 1 1 1 0 4 1 1 1 8	A*2THS	PSI*2 threshold setting from 0 to 5 Vdc representing the desired threshold state (X) selected TM (Vdc) State (X) 5.0 1 3.1 2 1.87 3 1.16 4 0.69 5 0.55 6 0.44 7 0.28 8
S1GX	Star 1 threshold = 1-8	"X" determines the threshold setting, 1 through 8, inserted in a register (Figure 3.3.2.3-2) for detection of stars from -1.0 mag to +1.2 mag, respectively, for the PSI* channel. Bits 38, 39, and 40 are the binary states used as follows: 38 39 40 X 0 0 0 1 1 1 0 4 1 1 1 8	A*1THS	PSI* threshold setting from 0 to 5 Vdc, representing the desired threshold state (X) selected. TM (Vdc) State (X) 5.0 1 3.1 2 1.87 3 1.16 4 0.69 5 0.55 6 0.44 7 0.28 8

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Table 3.3.4-1 (continued)

CMD	Command Title	Command Description & Subsystem Internal Response	TM Mnemonic	TM Title and/or Remarks
STRQ1 STRQA	Star Sensor Thresnold and Bandpass Select (Continued)			
S2BWX	Star 2 BP - Hi/Lo	"X" determines bandpass selection inserted in a register (Figure 3.3.2.3-2) for PSI2* channel. CMD Bit 33, Logical "0" indicates 19 Hz bandpass; "1" indicates 2 Hz bandpass. X = H for 19 Hz X = L for 2 Hz	A*2BPS	PSI2* bandpass state status = 1 for 2 Hz selected; = 0 for 19 Hz selected. (19 Hz recommended for all spin rates for nominal mission).
S1BWX	Star 1 BP - Hi/Lo	"X" determines bandpass selection inserted in a register (Figure 3.3.2.3-2) for PSI * channel. Cmd bit 37, Logical "0" indicates 19 Hz bandpass; "1" indicates 2 Hz bandpass X = H for 19 Hz X = L for 2 Hz	A*1BPS	PSI* bandpass state status =1 for 2 Hz selected; =0 for 19 Hz selected. (19 Hz recommended for all spin rates for nominal mission)

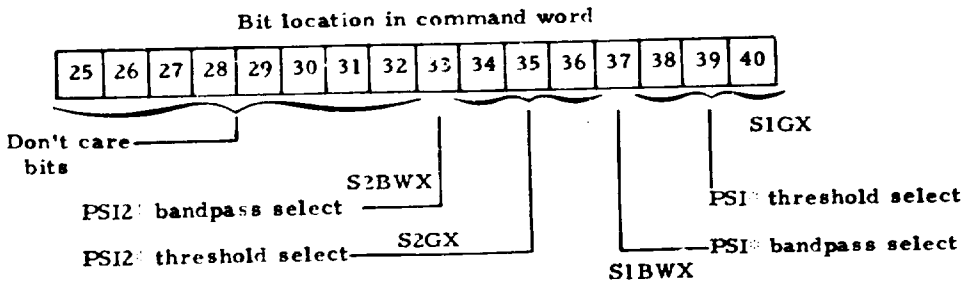
Table 3.3.4-1 (continued)

CMD	Command Title	Command Description & Subsystem Internal Response	TM Mnemonic	TM Title and/or Remarks

Star Sensor Threshold and Bandpass Select Quantitative Command Summary

CMD: STRQ1, S2GX, S1GX, S2BWX, S1BWX (Bits 25 through 40)

STRQA, S2GX, S1GX, S2BWX, S1BWX (Bits 25 through 40)



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Table 3.3.4-1 (continued)

CMD	Command Title	Command Description & Subsystem Internal Response	TM Mnemonic	TM Title and/or Remarks
ATTITUDE DATA PROCESSOR				
ADP19 ADPA9	Attitude Data Processor No. 1 ON	In Figure 3.3.2.3-1, provides spacecraft switched loads bus power to the ECU regulator, thereby activating the onboard ADP electronics except for JCE output buffer and solenoid drivers.	AADP1S	ADP 1 ON/OFF identifies that the ADP 1 is ON when value = 1.
			PBUSLI	Spacecraft loads current increases 250 mA.
ADP1Ø ADPAØ	Attitude Data Processor No. 1 OFF	In Figure 3.3.2.3-1, removes spacecraft switched loads bus power from ADP ECU regulator.	AADP1S	ADP 1 ON/OFF identifies that the ADP 1 is OFF when value = 0.
			PBUSLI	Spacecraft loads current decreases 250 mA.
VAL19 VALA9	Latch Valve No. 1 OPEN	In Figure 3.3.2.3-1, provides propellant to the upstream side of the thruster solenoid assembly for thrusters 3R, 4R, 5A, and 7A. This thruster combination is a minimum set required to perform spin-up, spin-down, velocity changes, and attitude changes.	VALV1S	Latch valve 1 OPEN/CLOSED identifies that latch valve 1 is OPENED when value = 1.
			PBUSLI	Spacecraft loads current increases 620 mA.
VAL1Ø VALAØ	Latch Valve No. 1 CLOSED	In Figure 3.3.2.3-1, terminates propellant from to the upstream side of the thruster solenoid assembly for thrusters 3R, 4R, 5A, and 7A.	VALV1S	Latch valve 1 OPEN/CLOSED identifies latch valve 1 is closed when value = 0.
			PBUSLI	Spacecraft loads current decreases 620 mA.

Table 3.3.4-1 (continued)

CMD	Command Title	Command Description & Subsystem Internal Response	TM Mnemonic	TM Title and/or Remarks
ADP29 ADPB9	Attitude Data Processor No. 2 ON	In Figure 3.3.2.3-1, provides spacecraft switched loads bus power to the ECU regulator thereby activating the onboard ADP electronics except for JCE output buffer and solenoid drivers.	AADP2S	ADP 2 ON/OFF identifies that the ADP 2 is ON when value = 1.
			PBUSLI	Spacecraft loads current increases 250 mA.
ADP2Ø ADPBØ	Attitude Data Processor No. 2 OFF	In Figure 3.3.2.3.1, removes spacecraft switched loads bus power from ADP ECU regulator.	AADP2S	ADP2 ON/OFF identifies that the ADP 2 is OFF when value = 0.
			PBUSLI	Spacecraft loads current decreases 250 mA.
VAL29 VALB9	Latch Valve No. 2 OPEN	In Figure 3.3.2.3-1 provides propellant to the upstream side of the thruster solenoid assembly for thrusters 1R, 2R, and 6A. This thruster combination is minimum set required to perform spin-up, spin-down velocity changes and attitude changes.	VALV2S	Latch valve 2 OPEN/CLOSED identifies that latch valve 2 is opened when value = 1.
			PBUSLI	Spacecraft loads current increases 620 mA.
VAL2Ø VALBØ	Latch Valve No. 2 CLOSED	In Figure 3.3.2.3-1, terminates propellant fromto the upstream side of the thruster solenoid assembly for thrusters 1R, 2R, and 6A.	VALV2S	Latch valve 2 OPEN/CLOSED identifies that latch valve 2 is closed when value = 0.
			PBUSLI	Spacecraft loads current decreases 620 mA.

Table 3.3.4-1 (continued)

CMD	Command Title	Command Description & Subsystem Internal Response	TM Mnemonic	TM Title and/or Remarks
ATQ01 ATQ0A	Measurement Select		ATTM1Z ATTM2Z	Attitude measurement - these two 8-bit groups (ATTM1Z is the 8 LSB's group; ATTM2Z is the 8MSB's group) combine to form the 16 bit word representing either of the two time measurements selected ahead. The range is 0 to 16.38 seconds in 0.25 ms increments. Measurements alternate between the two measurements selected as A and B (or for 1 and 2). However, if the new measurement is not complete, the last measurement will be read out again.
			ATTMSS	Data measurement = when = 1, that measurement selected for MEA1 is telemetered in ATTM1Z and ATTM2Z. When = 0, that measurement selected for MEA2 is telemetered in ATTM1Z and ATTM2Z.
MEA1= SRR	Measurement A select: SRR-SRR	Start and stop of 16 bit counter selected to be successive occurrences of SRR. The spin period measurement is generated with the selected roll reference (SRR) being either the ψ, ψ^*, $\psi 2^*$ or simulated SRR as the primary input to the phase lock loop (PLL)	AM1ADS	Measurement A address = binary 000: "SRR to SRR" spacecraft spin period measurement selected.

Table 3.3.4-1 (continued)

CMD	Command Title	Command Description & Subsystem Internal Response	TM Mnemonic	TM Title and/or Remarks
ATQ01 ATQ0A	Measurement Select (Continued)			
MEA1= MFS	Measurement A Select: "SRR - Minor F Rate"	Start and stop of 16 bit counter selected to be occurrences of SRR and start of telemetry minor frame, respectively.	AM1ADS	Measurement A address = binary 100: "SRR - Minor Frame Sync" selected. Time between occurrence of SRR and start of telemetry minor frame. Used with the minor frame mode TM format. Used for inertial determination of the SRR with respect to any and all telemetry bit stream word locations.
MEA1= RIP	Measurement A Select: "SRR - RIP"	Start and stop of 16 bit counter selected to be occurrences of SRR and RIP, respectively.	AM1ADS	Measurement A address = binary 010: "SRR - RIP" selected. Time between occurrence of SRR and occurrence of roll index pulse, (used for spin angle (azimuth) determination of the roll index pulse with respect to the SRR).
MEA1= MIP	Measurement A Select: "SRR - MIP"	Start and stop of 16 bit counter selected to be occurrences of SRR and MIP, respectively.	AM1ADS	Measurement A address = 110: "SRR - MIP" selected. Time between occurrence of SRR and occurrence of master index pulse. (Used for spin angle determination of the master index pulse with respect to the SRR).

Table 3.3.4-1 (continued)

CMD	Command Title	Command Description & Subsystem Internal Response	TM Mnemonic	TM Title and/or Remarks
ATQ01 ATQ0A	Measurement Select (Continued)			
MEA1=GSB (Gated Star B)	Measurement A Select: "SRR - Gated Star B"	Start and stop of 16 bit counter selected to be occurrences of SRR and gated star B, respectively.	AM1ADS	Measurement A address = 001: "SRR - Gated Star B" selected. Time between occurrence of SRR and appearance of a star above the commanded threshold setting within ±5.625° of the center of Gate B. (Used for spin angle determination of the first gated star B with respect to the SRR).
MEA1=S2S (PSI2)	Measurement A Select: "SRR - PSI2"	Start and stop of 16 bit counter selected to be occurrences of SRR and appearance of sun in PSI2 window, respectively.	AM1ADS	Measurement A address = 101: "SRR - PSI2" selected. Time between occurrence of SRR and appearance of sun in PSI2 sensor window. Depending on the selected roll ref., the SRR - PSI2 measurement will represent the following: SRR → Measurement Ψ → Sun aspect angle Ψ^* → Sun - star separation Ψ_2^* → Sun - star separation Sim SRR → Free run mode drift rate and/or sim spin period verification with actual rotor spin rate.

Table 3.3.4-1 (continued)

CMD	Command Title	Command Description & Subsystem Internal Response	TM Mnemonic	TM Title and/or Remarks
ATQØ1 ATQØA MEA1= MAF	Measurement Select (Continued) Measurement A Select: "SRR - Major F Rate"	Start and stop of 16 bit counter selected to be occurrence of SRR and start of telemetry major frame, respectively.	AM1ADS	Measurement A address = 011: "SRR - Major Frame Sync" selected. Time between occurrence of SRR and occurrence of start of telemetry major frame. Used primarily with the subcom mode format. Used for inertial determination (azimuth) of the SRR with respect to any and all telemetry bit stream word locations.
MEA2= SRR	Measurement B Select: SRR-SRR	Start and stop of 16 bit counter selected to be successive occurrences of SRR. The spin period measurement is generated with the selected roll reference (SRR) being either the ψ, ψ^*, ψ_2 or simulated SRR as the primary input to the phase lock loop (PLL)	AM2ADS	Measurement B address = binary 000: "SRR to SRR" spacecraft spin period measurement selected.
MEA2= MFS	Measurement B Select: "SRR - Minor F Rate"	Start and stop of 16 bit counter selected to be occurrences of SRR and start of telemetry minor frame, respectively.	AM2ADS	Measurement B address = binary 100: "SRR - Minor Frame Sync" selected. Time between occurrence of SRR and start of telemetry minor frame. Used with the minor frame mode TM format. Used for inertial determination of the SRR with respect to any and all telemetry bit stream word locations.)

Table 3.3.4-1 (continued)

CMD	Command Title	Command Description & Subsystem Internal Response	TM Mnemonic	TM Title and/or Remarks
ATQ01 ATQ0A	Measurement Select (Continued)			
MEA2= RIP	Measurement B Select: "SRR - RIP"	Start and stop of 16 bit counter selected to be occurrences of SRR and RIP, respectively.	AM2ADS	Measurement B address = binary 010: "SRR - RIP" selected. Time between occurrence of SRR and occurrence of roll index pulse (used for spin angle (azimuth) determination of the roll index pulse with respect to the SRR)
MEA2= MIP	Measurement B Select: "SRR - MIP"	Start and stop of 16 bit counter selected to be occurrences of SRR and MIP, respectively.	AM2ADS	Measurement B address = 110: "SRR - MIP" selected. Time between occurrence of SRR and occurrence of master index pulse. (Used for spin angle determination of the master index pulse with respect to the SRR).
MEA2= GSB (Gated Star B)	Measurement B Select: "SRR - Gated Star B"	Start and stop of 16 bit counter selected to be occurrences of SRR and gated star B, respectively.	AM2ADS	Measurement B address = 001: "SRR - Gated Star B" selected. Time between occurrence of SRR and appearance of a star above the commanded threshold setting within $\pm 5.625^{\circ}$ of the center of gate B. (Used for spin angle determination of the first gated star B with respect to the SRR).

Table 3.3.4-1 (continued)

CMD	Command Title	Command Description & Subsystem Internal Response	TM Mnemonic	TM Title and/or Remarks
ATQ01 ATQ0A	Measurement Select (Continued)			Measurement B address = 101: "SRR - PSI2" selected. Time between occurrence of SRR and appearance of sun in PSI2 sensor window. Depending on the selected roll ref., the SRR - PSI2 measurement will represent the following: SRR — Measurement ψ — Sun aspect angle ψ^* — Sun-star separation $\psi_2{}^*$ — Sun-star separation Sim SRR — Free run mode drift rate and/or sim spin period verification with actual rotor spin rate.
MEA2= S2S (PSI2)	Measurement B Select: SRR - PSI2	Start and stop of 16 bit counter selected to be occurrences of SRR and appearance of sun in PSI2 window, respectively.	AM2ADS	
MEA2= MAF	Measurement B Select: "SRR - Major F Rate"	Start and stop of 16 bit counter selected to be occurrence of SRR and start of telemetry major frame, respectively.	AM2ADS	Measurement B address = 011: "SRR - Major Frame Sync" selected. Time between occurrence of SRR and occurrence of start of telemetry major frame. Used primarily with the sub com mode format. Used for inertial determination (azimuth) of the SRR with respect to any and all telemetry bit stream word locations.)

Table 3.3.4-1 (continued)

CMD	Command Title	Command Description & Subsystem Internal Response	TM Mnemonic	TM Title and/or Remarks

ADP Measurement Select Quantitative Command Summary

CMD: ATQ01, MEA1 = XXX, MEA2 = XXX (ADP No. 1 Only)
ATQ0A, MEA1 = XXX, MEA2 = XXX (ADP No. 2 Only)

Where XXX = SRR or MFS or RIP or MIP or GSB or S2S or MAF

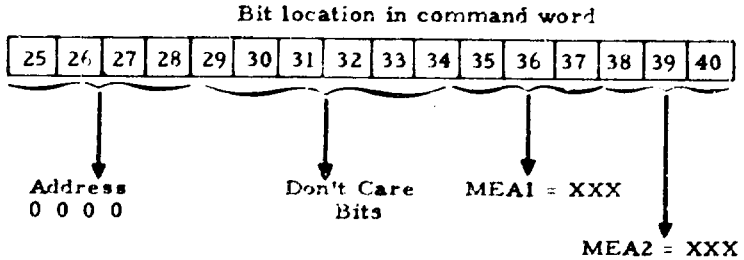


Table 3.3.4-1 (continued)

CMD	Command Title	Command Description & Subsystem Internal Response	TM Mnemonic	TM Title and/or Remarks
ATQ02 ATQ0B	Jet Control			
R1X	Jet Select - Jet 1R	X determines the Enabled/Disabled state for radial jet 1R X = E for ENABLE; D for Disable. In Figure 3.3.2.3-1, the 1R driver amplifier is enabled or disabled	AJET1S	Jet select bits: Bit 1 (radial jet 1R status) = binary 1 for enabled, = 0 for disabled.
R2X	Jet Select - Jet 2R	X determines the Enabled/Disabled state for radial jet 2R. X = E for ENABLE; X = D for Disable. In Figure 3.3.2.3-1, the 2R driver amplifier is enabled or disabled.	AJET2S	Jet select bits: Bit 2 (radial jet 2R status) = binary 1 for enabled, = 0 for disabled.
R3X	Jet Select - Jet 3R	X determines the Enabled/Disabled state for radial jet 3R. X = E for ENABLE; X = D for DISABLE. Jet select in Figure 3.3.2.3-1, the 3R driver amplifier is enabled or disabled.	AJET3S	Jet select bits: Bit 3 (radial jet 3R status) = binary 1 for enabled, = 0 for disabled.
R4X	Jet Select - Jet 4R	X determines the Enabled/Disabled state for radial jet 4R. X = E for ENABLE; X = D for DISABLE. In Figure 3.3.2.3-1, the 4R driver amplifier is enabled or disabled.	AJET4S	Jet select bits: Bit 4 (radial jet 4R) status) = binary 1 for enabled, = 0 for disabled.
A5X	Jet Select - Jet 5A	X determines the Enabled/Disabled state for axial jet 5A. X = E for ENABLE; X = D for DISABLE. In Figure 3.3.2.3-1, the 5A driver amplifier is enabled or disabled.	AJET5S	Jet select bits: Bit 5 (axial jet 5A status) = binary 1 for enabled, = 0 for disabled.

Table 3.3.4-1 (continued)

CMD	Command Title	Command Description & Subsystem Internal Response	TM Mnemonic	TM Title and/or Remarks
ATQ02 ATQ0B	Jet Control (Continued)			
A6X	Jet Select - Jet 6A	X determines the Enabled/Disabled state for axial jet 6A. X = E for ENABLE; X = D for DISABLE. In Figure 3.3.2.3-1, the 6A driver amplifier is enabled or disabled.	AJET6S	Jet select bits: Bit 6 (axial jet 6A status) = binary 1 for enabled, = 0 for disabled.
A7X	Jet Select - Jet 7A	X determines the Enabled/Disabled state for axial jet 7A. X = E for ENABLE; X = D. In Figure 3.3.2.3-1, the 7A driver amplifier is enabled or disabled.	AJET7S	Jet select bits: Bit 7 (axial jet 7A status) = binary 1 for enabled, = 0 for disabled.
FMX	Normal/Alternate Fire Mode	X determines selection of normal/alternate fire mode for thruster firing. Normal mode provides one thruster pulse per spacecraft spin revolution. Alternate mode provides two thruster pulse firings, separated by 180°, per spacecraft spin revolution. X = N for Normal; X = A for Alternate. Enable signal is provided to 0° AND gate only OR to (0° and 180°) AND gate in Figure 3.3.2.3-1.	AJETMS	Normal/alternate fire mode = 1 for normal select; = 0 for alternate select.
P/X	Pulse Width Select	Two selectable pulse widths, 128 ms or 512 ms, determined by X selection as follows: X — Pulse Duration 2 — 512 ms = 1/2 sec 8 — 128 ms = 1/8 sec Enable signal is provided to 128 ms AND gate OR to 512 ms AND gate in Figure 3.3.2.3-1.	APULLS	Pulse width select 512/128 ms = 1 for 512 ms selected; = 0 for 128 ms selected.

Table 3.3.4-1 (continued)

CMD	Command Title	Command Description & Subsystem Internal Response	TM Mnemonic	TM Title and/or Remarks
ATQ02 ATQ0B	Jet Control (Continued)			
CX	Pulse/Time Count Select	The firing duration is controlled by either counting time or the number of thruster pulse firings. This selection is independent of pulse/continuous firing mode selection, i.e., pulse Firing mode and time count can both be selected. X = P for pulse count selected; X = T for time count selected. Enable signal is provided to pulse count AND gate OR time count AND gate in Figure 3.3.2.3-1.	AMAGCS	Pulse/time count select = 1 for pulse count selected; = 0 for time count selected.
FX	Continuous/Pulse Fire Select	The thruster is fired either continuously or as a pulse. Pulse firing selection is further predicated on normal/alternate mode selection and pulse duration selection. X = T for continuous fire select; X = P for pulse fire select. Enable signal is provided to PULSE FIRE AND gate OR to a TIME FIRE AND gate in Figure 3.3.2.3-1.	AJETMS	Continuous/pulse fire select = 1 for continuous selected; = 0 for pulse selected.
SRDX	Spin Rate Detector Enable/Inhibit	Provides automatic thruster turn-off, if enabled, upon detection of spin speeds less than or equal to 3.7 rpm, and greater than or equal to 67 rpm. X = E for ENABLE selected; X = D for INHIBIT selected. Enable signal is or is not provided to spin rate detector in Figure 3.3.2.3-1.	ASDETS	Spin rate detector Enable/Inhibit = 1 for Enabled; = 0 for Inhibited.

Table 3.3.4-1 (continued)

CMD	Command Title	Command Description & Subsystem Internal Response	TM Mnemonic	TM Title and/or Remarks
ADP JET CONTROL QUANTITATIVE COMMAND SUMMARY CMD: ATQ∅2, R1X, R2X, R3X, R4X, R5X, R6X, R7X, FMX, P/X, CX, FX, SRDX (ADP No. 1 Only) ATQ∅B, R1X, R2X, R3X, R4X, R5X, R6X, R7X, FMX, P/X, CX, FX, SRDX (ADP No. 2 Only)				
ATQ∅3 ATQ∅C	ADP Mode Select			
SRXX	PLL Spin Range - Select	Provides for wide PLL dynamic range in four discrete steps determined by XX selection as follows: XX — Range Selected 04 — 1) 4-9 rpm 08 — 2) 8-18 rpm 16 — 3) 16-35 rpm 32 — 4) 32-71 rpm Selected range must encompass actual spacecraft rotor rate for proper acquisition and tracking. One of four countdown durations is selected in a variable countdown circuit in Figure 3.3.2.3-1.	ASPINS	PLL spin range select: identifies selected spin range as follows: State of TM word 5, bits 0 1 — Range (rpm) 0 0 — 4-9 1 0 — 8-18 0 1 — 16-35 1 1 — 32-71

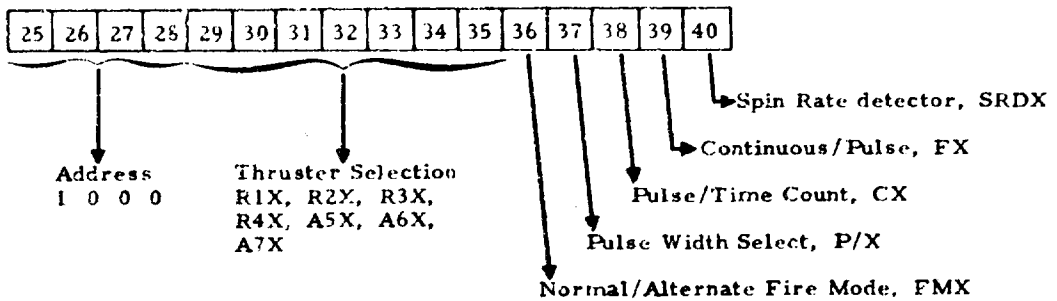


Table 3.3.4-1 (continued)

CMD	Command Title	Command Description & Subsystem Internal Response	TM Mnemonic	TM Title and/or Remarks
ATQ03 ATQ0C	ADP Mode Select (Continued)			
PLX	PLL Loss of Lock Enable/Inhibit	In Figure 3.3.2.3-1, selecting the PLL loss of lock enabled provides for automatic transfer of SRR to simulated SRR and JCE turn-off, upon detection of two consecutive missing SRR pulses. Enabled, the PLL loss of lock prevents the PLL frequency from being updated if the SRR pulse is outside gate A. Inhibited, the loss of lock detection circuit is forced to an in-lock condition independent of whether or not the SRR pulse is within gate A. If the SRR pulse occurs within gate A the PLL will maintain frequency and phase to keep the SRR pulse in the center of gate A. If a pulse occurs outside gate A when in the gated mode (star or gated sun mode) the loop will not be corrected and will maintain phase and frequency to within the drift characteristics of the PLL circuitry. If a pulse occurs outside gate A in the ungated sun mode the loop performs as in the sun acquisition mode and will slew in frequency and phase to lock onto detected pulse. A count of the number of missing SRR pulses is recorded. X = E for enabled; X = D for inhibited.	ALOLES	PLL loss of lock enable/ inhibit = 1 if enabled; = 0 if inhibited.
			ALOCKS	PLL loss of lock: With PLL loss of lock enabled, bit will = 1 if SRR is the PLL input; bit will = 0 if input is transferred to SIM SRR. With PLL loss of lock disabled, bit will always be = 1.

Table 3.3.4-1 (continued)

CMD	Command Title	Command Description & Subsystem Internal Response	TM Mnemonic	TM Title and/or Remarks
ATQ03 ATQ0C	ADP Mode Select) (Continued)			
SGAX	Star Gate A Channel Select	In Figure 3.3.2.3-1, selects either the ψ^* channel or υ_2^* channel output as the Gated A star output. If star acquisition CMD (STX) and SRR advance CMD (SRAX) are also commanded then gate A is centered over first star in the spin revolution following receipt of star advance CMD that is above the selected threshold setting. Gate A is 45° until star normal is commanded, when it becomes 11.25°. X = 1 for υ^* channel selected; = 2 for υ_2^* channel selected.	A*GASS	Star Gate A; channel 1/2 select = 1 if υ^* channel is selected; = 0 if υ_2^* channel is selected.
SGBX	Star Gate B Channel Select	In Figure 3.3.2.3-1, selects either the ψ^* channel for υ_2^* channel as the gated B star output for attitude measurements. X = 1 for ψ^* channel select. = 2 for υ_2^* channel selected.	A*GBSS	Star Gate B; channel 1/2 select = 1 if ψ^* channel is selected; = 0 if υ_2^* channel is selected
SSRX	Sun Sensor (Range) Select	In Figure 3.3.2.3-1, provides for selection of one of the three sun sensor viewing ranges as follows: Viewing Range / X / State of CMD Bits 34, 35 Mid / M / 00 Upper / U / 10 Lower / L / 01	ASUNSS	Sun Sensor Select - State of TM Word 4, Bits 3 & 4 / Indicates Selected State 00 / Mid 10 / Upper 01 / Lower

Table 3.3.4-1 (continued)

CMD	Command Title	Command Description & Subsystem Internal Response	TM Mnemonic	TM Title and/or Remarks
ATQ$3 ATQ$C REFXXX	ADP Mode Select (Continued) Sun/Star/Simulated Select	In Figure 3.3.2.3-1, provides for selection of one of three roll references (sun, star, or simulated SRR) as the primary input to PLL. XXX = SUN OR = STR (STAR) OR = SIM (SIMULATED)	ASRRMS	Identifies reference selection as follows: Ref — State of TM Word 4, Bits 5 and 6 Sun: 1 1 Star: 0 1 Simulated SRR (1 or 0): 0
STX	Star Acquisition/ Normal	In Figure 3.3.2.3-1, provides for PLL frequency and phase locking to any star in the spin revolution that is capable of being detected by the star sensor threshold setting. The normal (11.25°) gate A is expanded to 45° when star acquisition is selected, to allow for the frequency difference between PLL and the actual rotor rate to be slightly in excess of ±1.5%. This CMD is used in conjunction with the SRR advance CMD (SRAX) X = A for acquisition selected; = N for normal selected.	A*ACQS	Star acquisition/normal = 1 if star acquisition is selected; = 0 if normal is selected
SUGX	Sun Gate Enable/Disable	In Figure 3.3.2.3-1, selecting the enabled state provides a "gate A" to be centered over the sun SRR input. This allows the detection of the number of missing sun pulses, as well as automatic switching to simulated SRR upon detection of two consecutive missing sun SRRs. If Disabled state is selected, the	ASUNGS	Enable/Disable Sun Gate = 1 if enabled; = 0 if disabled.

Table 3.3.4-1 (continued)

CMD	Command Title	Command Description & Subsystem Internal Response	TM Mnemonic	TM Title and/or Remarks
ATQ03 ATQ0C SUGX (cont)	ADP Mode Select (Continued)	PLL is forced to frequency search and phase lock to the sun at all times. x = E for Enable selected; = D for Disable selected.		
SRAXXX	SRR Advance	In Figure 3.3.2.3-1, allows PLL reference switching from sun to star, or from star to star, or from star to sun or from simulated SRR to sun; and, in rare cases when switching from simulated SRR to star (called star acquisition when sun is not available.). XXX = ADV if advance state is selected; = NRM, if normal state is selected.	ADVANS	SRR Advance Status = 1 if Advance state is selected = 0 if normal state is selected

Table 3.3.4-1 (continued) ADP MODE SELECT QUANTITATIVE COMMAND SUMMARY

CMD	Command Title	Command Description & Subsystem Internal Response	TM Mnemonic	TM Title and/or Remarks

ADP MODE SELECT QUANTITATIVE COMMAND SUMMARY

CMD: ATQ03, SRXX, PLX, SGAX, SGBX, SSRX, REFXXX, STX, SUGX, SRAXXX. (ADP No. 1 Only)

ATQ0C, SRXX, PLX, SGAX, SGBX, SSRX, REFXXX, STX, SUGX, SRAXXX. (ADP No. 2 Only)

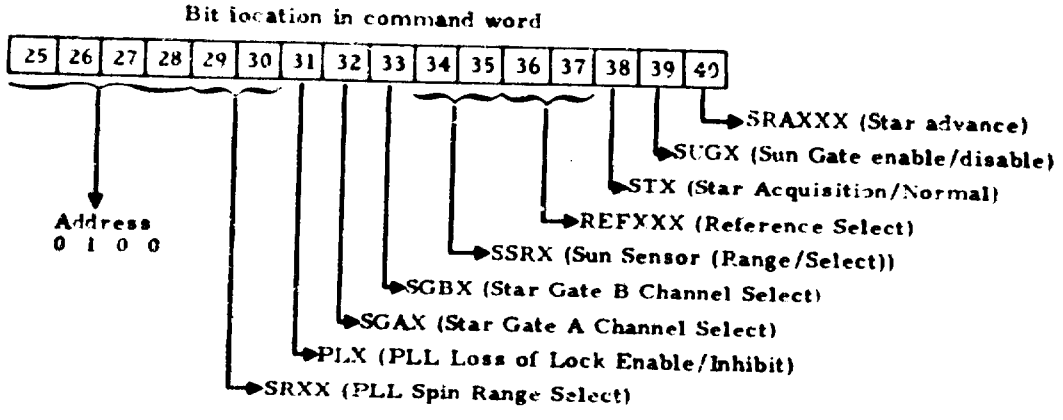


Table 3.3.4-1 (continued)

CMD	Command Title	Command Description & Subsystem Internal Response	TM Mnemonic	TM Title and/or Remarks
ATQ04 ATQ0D	Jet Countdown	In Figure 3.3.2.3-1, provides firing duration of selected thrusters in terms of time or number of pulse firings as follows (12 bits): 0 - 2096.64 sec in 0.512 sec steps or 0 - 4095 pulses.	AJMAGC	JCE Countdown (12 bits TM word) - identifies the Jet Countdown Command Register contents at all times. (Stops at maximum count of all "1"s)
ATQ05 ATQ0E	ACS Angle Delay Magnitude	In Figure 3.3.2.3-1, provides a phase angle delay with respect to the SRR for gated Star B control and Jet Fire Angle Control. Angle delay capability from 0.352° to 360.0° in increments of 0.3515625° is provided. Actual ACS Angle Delay is 0.352° or 1 count larger than the commanded value.	AACSAD	ACS Angle Delay Magnitude (10 bits) TM word) - identifies the ACS Angle Delay countdown register contents at all times. (Same range and increment as for command.)
ATQ06 ATQ0F	Roll Index Delay Magnitude	In Figure 3.3.2.3-1, provides a phase angle delay with respect to the SRR for Science and the DCE. The DCE utilizes the RIP to generate the variable phase delays, phase delayed from the RIP, for RAM and NADIR Science Signal. Actual Delay equals the number of Magnitude Counts Commanded. Spin Angle delay capability of 0 to 359.65° in 0.351° steps is provided.	ARIPAD	Roll Index Delay Magnitude (10 bits TM word) - identifies the selected RIP Phase Angle delay. (Same range and increment as for command.)
ATQ07 ATQ0G	PLL Spin Period, Magnitude (MSPs)	In Figure 3.3.2.3-1, controls the eight most significant bits of a 16 bit counter that represents the onboard free-run facsimile of actual spacecraft rotor rate. Commandable period is 0 to 16.32 sec in increments of 0.064 sec.	ASIMSZ	PLL Spin Period Magnitude (8 MSBs): Identifies the 8 MSBs of the Selected Spin Period Control.

Table 3.3.4-1 (continued)

CMD	Command Title	Command Description & Subsystem Internal Response	TM Mnemonic	TM Title and/or Remarks
ATQ08 ATQ0H	PLL Spin Period, Magnitude (LSBs)	In Figure 3.3.2.3-1, controls the eight least significant bits of a 16 bit counter that represents the onboard free run facsimile of the actual spacecraft rotor rate. Commandable period is 0 to 0.06375 sec in 0.25 ms increments.	ASIMSZ	PLL Spin Period Magnitude (8 LSBs): identifies the 8 LSBs of the selected Spin Period Control.
ATQ09 ATQ0I	JCE Buffer Output Enable	In Figure 3.3.2.3-1, provides +15V to the JCE buffers that drive the solenoid drivers. JCE buffer enable is a necessary condition for any thruster firing. Receipt of the JET fire interlock and JET fire CMD in that order, without an intervening JCE buffer disable CMD, is the second necessary condition for any thruster firing.	AJCE1S	Jet Control Electronics 1 Buffer Output State
			AJCE2S	Jet Control Electronics 2 Buffer Output State
				Each above identifies (1=ON) JCE output power is "ON."
ATQ10 ATQ0J	JCE Power Buffer Output Disable	In Figure 3.3.2.3-1, removes +15V from the JCE buffers that drive the solenoid driver. JCE buffer disable or more appropriately called JCE "OFF", prevents any thruster firing after CMD receipt. This command also resets the JET fire interlock latch and the JET fire enable latch.	AJCE1S	Jet Control Electronics 1 Buffer Output State
			AJCE2S	Jet Control Electronics 2 Buffer Output State
				Each above identifies (0=OFF) JCE output power is "OFF"
			AJCEFS	Jet Fire enable status = Disabled (= 0).

Table 3.3.4-1 (continued)

CMD	Command Title	Command Description & Subsystem Internal Response	TM Mnemonic	TM Title and/or Remarks
ATQ11 ATQ0K	Jet Fire Interlock	In Figure 3.3.2.3-1, sets the Jet Fire interlock latch. This interlock logic does not allow the jets to fire until both the jet fire interlock CMD and the jet fire CMD have been received in that order without an intervening JCE Buffer Disable CMD. There is no time lapse requirement between the jet fire interlock CMD and the jet fire CMD.		There is no TM that directly indicates the state of the Jet Fire Interlock.
ATQ12 ATQ0L	Jet Fire	In Figure 3.3.2.3-1, sets the jet fire enable latch. In the continuous fire mode, receipt of the jet fire CMD (assuming interlock sequence is complete) starts thruster firing. In the pulse mode, the leading edge of the first ACS delay pulse after receipt of jet fire CMD starts the thruster firing.	AJCEFS	Jet fire enable status: Identifies thruster firing sequence is activated when value = 1.
				Each thruster valve requires a maximum of 21 watts to open. This power consumption may be detected via PLIMTI (bus voltage limiter current) when the bus limiters are dissipating excess spacecraft power; or may be detected via PBUSLI (Spacecraft Loads Current).
ADP ATQ04 THROUGH ATQ12 QUANTITATIVE COMMAND SUMMARY CMD: ATQ04 (ADP No. 1 Only) ATQ0D (ADP No. 2 Only) Bit location in command word 25 26 27 28 29 30 31 32 33 34 35 36 37 38 39 40 Address: 1 1 0 0 Magnitude Information: Jet Countdown				

Table 3.3.4-1 (continued)

CMD	Command Title	Command Description & Subsystem Internal Response	TM Mnemonic	TM Title and/or Remarks

ADP ATQ04 THROUGH ATQ12 QUANTITATIVE COMMAND SUMMARY

CMD: ATQ05 (ADP No. 1 Only)
ATQ0E (ADP No. 2 Only)

Bit location in command word

25	26	27	28	29	30	31	32	33	34	35	36	37	38	39	40
Address 0 1 0 0				Magnitude Information ACS Delay										Don't Care	

CMD: ATQ06 (ADP No. 1 Only)
ATQ0F (ADP No. 2 Only)

Bit location in command word

25	26	27	28	29	30	31	32	33	34	35	36	37	38	39	40
Address 1010				Magnitude Information Roll Index Delay										Don't Care	

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Table 3.3.4-1 (continued)

CMD	Command Title	Command Description & Subsystem Internal Response	TM Mnemonic	TM Title and/or Remarks

ADP ATQ04 THROUGH ATQ12 QUANTITATIVE COMMAND SUMMARY

CMD: ATQ07 (ADP No. 1 Only)
ATQ0G (ADP No. 2 Only)

Bit location in command word

25	26	27	28	29	30	31	32	33	34	35	36	37	38	39	40

Address 0 1 1 0 — Magnitude Information PLL Spin Period, MSBs — Don't Care

CMD: ATQ08 (ADP No. 1 Only)
ATQ0H (ADP No. 2 Only)

Bit location in command word

25	26	27	28	29	30	31	32	33	34	35	36	37	38	39	40

Address 1 1 1 0 — Magnitude Information PLS Spin Period, LSBs — Don't Care

CMD: ATQ09 (ADP No. 1 Only)
ATQ0I (ADP No. 2 Only)

Bit location in command word

25	26	27	28	29	30	31	32	33	34	35	36	37	38	39	40

Address, JCE Buffer Enable 0 0 0 1 — Don't Care

Table 3.3.4-1 (continued)

CMD	Command Title	Command Description & Subsystem Internal Response	TM Mnemonic	TM Title and/or Remarks

ADP ATQ04 THROUGH ATQ12 QUANTITATIVE COMMAND SUMMARY

CMD: ATQ10 (ADP No. 1 Only)
ATQ0J (ADP No. 2 Only)

Bit location in command word

25	26	27	28	29	30	31	32	33	34	35	36	37	38	39	40
	Address, JCE Buffer Disable 1 0 0 1									Don't Care					

CMD: ATQ11 (ADP No. 1 Only)
ATQ0K (ADP No. 2 Only)

Bit location in command word

25	26	27	28	29	30	31	32	33	34	35	36	37	38	39	40
	Address, JCE Fire Interlock 0 1 0 1									Don't Care					

CMD: ATQ12 (ADP No. 1 Only)
ATQ0L (ADP No. 2 Only)

Bit location in command word

25	26	27	28	29	30	31	32	33	34	35	36	37	38	39	40
	Address, Jet Fire 1 1 0 1									Don't Care					

Table 3.3.4-1 (continued)

CMD	Command Title	Command Description & Subsystem Internal Response	TM Mnemonic	TM Title and/or Remarks
Despin Control Electronics				
DCE19 DCEA9	Despin Control Electronics 1 ON/2 OFF	In Figure 3.3.2.4-1, provides spacecraft switched loads bus power to the DCE ECU regulators and toggles unit 2 off. DCE will initialize in a random manner. This command should be followed by DCE configure quantitative command before sending either MOTOR DRIVE ON command.	ADCE1V	DCE 1 power supply voltage - rises from 0 V to a nominal +5 V.
			ADCE2V	DCE 2 power supply voltage - reads 0 V.
			ADCE1T	DCE 1 temperature - rises and settles at ___°F.
			PBUSLI	Spacecraft load current increases 275 milliamps if both DCE units were off prior to this command.
DCE29 DCEB9	Despin Control Electronics 2 ON/1 OFF	In Figure 3.3.2.4-1, provides spacecraft switched loads bus power to the DCE 2 ECU regulators and toggles unit 1 OFF. DCE will initialize in a random manner. This command should be followed by DCE. Configure quantitative command before sending either MOTOR DRIVE ON command.	ADCE1V	DCE 1 power supply voltage - reads 0 V.
			ADCE2V	DCE 2 power supply voltage - rises from 0 V to nominal +5 V.
			ADCE2T	DCE 2 temperature - rises and settles at ___°F.
			PBUSLI	Spacecraft load current increases 275 milliamps if both DCE units were off prior to this command.
DCE10 DCEA0	Despin Control Electronics 1 OFF/2 OFF	In Figure 3.3.2.4-1, removes spacecraft switched loads bus power from both DCE ECU regulators.	ADCE1V	DCE 1 power supply voltages - reads 0 V.
			ADCE2V	DCE 2 power supply voltage - reads 0 V.

Table 3.3.4-1 (continued)

CMD	Command Title	Command Description & Subsystem Internal Response	TM Mnemonic	TM Title and/or Remarks
DCE1Ø DCEAØ (cont)			ADCE1T	DCE 1 temperature - decreases if DCE 1 had been on.
			ADCE2T	DCE 2 temperature - decreases if DCE 2 had been on.
			PBUSLI	Spacecraft loads current decreases as minimum of 275 mA.
MTR19 MTRA9	Motor Drive 1 ON	In Figure 3.3.2.4-1, provides fused 28 Vdc (fuse source is 28 V switched loads bus). Drive power to the BAPTA motor based upon torque command inputs generated within DCE 1. Antenna will despin and assume a position in azimuth and elevation based upon what had been commanded in the DCE configure conmand or what had been generated randomly upon DCE 1 turnon or DCE 2 turnon.	AMTR1S	Despin motor drive 1 ON/ off = 1 - identifies Motor 1 ON.
			AMTR2S	Despin motor drive 2 ON/ OFF = 0 - identifies Motor 2 OFF.
			PBUSLI	Spacecraft load current increases 150 milliamps if both motor drivers were OFF.
			ARATEQ	Rate torque - assumes a nominal 2.75 V (precalibrated) with no torque commands present.
			APOSTQ	Position torque - assumes a nominal 2.5 V (precalibrated) with no torque commands present.

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Table 3. 3. 4-1 (continued)

CMD	Command Title	Command Description & Subsystem Internal Response	TM Mnemonic	TM Title and/or Remarks
MTR19 MTRA9 (cont)			AMOTRI	Despin motor drive current - assumes a nominal 0 amps with no torque commands present.
MTR29 MTRB9	Motor Drive 2 ON	In Figure 3. 3. 2. 4-1, provides fused 28 Vdc (fused source is 28 V switched loads bus). Drive power to the BAPTA motor based upon torque command inputs generated within DCE 2. Antenna will despin and assume a position in azimuth and elevation based upon what had been commanded in the DCE configure command or what had been generated randomly upon DCE 1 turnon or DCE 2 turnon.	AMTR1S	Despin motor drive 1 ON/ OFF = 0, identifies unit 1 OFF.
			AMTR2S	Despin motor drive 2 ON/ OFF = 1, identifies unit 2 ON.
			PBUSLI	Spacecraft load current increases 150 milliamps if both motor drivers were OFF.
			ARATEQ	Rate torque - assumes a nominal 2. 75 V (precalibrated) with no torque commands present.
			APOSTQ	Rate torque - assumes a nominal 2. 5 V (precalibrated) with no torque commands present.
			AMOTRI	Despin motor drive current - assumes a nominal 0 amp with no torque commands present.

Table 3.3.4-1 (continued)

CMD	Command Title	Command Description & Subsystem Internal Response	TM Mnemonic	TM Title and/or Remarks
MTR1Ø MTRAØ	Motor Drivers 1 and 2 OFF	In Figure 3.3.2.4-1, removes fused 28 Vdc (fused source is 28 V switched loads bus) - drive power to the BAPTA motor.	AMTR1S	Despin motor drive 1 ON/ OFF = 0 - identifies unit 1 OFF.
			AMTR2S	Despin motor drive 2 ON/ OFF = 0 - identifies unit 2 OFF.
			PBUSLI	Spacecraft loads current decreases 150 milliamps if one motor drive was ON.
DCEQ1 DCEQA	Azimuth Pointing Slew	In Figure 3.3.2.4-1, provides for slewing the despun antenna from 0° to 164.78° per minute in increments of 0.04024° per minute steps (13 bits resolution).	ATTM1Z ATTM2Z	Time interval measurement of PLL to MIP (master index pulse) provides despun antenna position profile versus GMT time (16 bits resolution).
DCEQ2 DCEQB	RAM Variable Phase Delay	In Figure 3.3.2.4-1, provides for additional spin angle delay of roll index pulse (RIP). Magnitude of delay is the integral of the commanded RAM variable phase delay rate. Capability to vary the rate from 0° to 164.78° per minute in increments of 0.04024° per minute steps is provided (13 bits resolution).	AVPDRD	RAM variable phase delay - identifies magnitude (position) of the resultant spin angle delay of RAM VPD with respect to RIP. This is a pulse that is routed once per spin period to selected scientific instruments (OUVS, ONMS, and ORPS) as the RAM pulse (12 bits resolution).
DCEQ3 DCEQC	NADIR Variable Phase Delay	In Figure 3.3.2.4-1, provides for additional delay of the roll index pulse (RIP). Magnitude of delay is the integral of the commanded NADIR variable pulse delay rate. Capability to vary the rate from	AVPDND	NADIR variable phase delay - identifies magnitude (position) of the resultant spin angle delay of the NADIR VPD with respect to RIP. This is a pulse that is

Table 3.3.4-1 (continued)

CMD	Command Title	Command Description & Subsystem Internal Response	TM Mnemonic	TM Title and/or Remarks
DCEQ3 DCEQC (cont)		0° to 164.78° per minute in increments of 0.04024° for minute steps is provided (13 bits resolution).		routed once per spin period to selected scientific instruments (ORAD and OUVS) as the NADIR pulse (12 bits resolution).
DCEQ4 DCEQD	HGA Elevation Positioner Slew	In Figure 3.3.2.4-1, provides for slewing the HGA antenna in elevation at a rate from 0° to 39.93° per minute in increments of 0.00975 deg/min (13 bits resolution).	AELEVD	HGA elevation position - (analog voltage) provides the slew profile versus GMT time (8 bits resolution).
DCEQ5 DCEQE	Azimuth Position	In Figure 3.3.2.4-1, provides for positioning despun antenna at a specific spin angle delay from the selected REF. (SRR). Capability is provided to position the despun antenna from 0° to 360° in increments of 0.087933° using 2's complement notation (12 bits resolution).	ATTM1Z ATTM2Z	Time interval measurement of SRR to MIP (master index pulse) provides the despun antenna position. Profile versus GMT time. (16 bits resolution)
DCEQ6 DCEQF	Mode Command	In Figure 3.3.2.4-1, provides for control of internal functions as follows: 1) Phase reset enable/disable - This command is used to minimize phase transients generated by phase lock loop from being propagated to the despun antenna control. Phase transients are caused when switching from one inertial reference to another. Prior to switching from one inertial reference to	ADSPNS	Despin phase reset enable/disable - identifies the commanded state of the phase reset logic (= 1 for enabled state; = 0 for disabled state).

Table 3.3.4-1 (continued)

CMD	Command Title	Command Description & Subsystem Internal Response	TM Mnemonic	TM Title and/or Remarks
DCEQ6 DCEQF (cont)		another, this command should be sent in the disable state. After transients settle (as indicated by constant of SRR to MIP delay) and at all other times, the phase reset should be in the enable state.		
		2) RAM variable phase delay reset - resets the cumulative spin angle delay to be coincident with RIP delay minus the VPD pulse width (≈0.004°) (with respect to the SRR).	AVPDRD	RAM variable phase delay - identifies magnitude of the additional spin angle delay from the RIP - will read 0° in response to reset command.
		3) NADIR variable phase delay reset - resets the cumulative spin angle delay to be coincident with RIP delay minus VPD pulse width (≈0.004°)	AVPDND	NADIR variable phase delay - identifies magnitude of the additional spin angle delay from the RIP - will read 0° in response to the reset command.
		4) HGA positioner ON/OFF - provides fused 28 Vdc (fused source is 28 V switched loads bus - shown in Figure 3.3.2.4-1) drive power to the stepper motor based upon the selected HGA elevation positioner slew command.	APOS1S	HGA positioner electronics 1 ON/OFF = 1 for ON; = 0 for OFF
			APOS2S	HGA positioner electronics 2 ON/OFF = 1 for ON; = 0 for OFF

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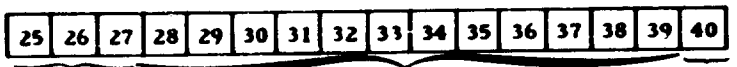
Table 3.3.4-1 (continued)

CMD	Command Title	Command Description & Subsystem Internal Response	TM Mnemonic	TM Title and/or Remarks

DCE QUANTITATIVE COMMAND SUMMARY

CMD: DCEQ1 (DCE No. 1 Only)
DCEQA (DCE No. 2 Only)

Bit location in command word



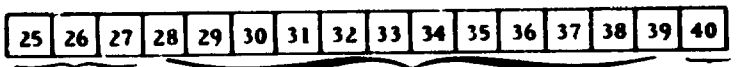
Address 0 0 0

Magnitude Information
Azimuth Pointing Slew

MSB
Direction of rotation:
(0 = same as rotor spin;
1 = despin direction)

CMD: DCEQ2 (DCE No. 1 Only)
DCEQB (DCE No. 2 Only)

Bit location in command word



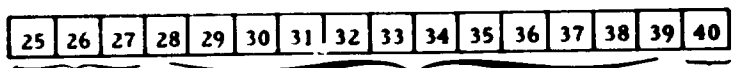
Address 1 0 0

Magnitude Information
RAM Variable Phase Delay

MSB
("0" advances RAM pulse in direction of spin; "1" moves RAM pulse opposite to direction of spin)

CMD: DCEQ3 (DCE No. 1 Only)
DCEQC (DCE No. 2 Only)

Bit location in command word



Address 0 1 0

Magnitude Information
NADIR Variable Phase Delay

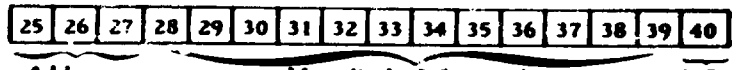
MSB
("0" advances NADIR pulse in direction of spin; "1" moves NADIR pulse opposite to direction of spin)

Table 3. 3. 4-1 (continued)

CMD	Command Title	Command Description & Subsystem Internal Response	TM Mnemonic	TM Title and/or Remarks

CMD: DCEQ4 (DCE No. 1 Only)
DCEQD (DCE No. 2 Only)

Bit location in command word



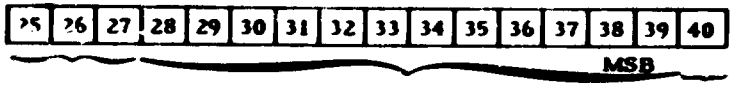
Address 1 1 0

Magnitude Information
HGA Elevation Positioner Slew

MSB
("0" rotates dish toward +Z axis;
"1" rotates dish toward -Z axis)

CMD: DCEQ5 (DCE No. 1 Only)
DCEQE (DCE No. 2 Only)

Bit location in command word



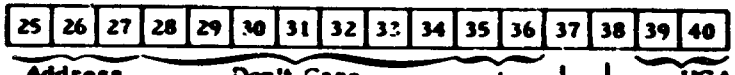
Address 0 0 1

Magnitude Information
Azimuth Position

Don't Care

CMD: DCEQ6 (DCE No. 1 Only)
DCEQF (DCE No. 2 Only)

Bit location in command word



Address 1 0 1

Don't Care

HGA Positioner ON/OFF Control

RAM VPD Reset

NADIR VPD Reset

Phase Reset Enable/Disable Control

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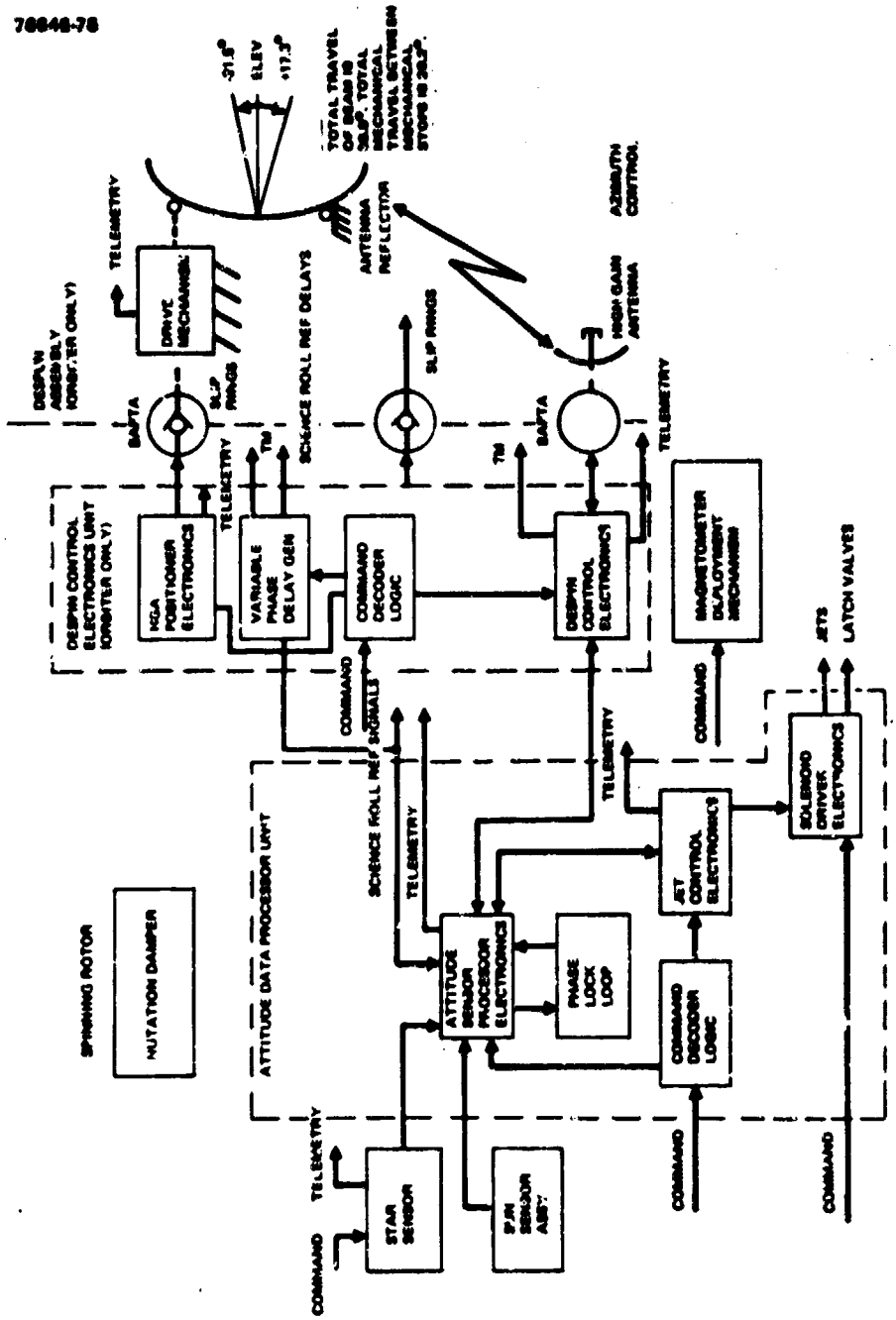


FIGURE 3.3.1-1. CONTROLS SUBSYSTEM BLOCK DIAGRAM

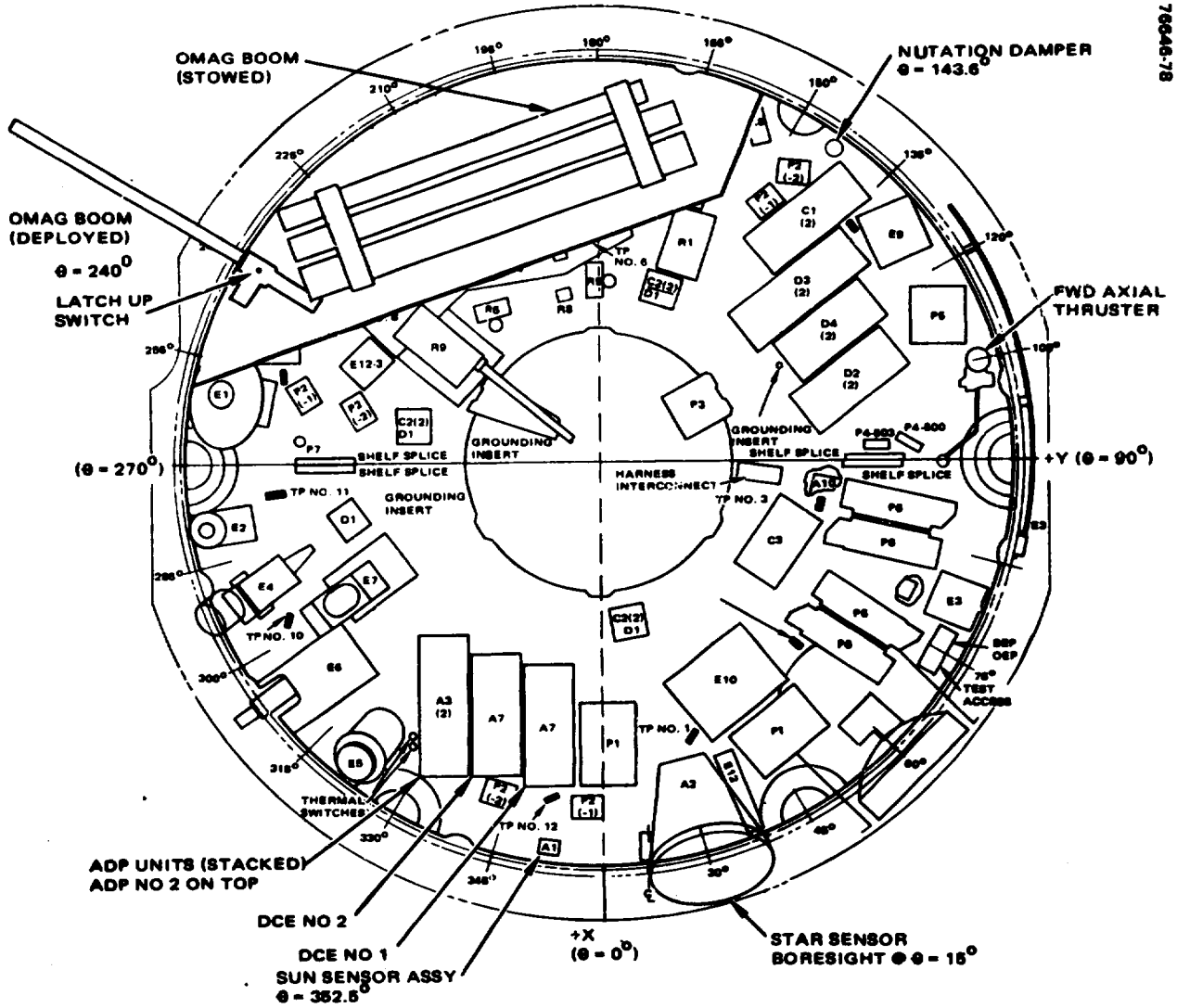


FIGURE 3.3.1-2. SHELF LAYOUT OF CONTROLS SUBSYSTEM UNITS

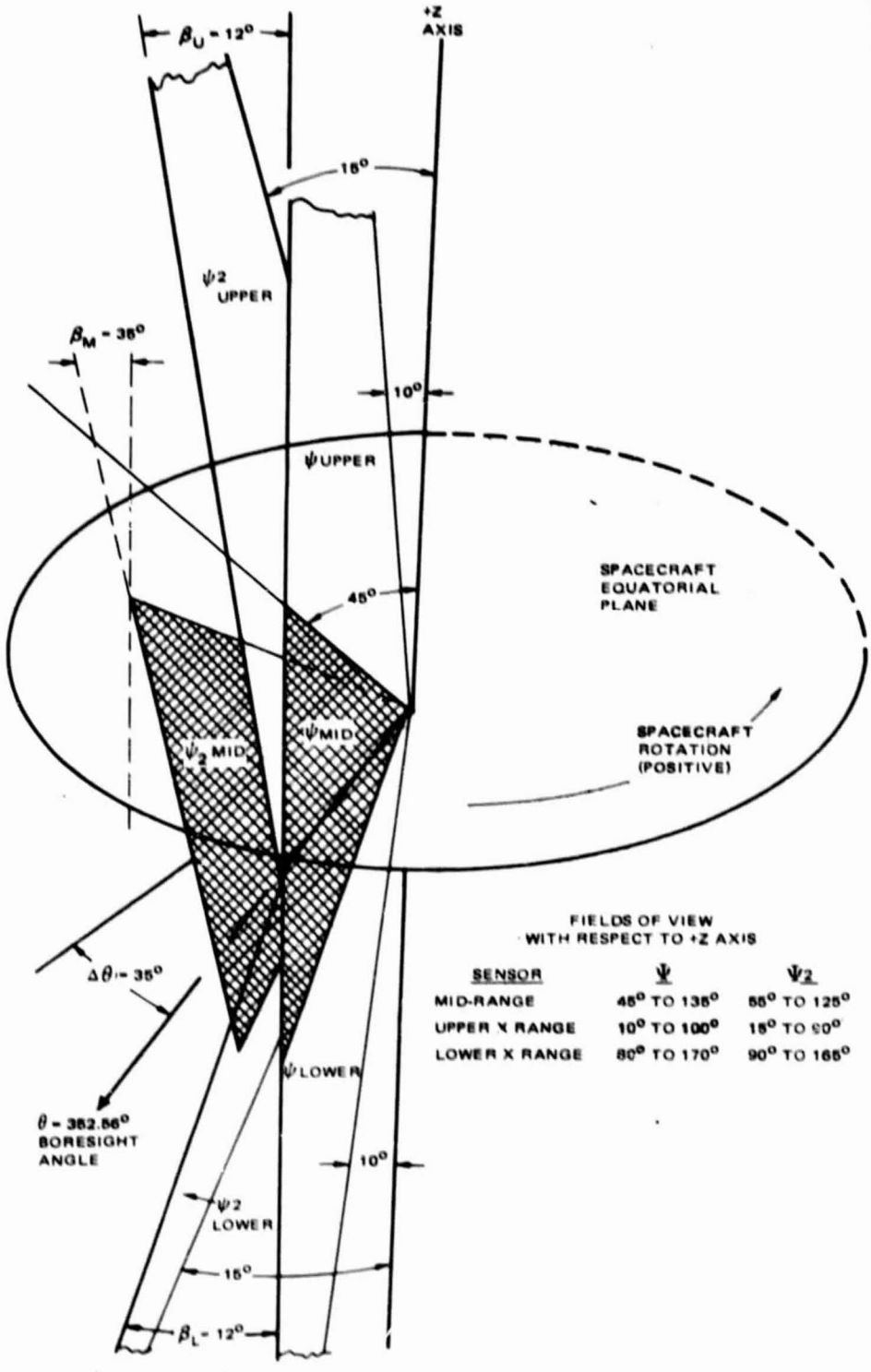
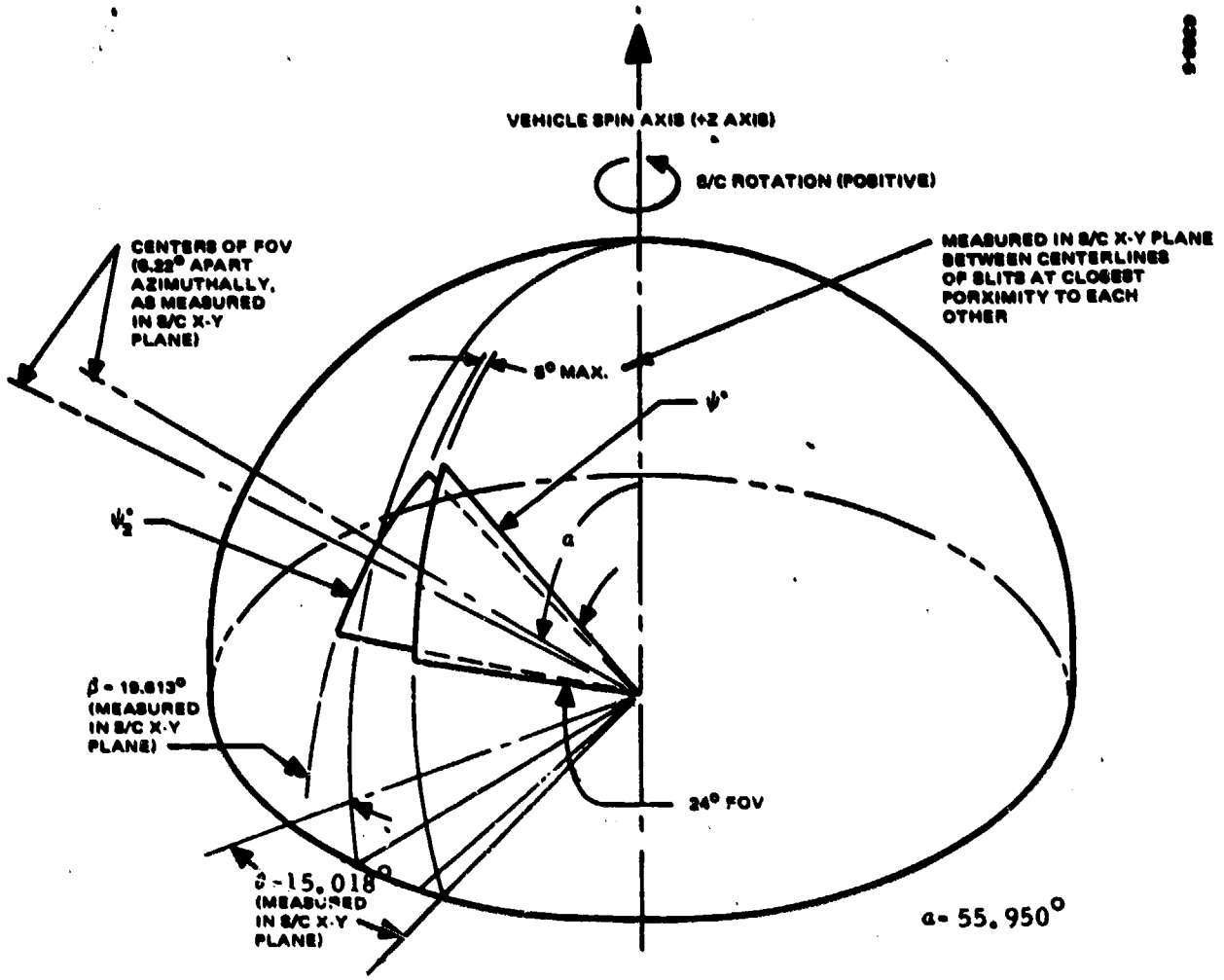
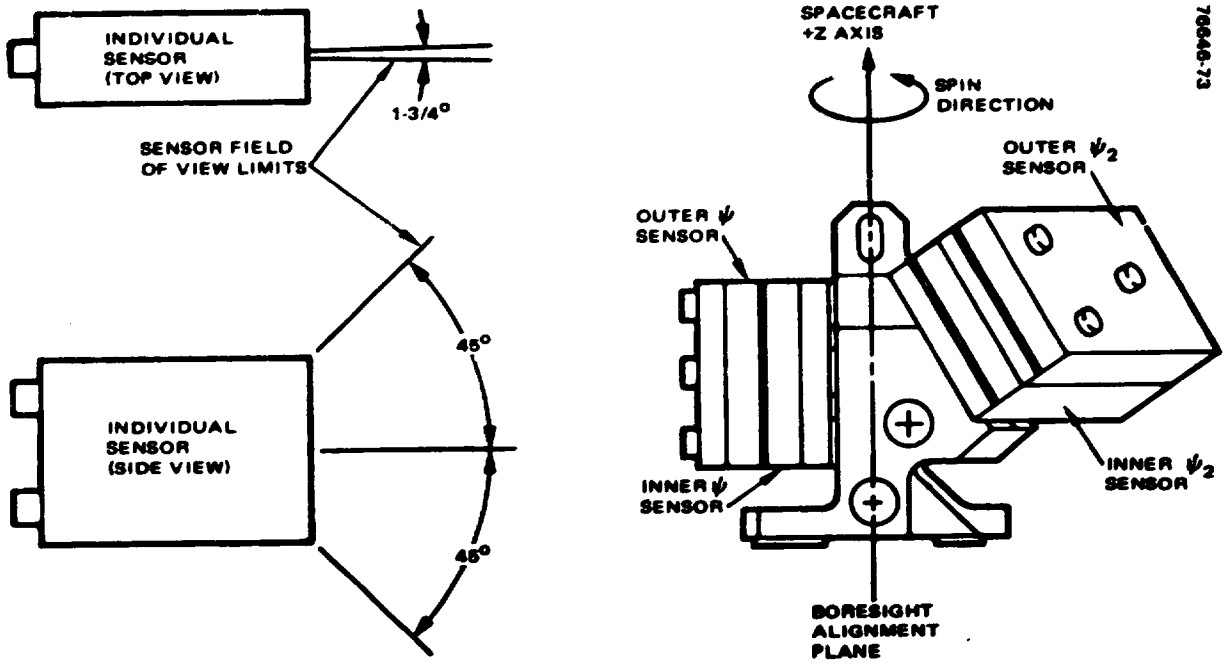


FIGURE 3.3.1-3. SUN SENSOR FIELDS OF VIEW



BORESIGHT OF STAR SENSOR IS TRAILING EDGE OF PSI 2° SLIT AT THE CENTER OF ELEVATION FOV.

Figure 3.3.1-4. Star Sensor Field-Of-View

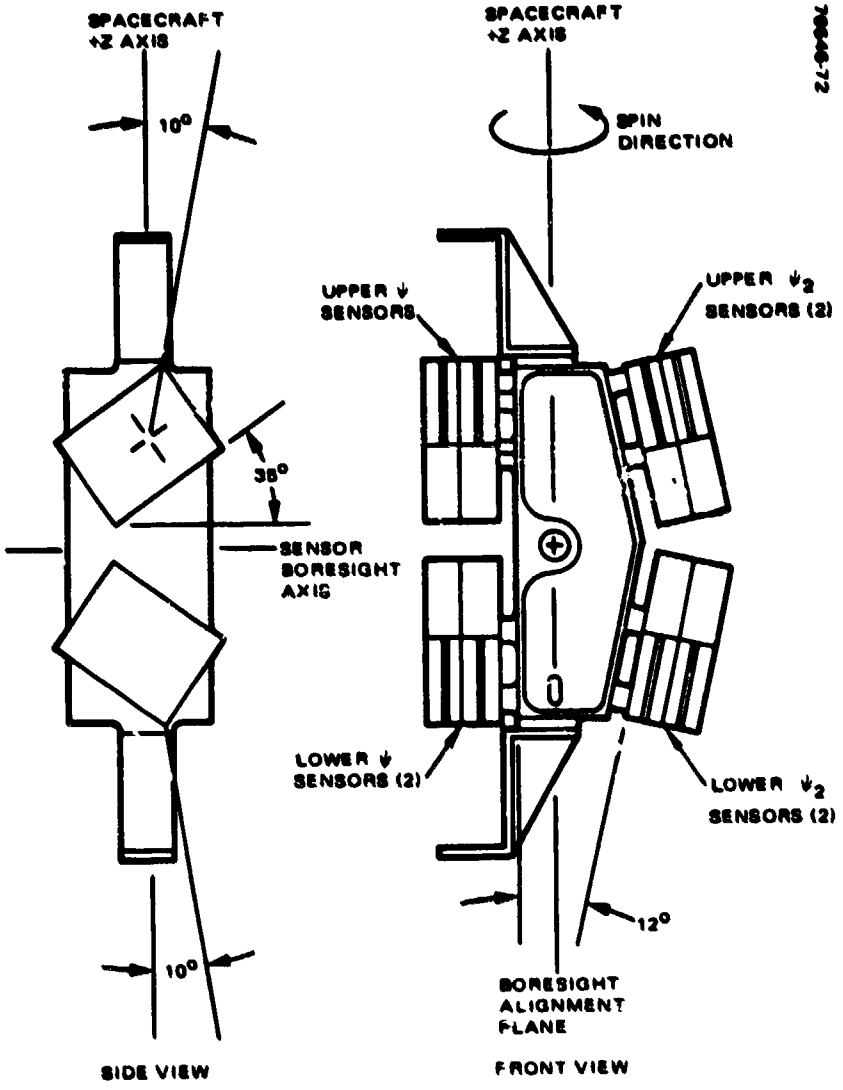


"$\psi - \psi_2$" = $\sin^{-1}$ [COT γ TAN 35°] + 35°

WHERE "$\psi - \psi_2$" EQUALS THE NOMINAL SPIN ANGLE BETWEEN ψ AND ψ_2 PULSES.

(35° TERM REFLECTS 35° BIAS SEEN FOR MID-RANGE ψ_2 IN FIGURE 3.3.2.1-5.)

FIGURE 3.3.2.1-1. MID-RANGE SUN SENSOR GEOMETRY



FOR UPPER SENSORS:

$$\text{"}\psi - \psi_2\text{"} = \sin^{-1}[\cot\gamma \tan 12^\circ]$$

FOR LOWER SENSORS:

$$\text{"}\psi - \psi_2\text{"} = -\sin^{-1}[\cot\gamma \tan 12^\circ]$$

WHERE "$\psi - \psi_2$" EQUALS NOMINAL SPIN ANGLE BETWEEN ψ AND ψ_2 PULSES

FIGURE 3.3.2.1-2. EXTENDED RANGE SUN SENSOR GEOMETRY

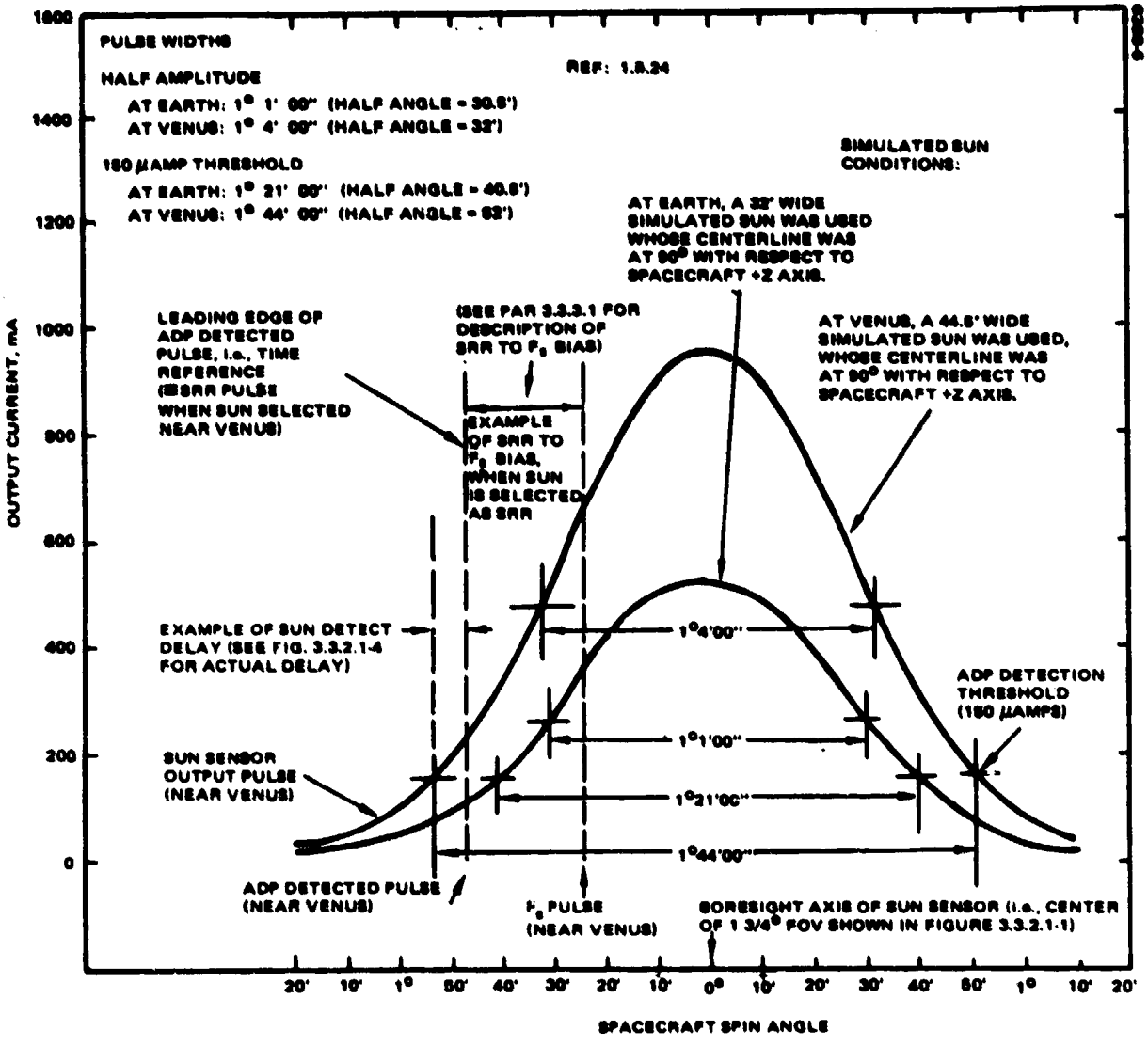


FIGURE 3.3.2.1-3. SUN SENSOR PULSE VARIATION AND TIME RELATIONSHIP TO SRR PULSE AND F_8 PULSE

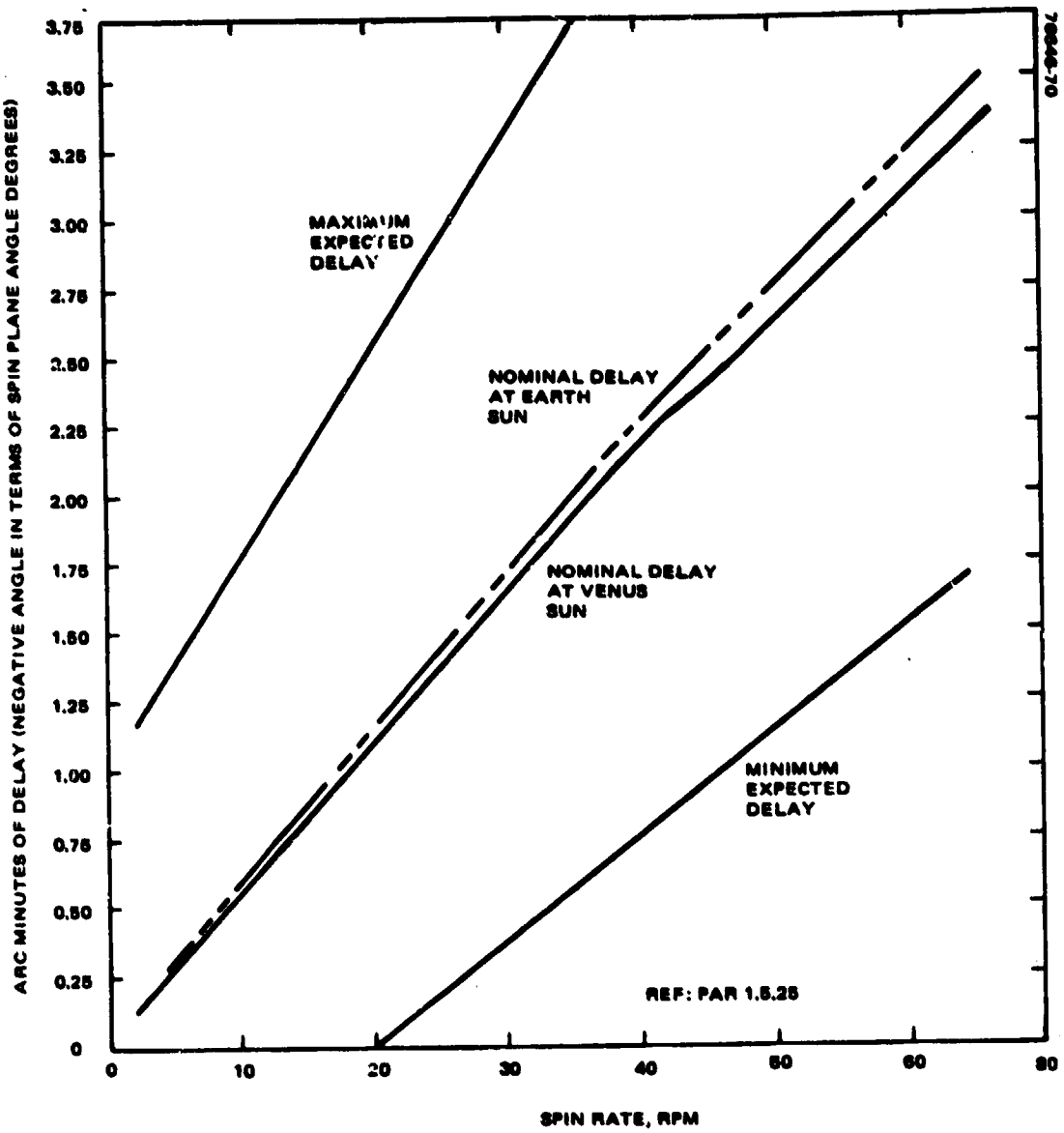


FIGURE 3.3.2.1-4. SUN DETECT DELAY IN SUN SENSOR OUTPUT PULSE vs SPACECRAFT SPIN RATE

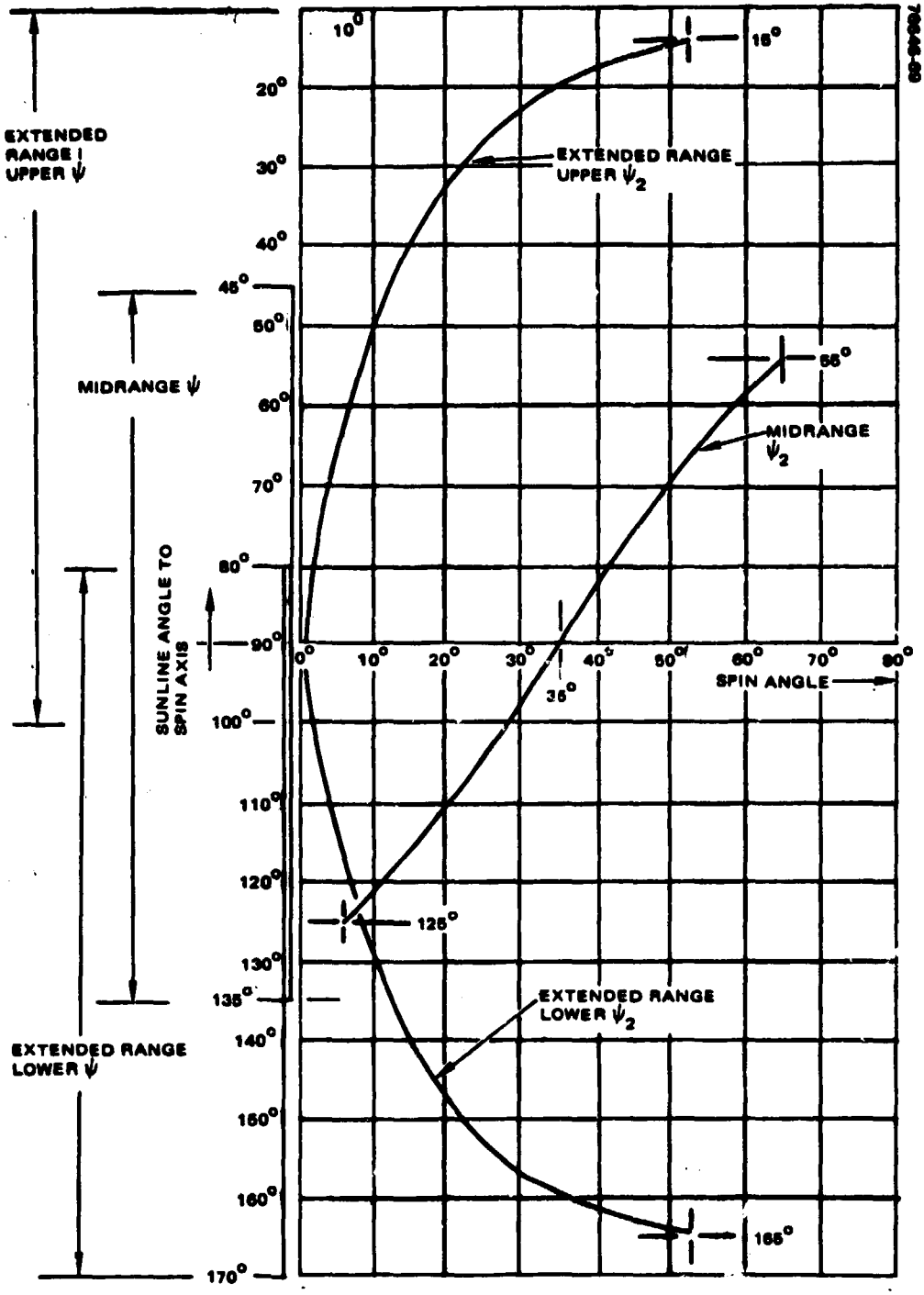


FIGURE 3.3.2.1-5. NOMINAL PSI_2 SPIN ANGLE DELAY vs SUNLINE ANGLE

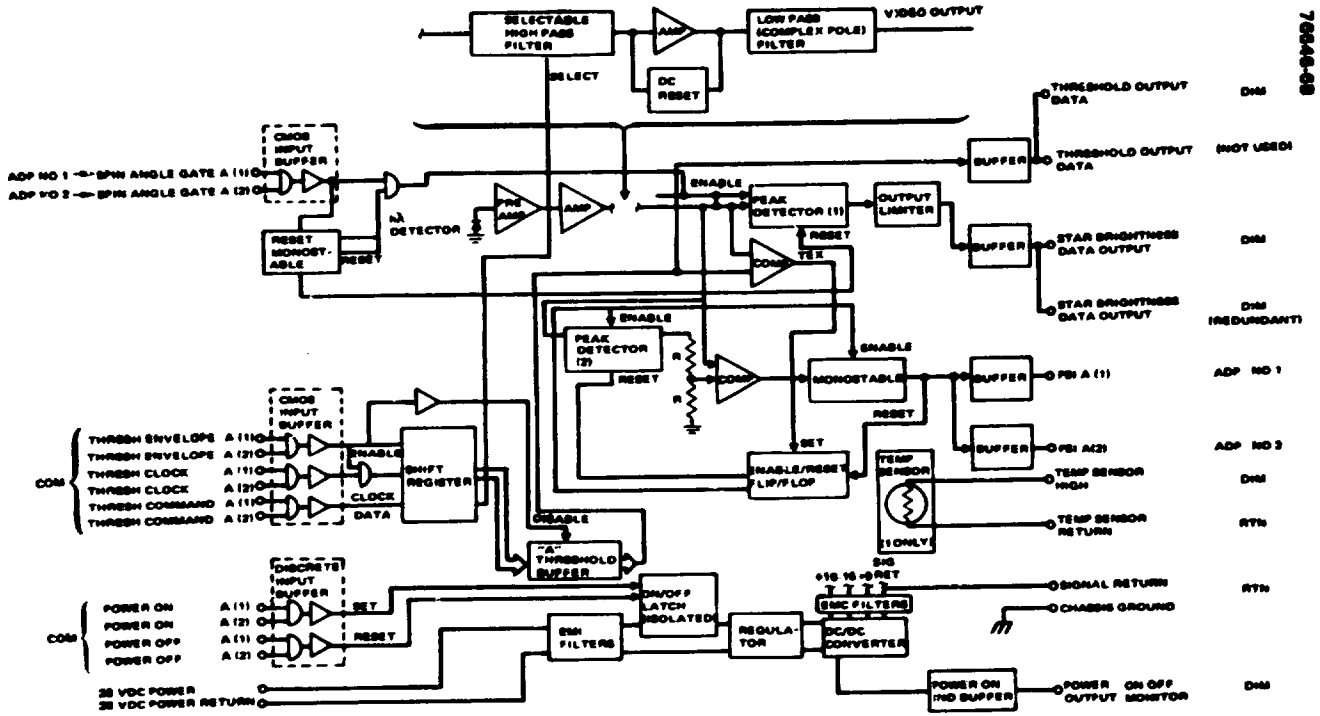


FIGURE 3.3.2.2-1. STAR SENSOR BLOCK DIAGRAM (ONE OF TWO CHANNELS)

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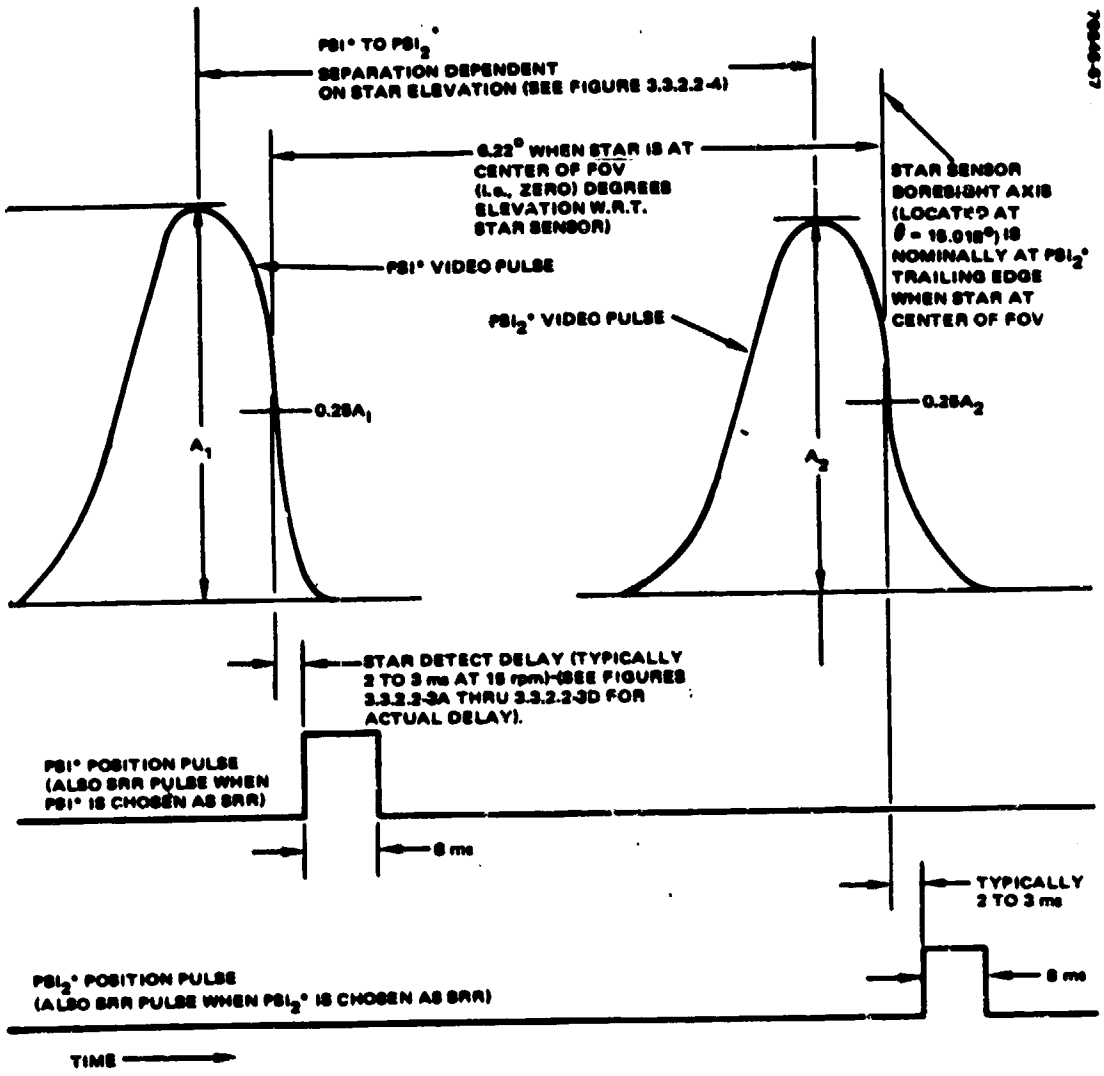


FIGURE 3.3.2.2-2. STAR SENSOR VIDEO PULSE AND POSITION PULSE RELATIONSHIPS

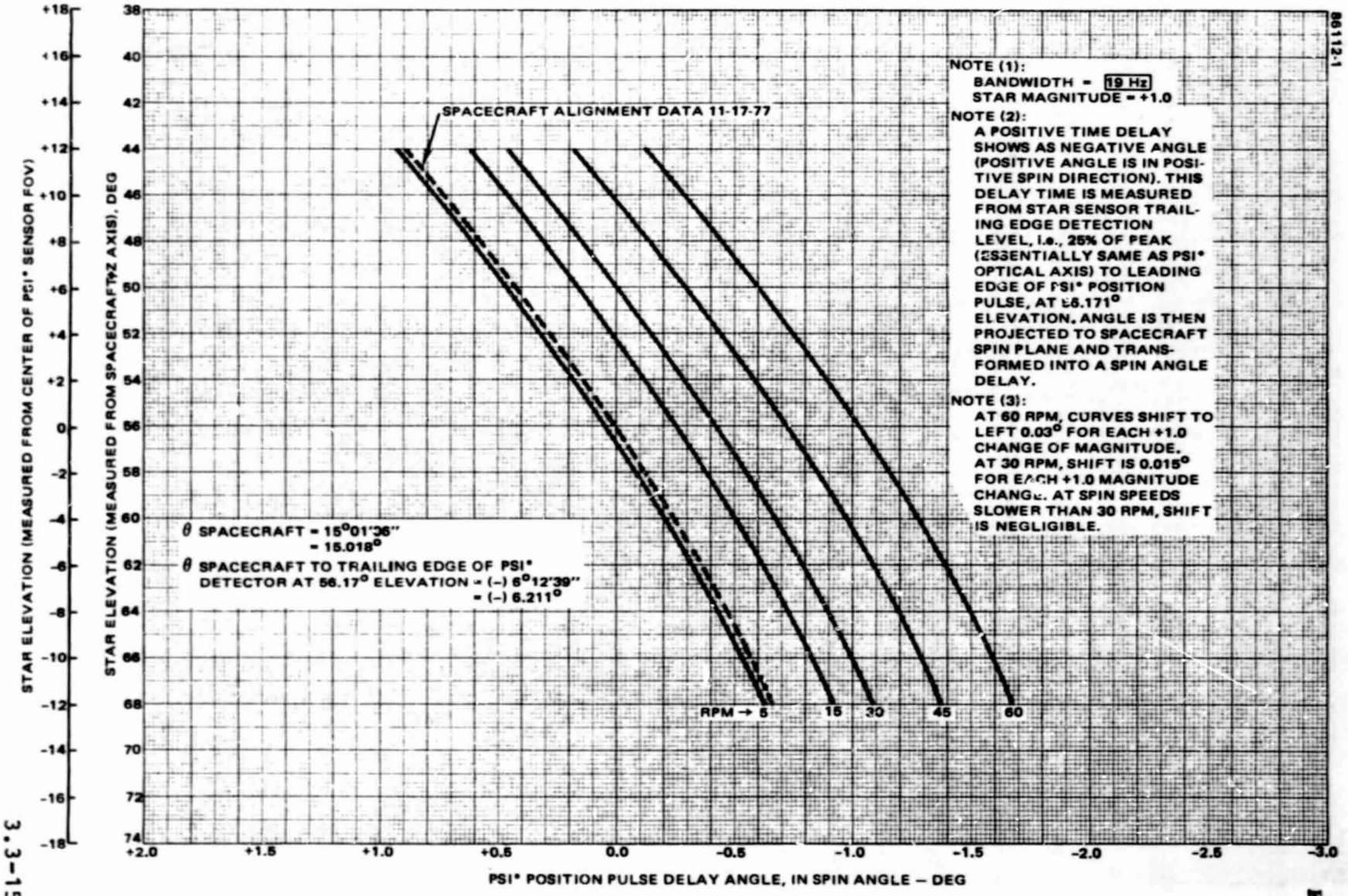


FIGURE 3.3.2.2-3a. ORBITER PSI* POSITION PULSE DELAY ANGLE VERSUS STAR ELEVATION, STAR BRIGHTNESS, AND SPACECRAFT SPIN RATE (19 Hz)

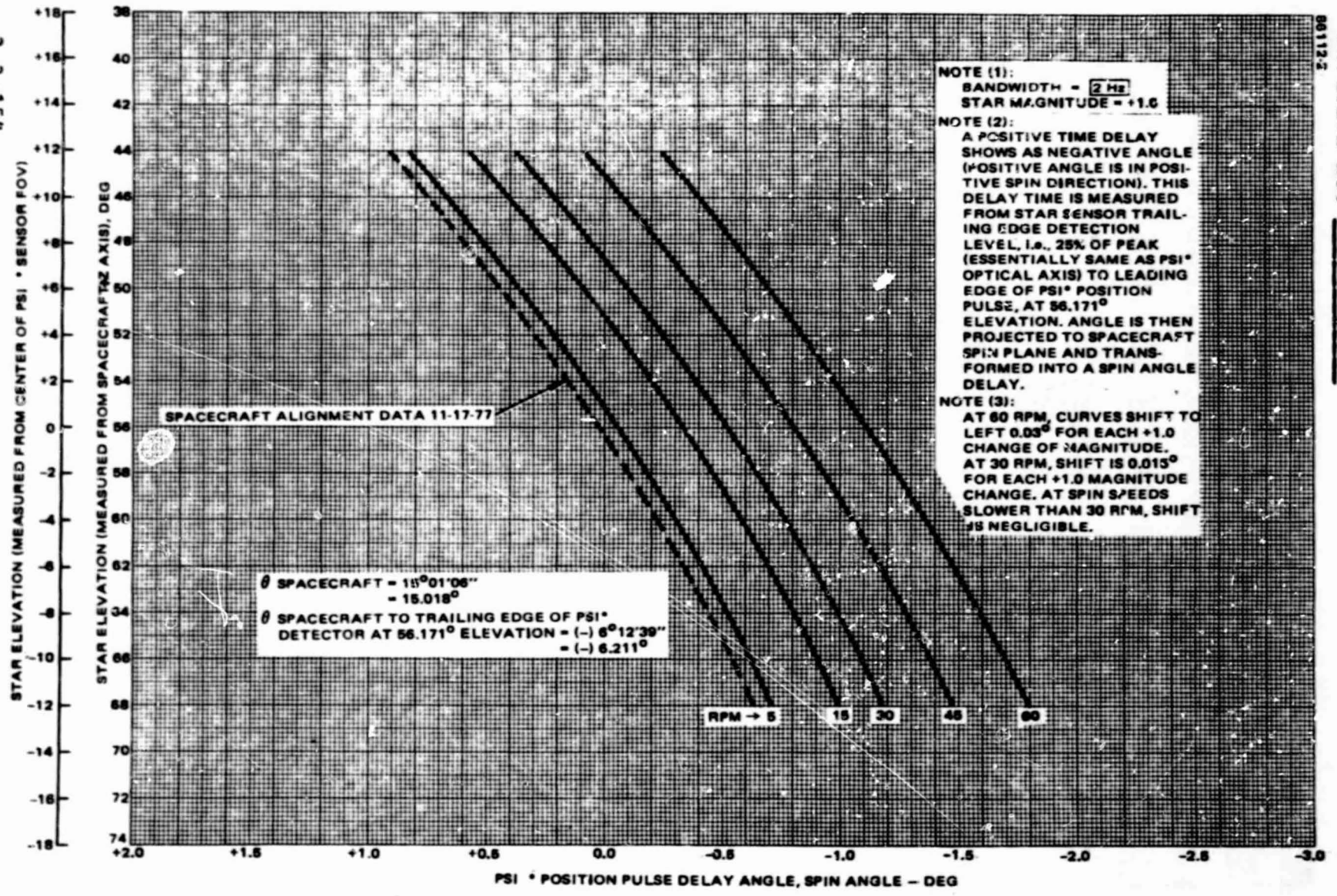


FIGURE 3.3.2.2-3b. ORBITER PSI* POSITION PULSE DELAY ANGLE VERSUS STAR ELEVATION, STAR BRIGHTNESS, AND SPACECRAFT SPIN RATE (2 Hz)

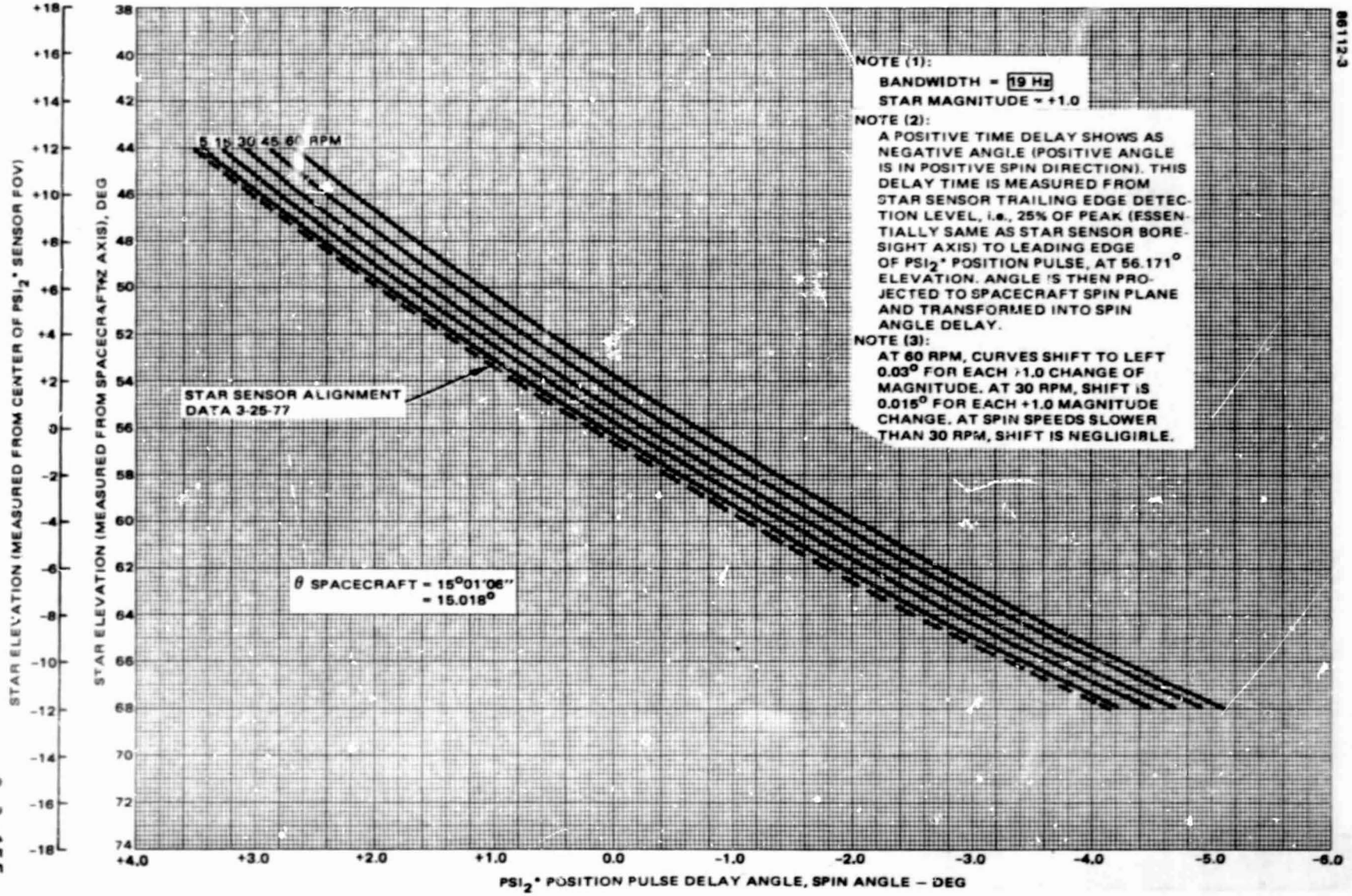


FIGURE 3.3.2.2-3c. ORBITER PSI_2* POSITION PULSE DELAY ANGLE VERSUS STAR ELEVATION, STAR BRIGHTNESS, AND SPACECRAFT SPIN RATE (19 Hz)

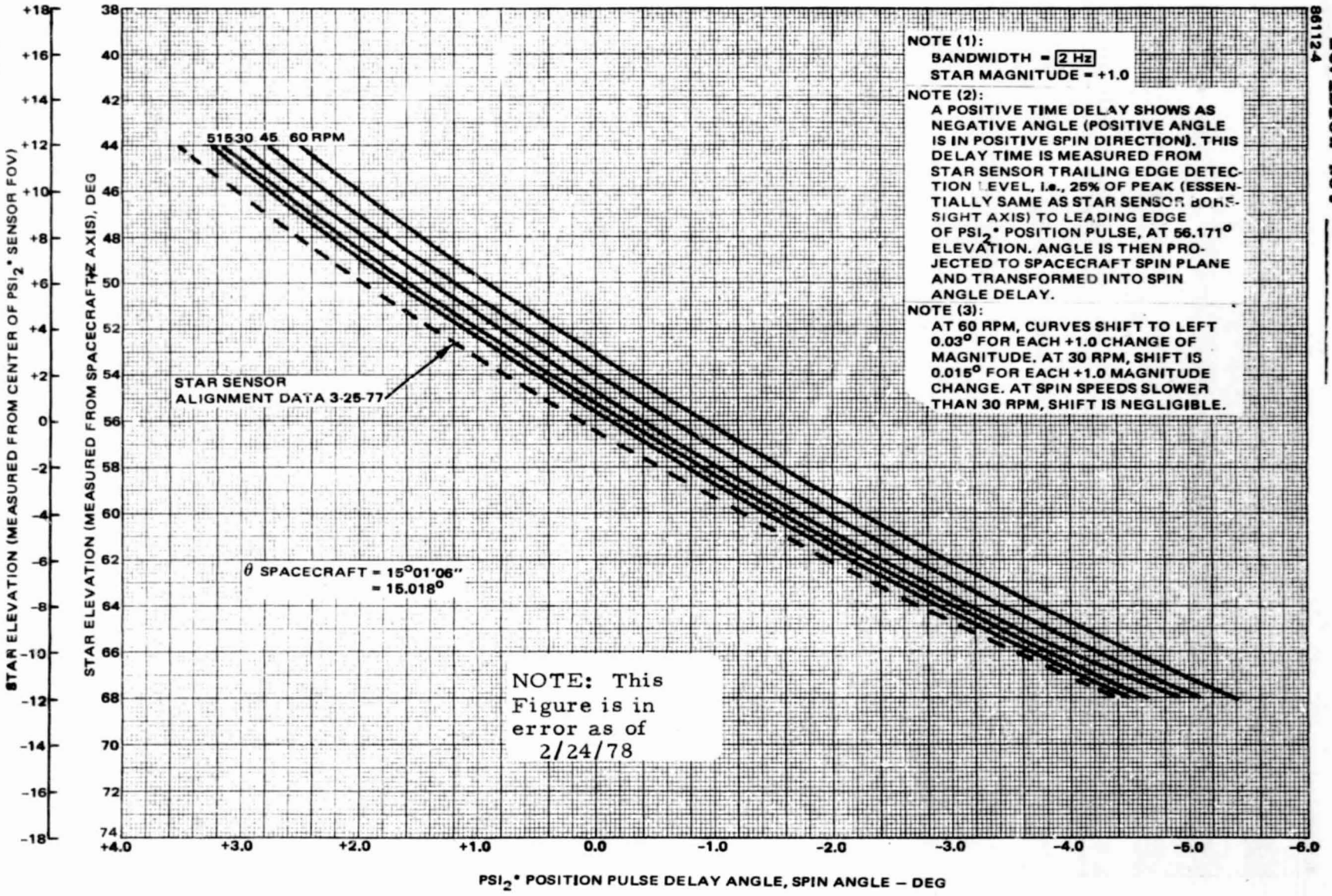
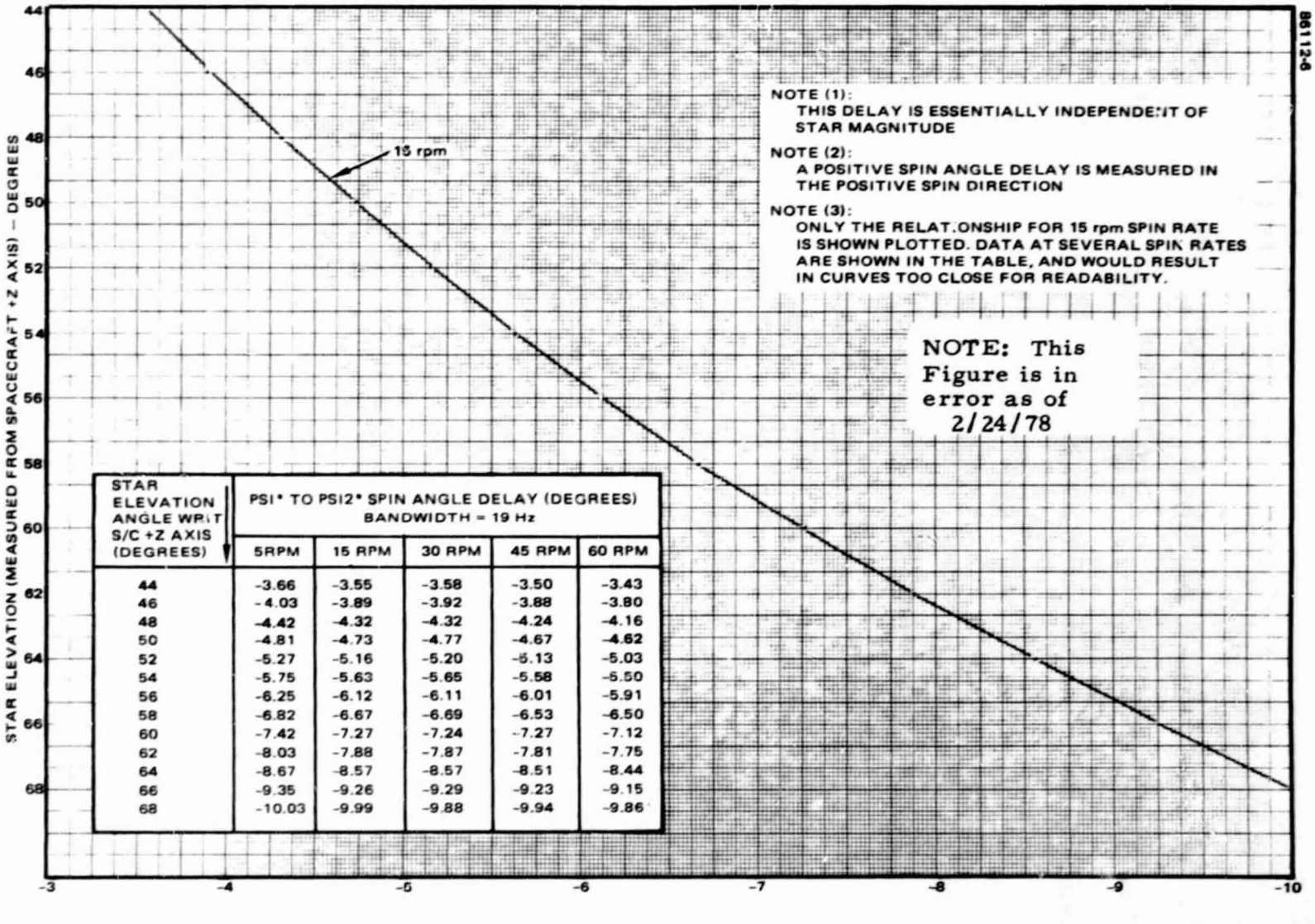


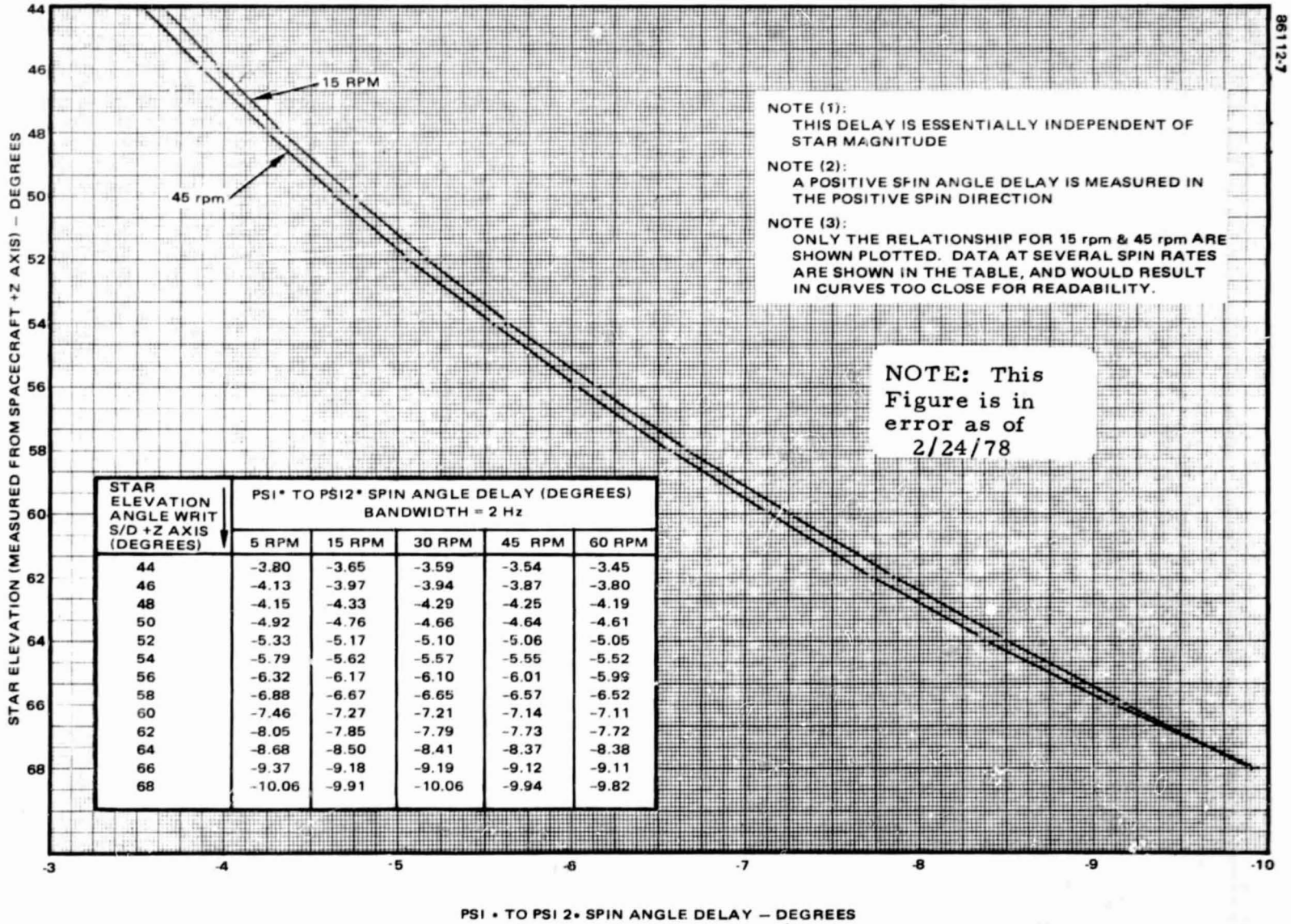
FIGURE 3.3.2.2-3d. ORBITER PSI_2^* POSITION PULSE DELAY ANGLE VERSUS STAR ELEVATION, STAR BRIGHTNESS, AND SPACECRAFT SPIN RATE (2 Hz)



STAR ELEVATION ANGLE WRT S/C +Z AXIS (DEGREES)	PSI* TO PSI2* SPIN ANGLE DELAY (DEGREES) BANDWIDTH = 19 Hz				
	5RPM	15 RPM	30 RPM	45 RPM	60 RPM
44	-3.66	-3.55	-3.58	-3.50	-3.43
46	-4.03	-3.89	-3.92	-3.88	-3.80
48	-4.42	-4.32	-4.32	-4.24	-4.16
50	-4.81	-4.73	-4.77	-4.67	-4.62
52	-5.27	-5.16	-5.20	-5.13	-5.03
54	-5.75	-5.63	-5.65	-5.58	-5.50
56	-6.25	-6.12	-6.11	-6.01	-5.91
58	-6.82	-6.67	-6.69	-6.53	-6.50
60	-7.42	-7.27	-7.24	-7.27	-7.12
62	-8.03	-7.88	-7.87	-7.81	-7.75
64	-8.67	-8.57	-8.57	-8.51	-8.44
66	-9.35	-9.26	-9.29	-9.23	-9.15
68	-10.03	-9.99	-9.88	-9.94	-9.86

FIGURE 3.3.2.2-4a. PSI* TO PSI2* SPIN ANGLE DELAY, WHEN BOTH SLITS ARE USED SEQUENTIALLY FOR GATE B AND THE SAME MEASUREMENT STAR (BANDWIDTH = 19 Hz)

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STAR ELEVATION ANGLE WRT S/D +Z AXIS (DEGREES)	PSI* TO PSI2* SPIN ANGLE DELAY (DEGREES) BANDWIDTH = 2 Hz				
	5 RPM	15 RPM	30 RPM	45 RPM	60 RPM
44	-3.80	-3.65	-3.59	-3.54	-3.45
46	-4.13	-3.97	-3.94	-3.87	-3.80
48	-4.15	-4.33	-4.29	-4.25	-4.19
50	-4.92	-4.76	-4.66	-4.64	-4.61
52	-5.33	-5.17	-5.10	-5.06	-5.05
54	-5.79	-5.62	-5.57	-5.55	-5.52
56	-6.32	-6.17	-6.10	-6.01	-5.99
58	-6.88	-6.67	-6.65	-6.57	-6.52
60	-7.46	-7.27	-7.21	-7.14	-7.11
62	-8.05	-7.85	-7.79	-7.73	-7.72
64	-8.68	-8.50	-8.41	-8.37	-8.38
66	-9.37	-9.18	-9.19	-9.12	-9.11
68	-10.06	-9.91	-10.06	-9.94	-9.82

Figure 3.3.2.2-4B. PSI* to PSI2* Spin Angle Delay, When Both Slits are used Sequentially for Gate B and the Same Measurement Star (Bandwidth = 2 Hz)

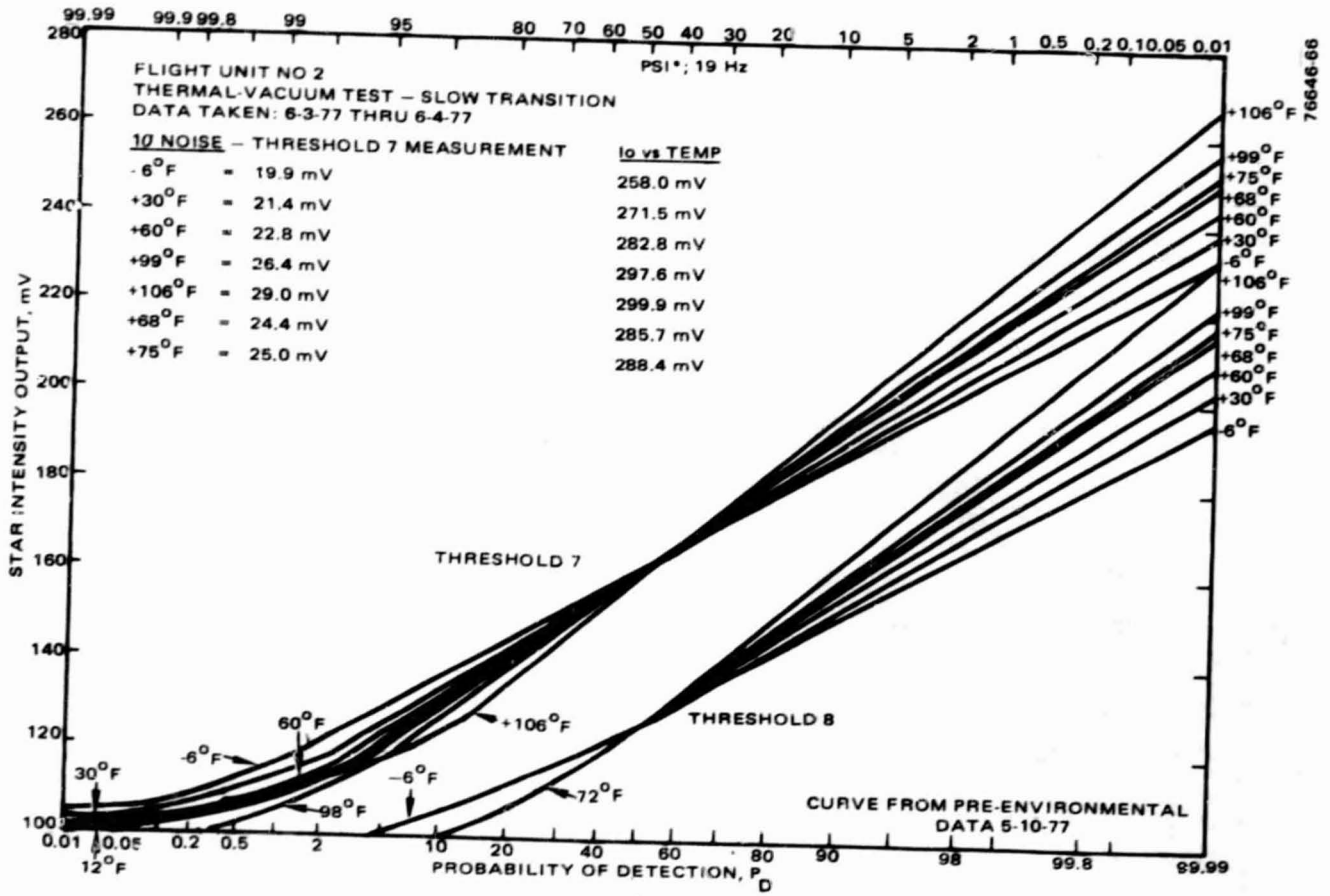


FIGURE 3.3.2.2-5A. EFFECT OF TEMPERATURE ON STAR P_D

PSI* Channel& 19Hz Bandpass in Use

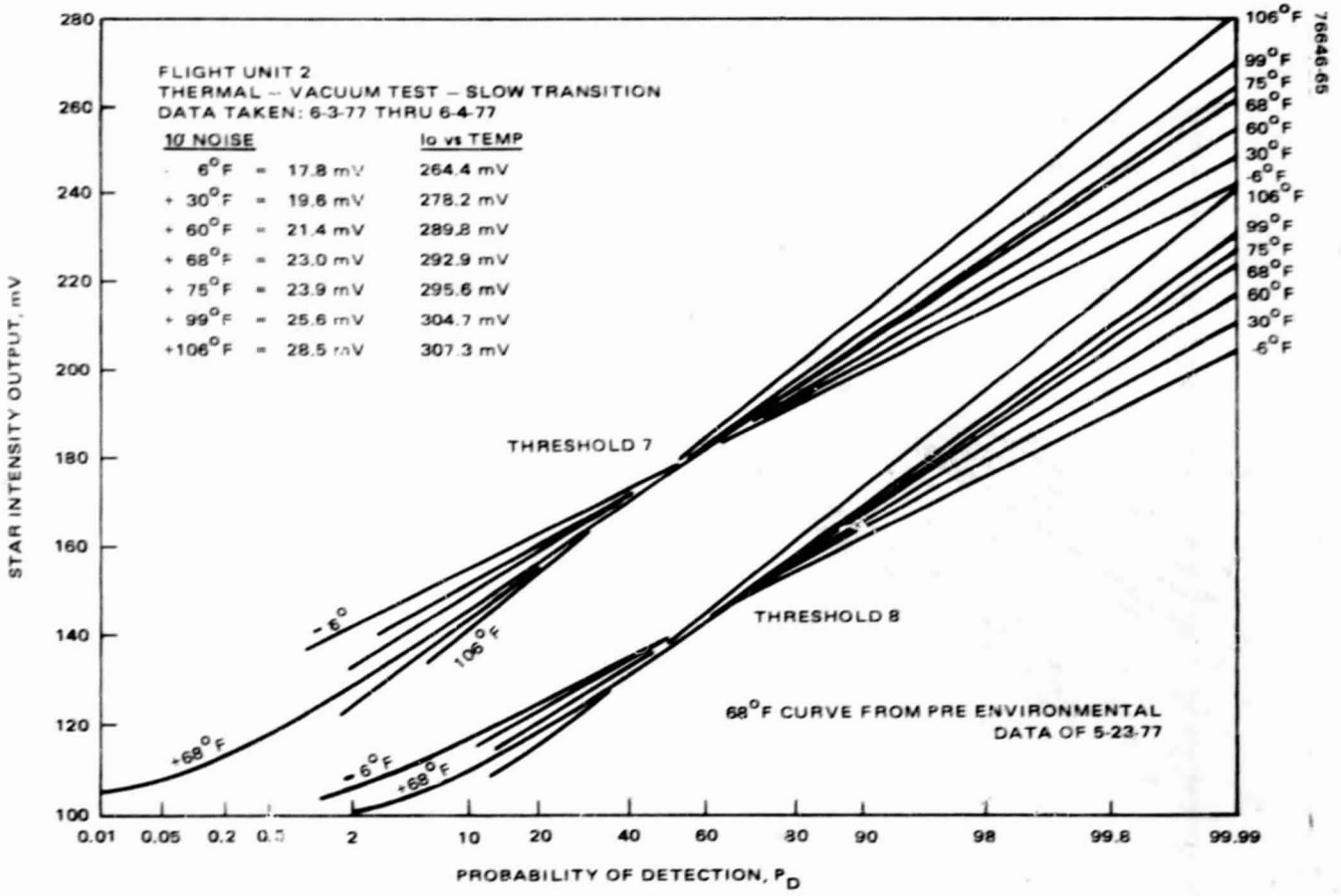


Figure 3.3.2.2-58. Effect of Temperature on Star P_D PSI2* Channel and 19 Hz Bandpass in Use

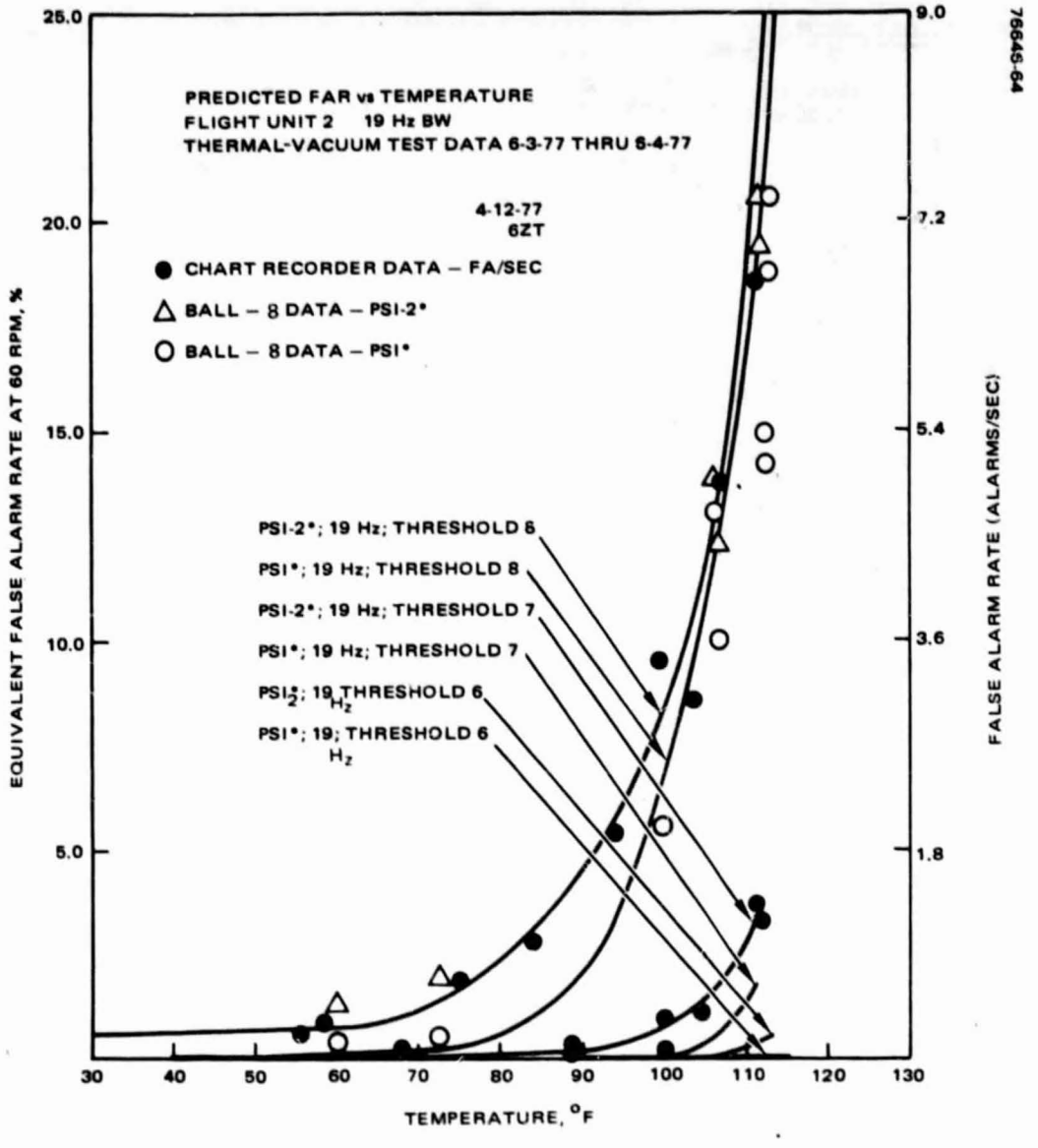


FIGURE 3.3.2.2-5C. FALSE ALARM RATE (STAR DETECTION) vs TEMPERATURE, STAR CHANNEL, AND SELECTED THRESHOLD

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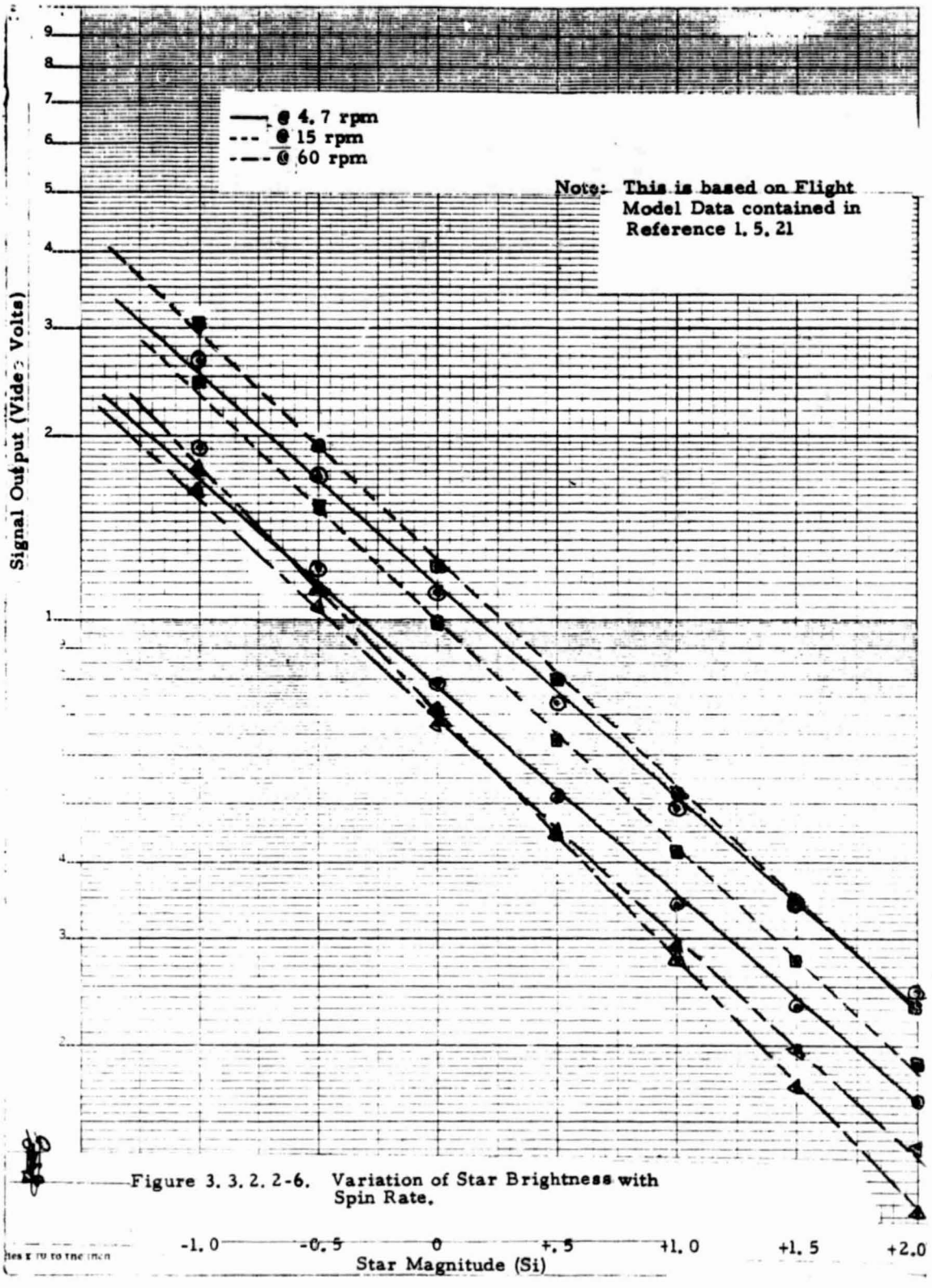


Figure 3.3.2.2-6. Variation of Star Brightness with Spin Rate.

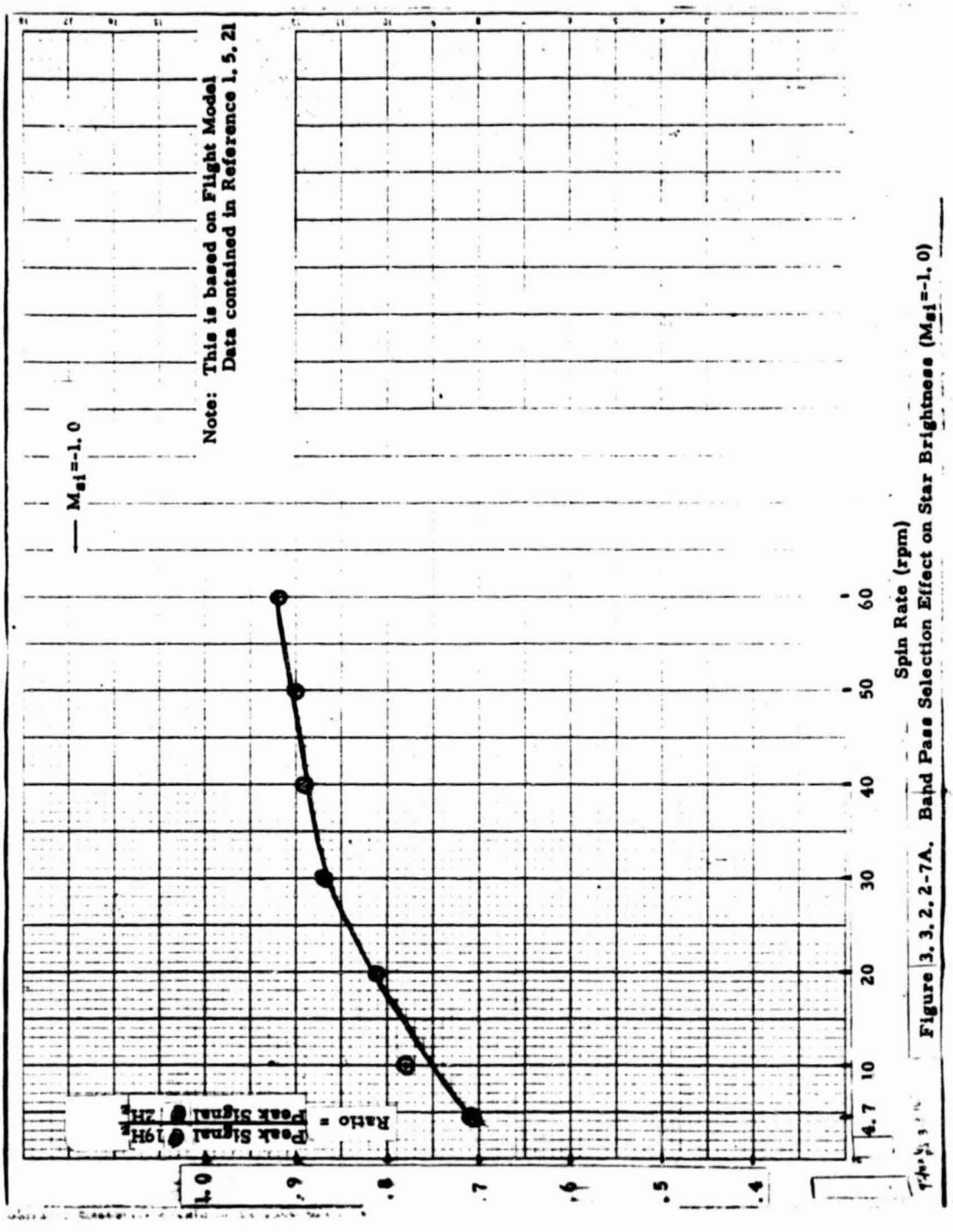


Figure 3.3.2.2-7A. Band Pass Selection Effect on Star Brightness (M_{sl}=-1.0)

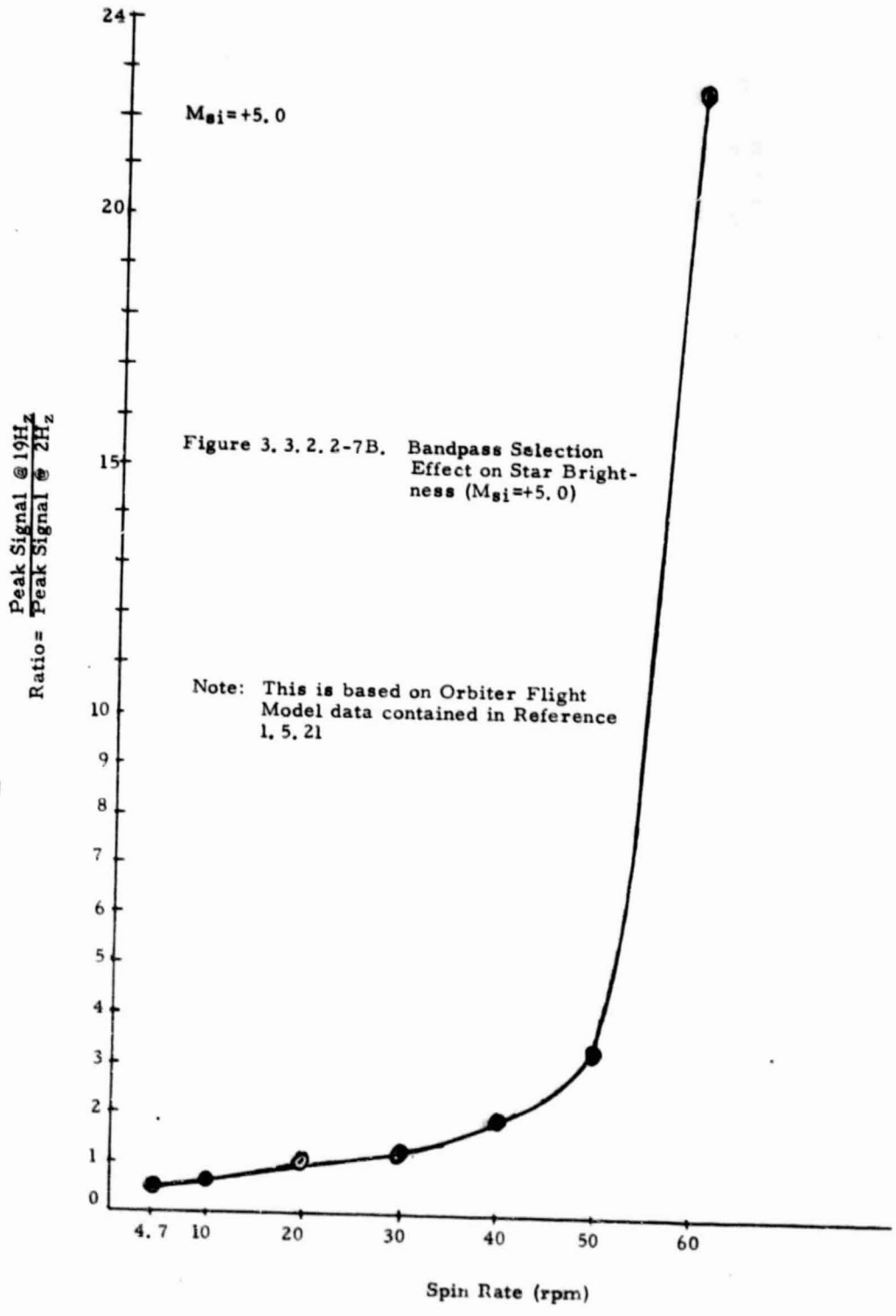


Figure 3.3.2.2-7B. Bandpass Selection Effect on Star Brightness (M_{si}=+5.0)

Note: This is based on Orbiter Flight Model data contained in Reference 1.5.21

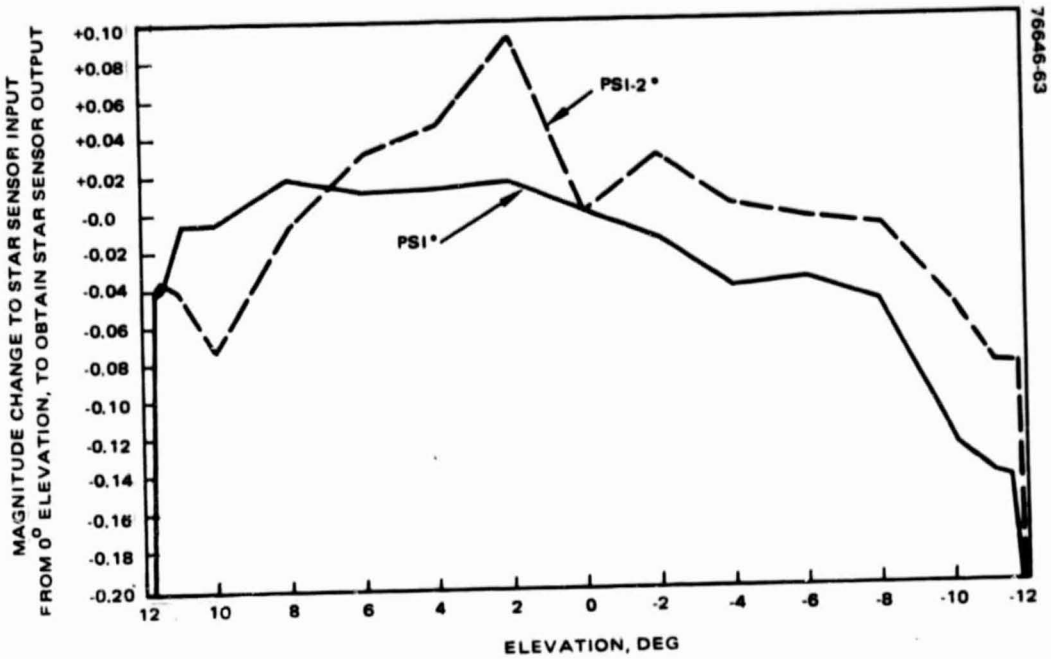


FIGURE 3.3.2.2-8. VARIATION OF STAR BRIGHTNESS vs ELEVATION FOV WITH RESPECT TO STAR SENSOR CENTER FOV (56° FROM SPACECRAFT +Z AXIS)

This Figure is a Foldout.

See APPENDIX C

Figure 3.3.2.3-1.

Sun Sensors, Star Sensors Electronics, ADP,
Jets & Latch Valves Cmd. & TM Logic Flow Diagram

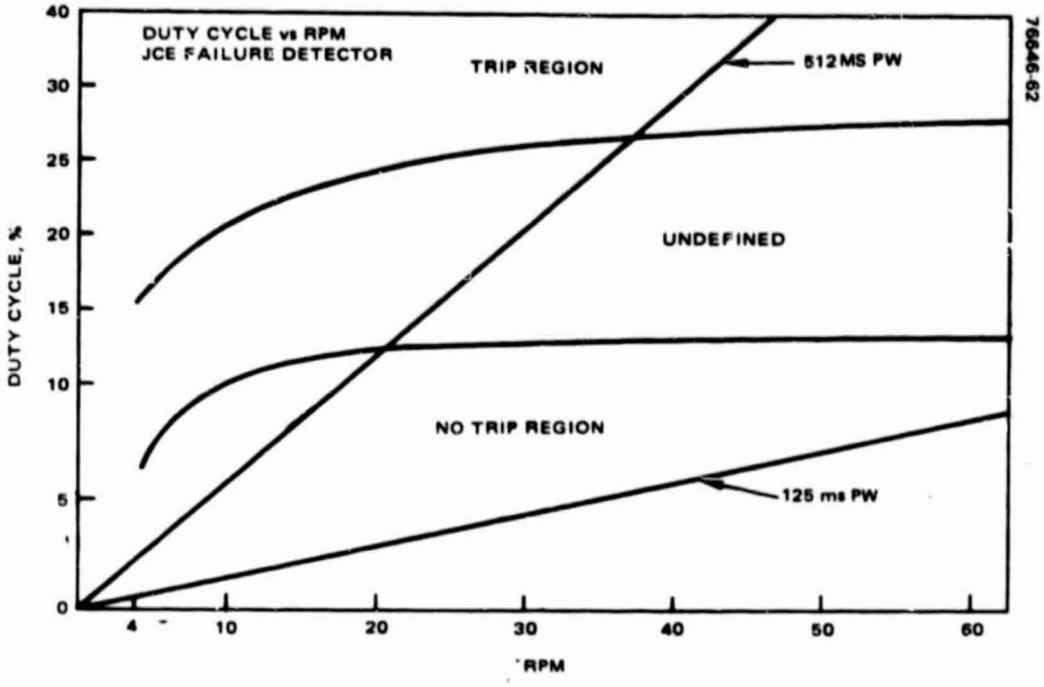


FIGURE 3.3.2.3-2. JCE DUTY CYCLE FAILURE DETECTION

This Figure is a Foldout.

See APPENDIX C

Figure 3.3.2.4-1.

DCE Command & TM Logic Flow Diagram

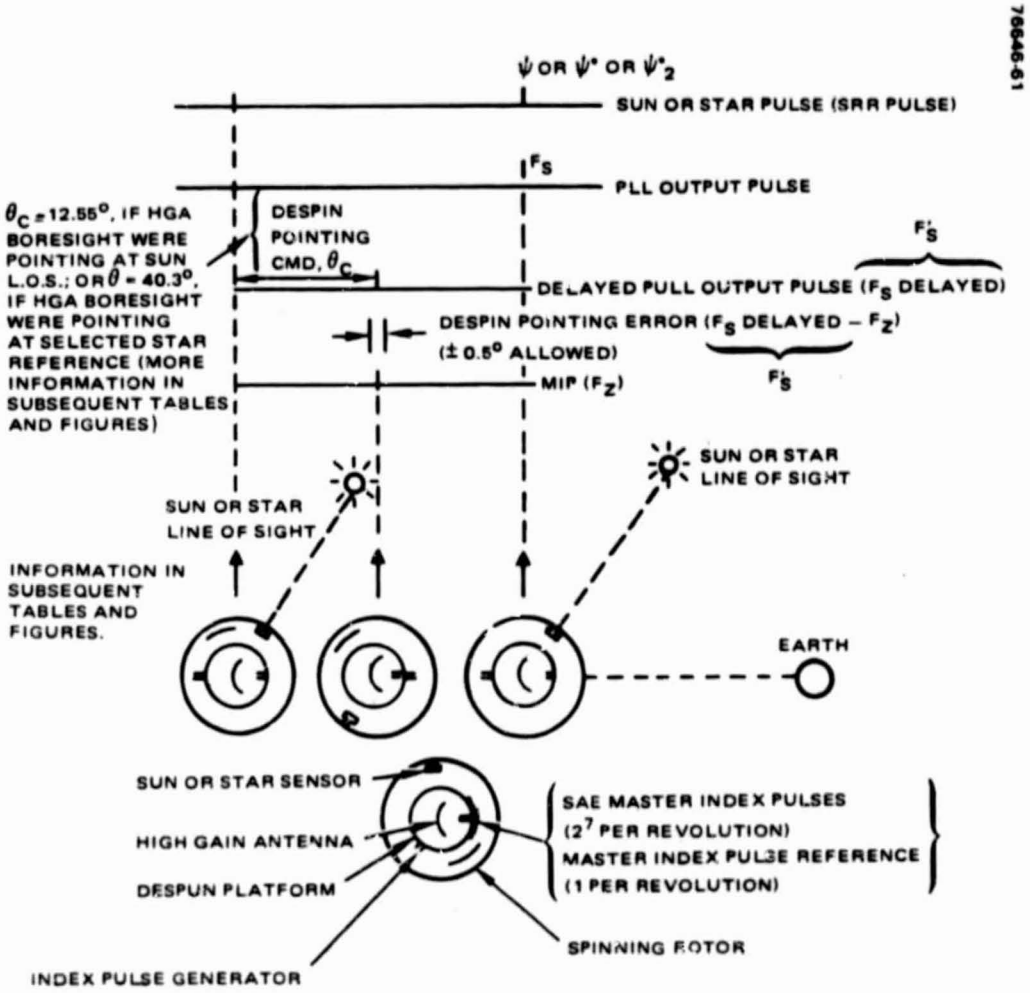
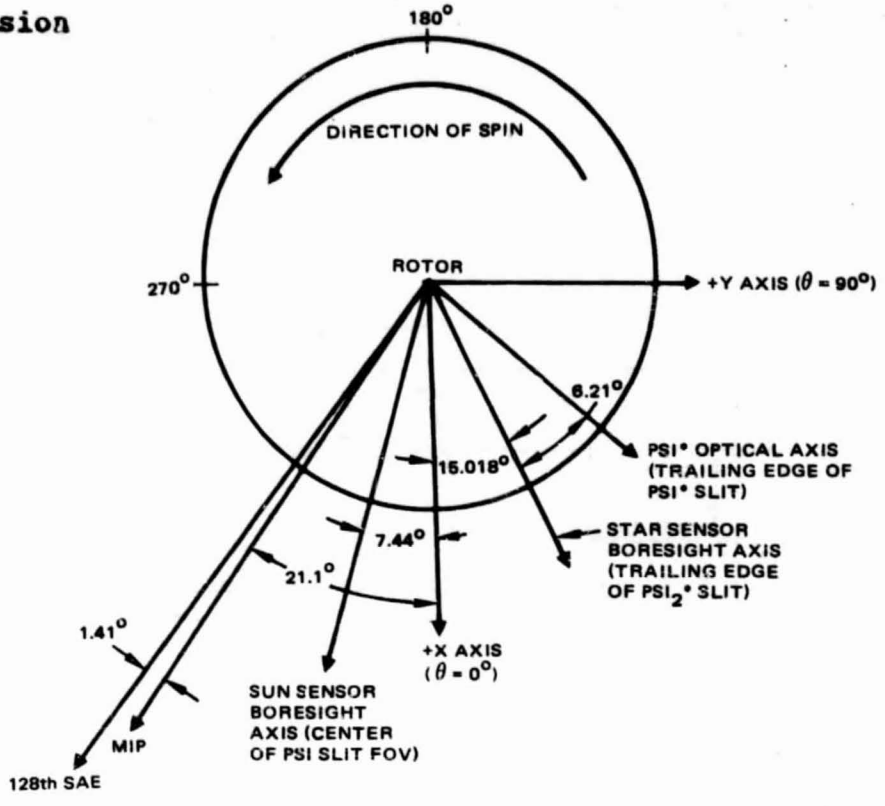
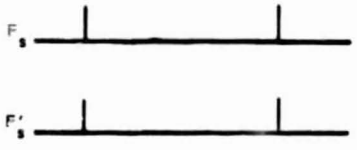
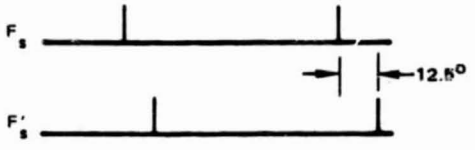


FIGURE 3.3.2.4-2. DESPIN ERROR FORMATION

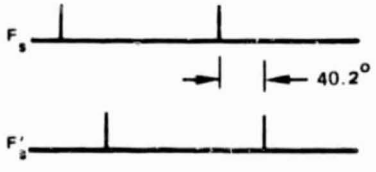




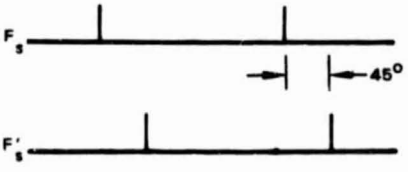
a) WITH 0 ANGLE OFFSET VALUE CMD, AND SUN AS SRR, ANTENNA BORESIGHT AXIS POINTS ALONG 128th SAE LOCATION. (ψ TO MIP) TELEMETERED TIME MEASUREMENT WILL TRANSLATE IN ANGULAR DEGREES TO READ +1.41°



b) WITH 12.5° ANGLE OFFSET VALUE CMD ANTENNA POINTS AT SUN LOS WITH SUN AS SRR

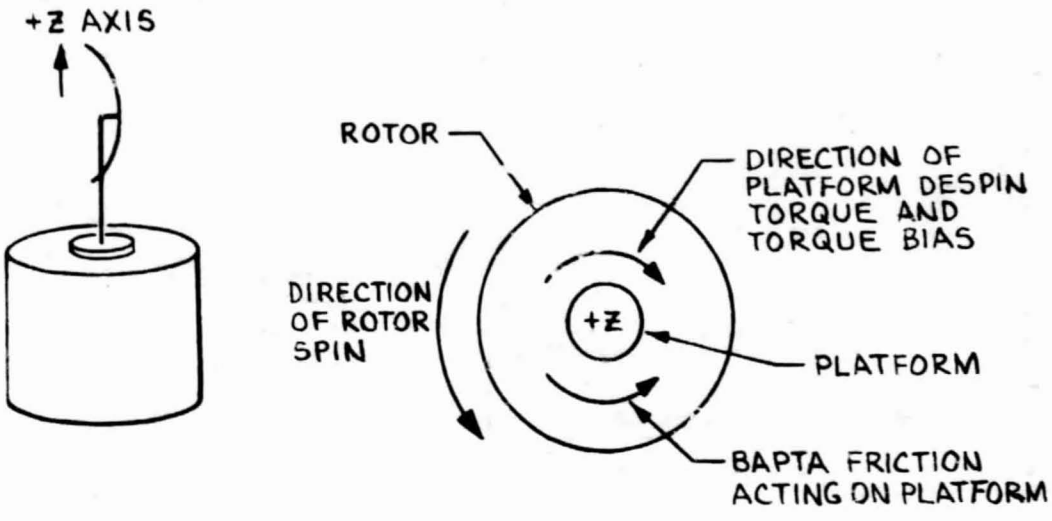


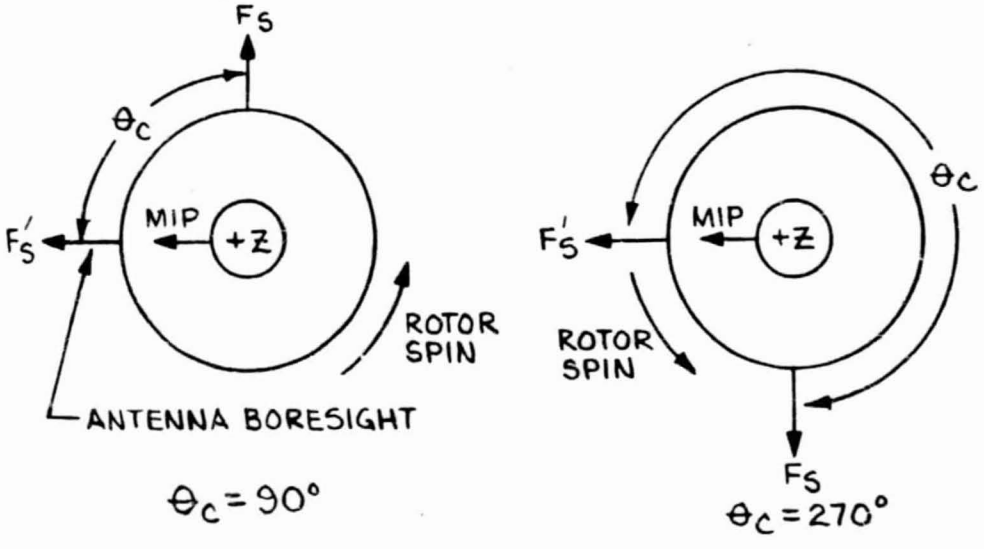
c) WITH 40.2° ANGLE OFFSET VALUE CMD ANTENNA POINTS AT SELECTED STAR REFERENCE, USING THAT STAR IN PSI* SLIT AS SRR



d) WITH 45° ANGLE OFFSET VALUE CMD ANTENNA POINTS 32.5° PAST SUN LINE, MEASURED IN DIRECTION OF SPACECRAFT SPIN, WITH SUN AS SRR

FIGURE 3.3 2.4-3. SUN/STAR SENSOR GEOMETRY





SPACECRAFT ANGLE INCREASES IN A COUNTERCLOCKWISE DIRECTION (DIRECTION OF ROTOR SPIN) WITH RESPECT TO THE F_S REFERENCE

Figure 3.3.2.4-4. DCE/Spacecraft Polarity Conventions

Section No. 3.3.4
Doc. No. PC-402
Orig. Issue Date 2/28/78
Revision No. _____

Revision

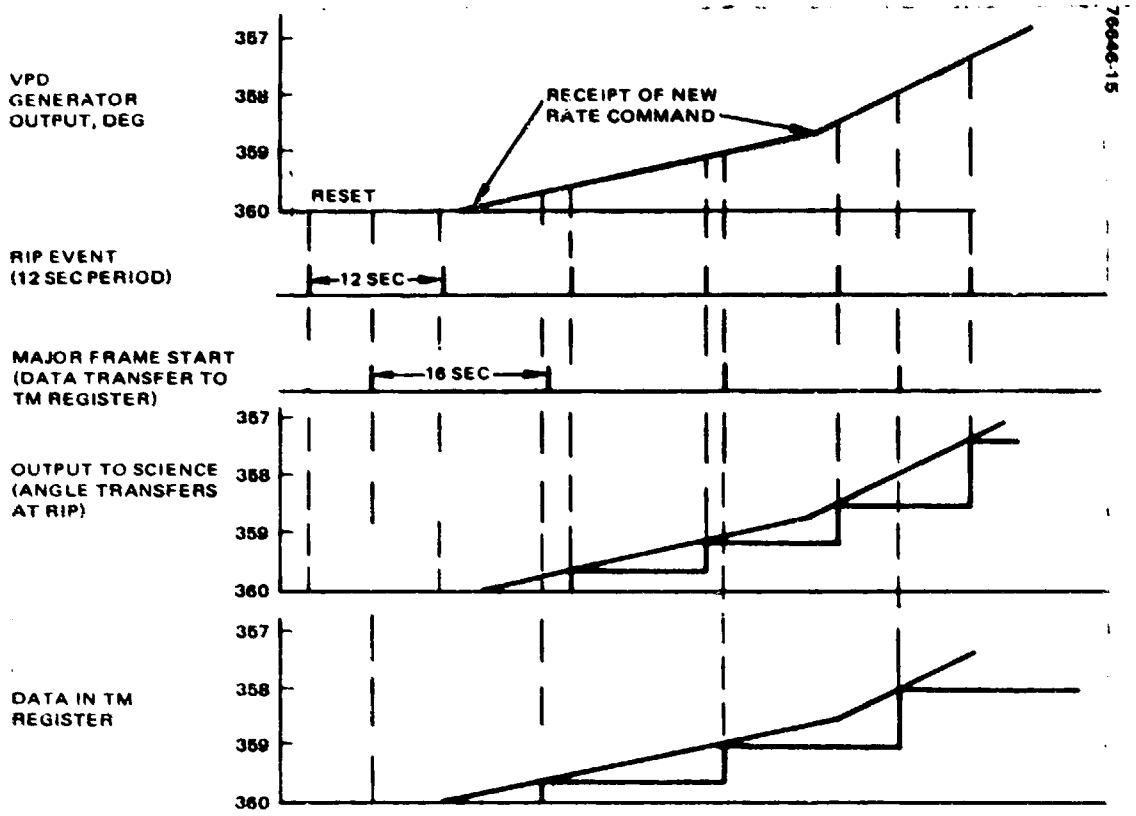


FIGURE 3.3.2.4-5. RAM (NADIR) OUTPUT RELATIONSHIPS

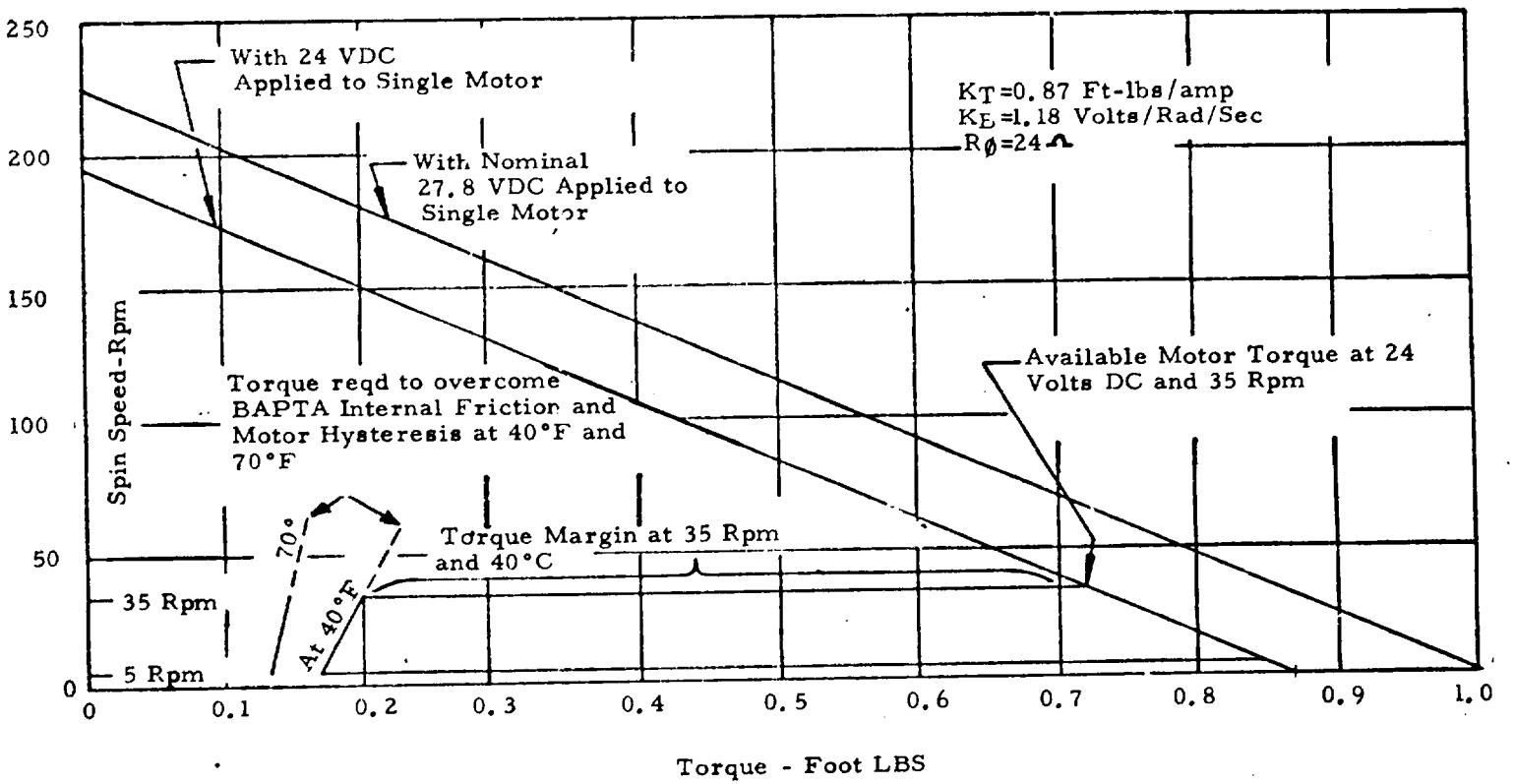
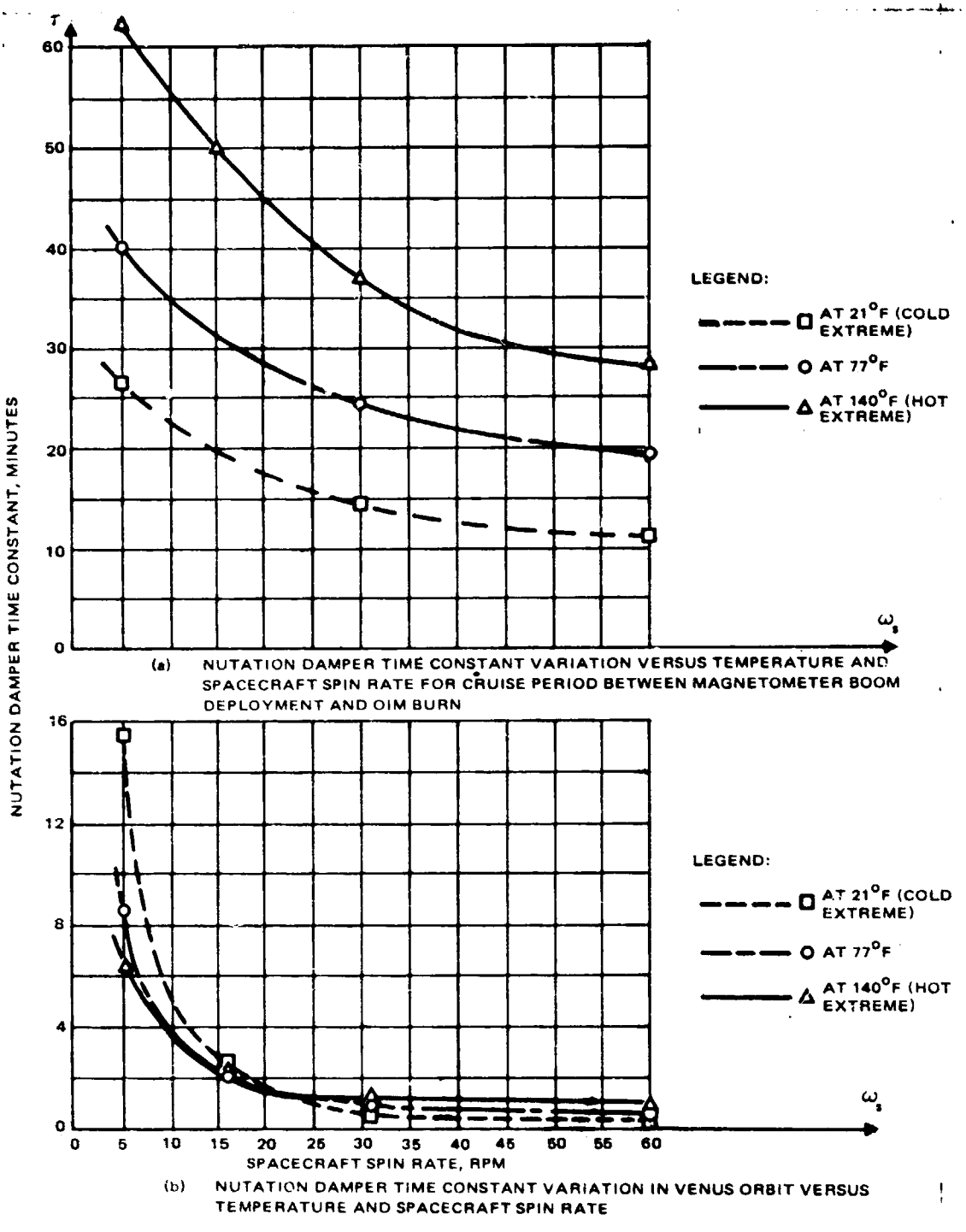


Figure 3.3.2.5-1. BAPTA Torque Capability (Foot Pounds).



(a) NUTATION DAMPER TIME CONSTANT VARIATION VERSUS TEMPERATURE AND SPACECRAFT SPIN RATE FOR CRUISE PERIOD BETWEEN MAGNETOMETER BOOM DEPLOYMENT AND OIM BURN

(b) NUTATION DAMPER TIME CONSTANT VARIATION IN VENUS ORBIT VERSUS TEMPERATURE AND SPACECRAFT SPIN RATE

FIGURE 3.3.2.6-1. NUTATION DAMPER TIME CONSTANT VARIATION.

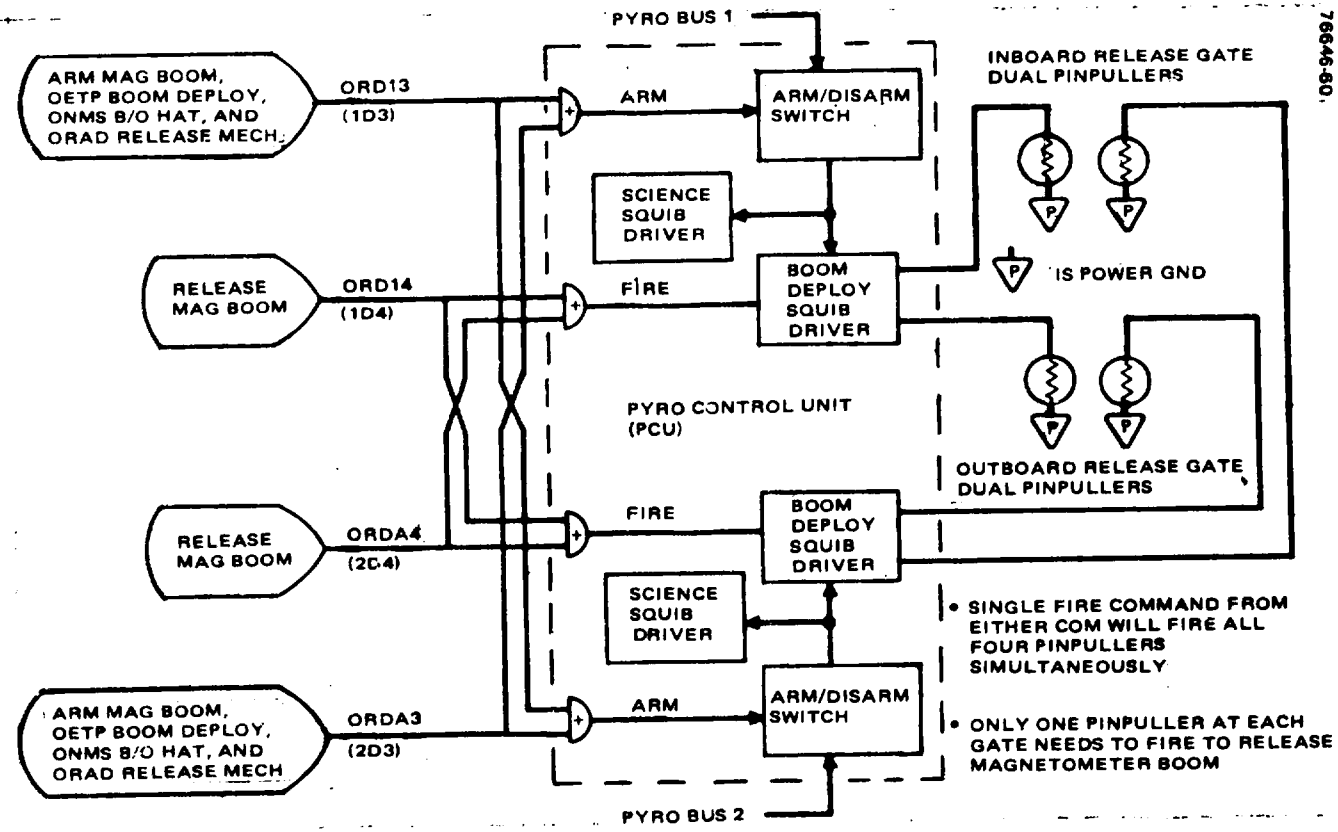


FIGURE 3.3.2.7-1. MAGNETOMETER DEPLOYMENT MECHANISM (PINPULLER RELEASE INTERFACE)

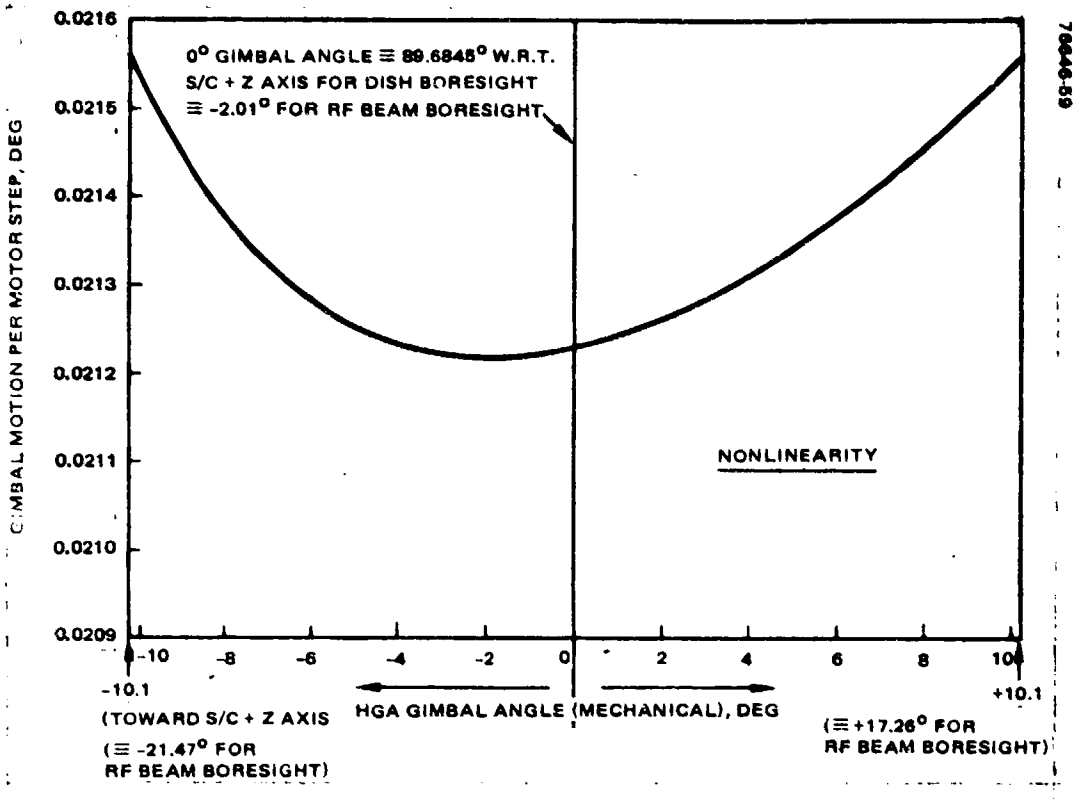


FIGURE 3.3.2.8-1. HGA GIMBAL STEP SIZE VARIATION

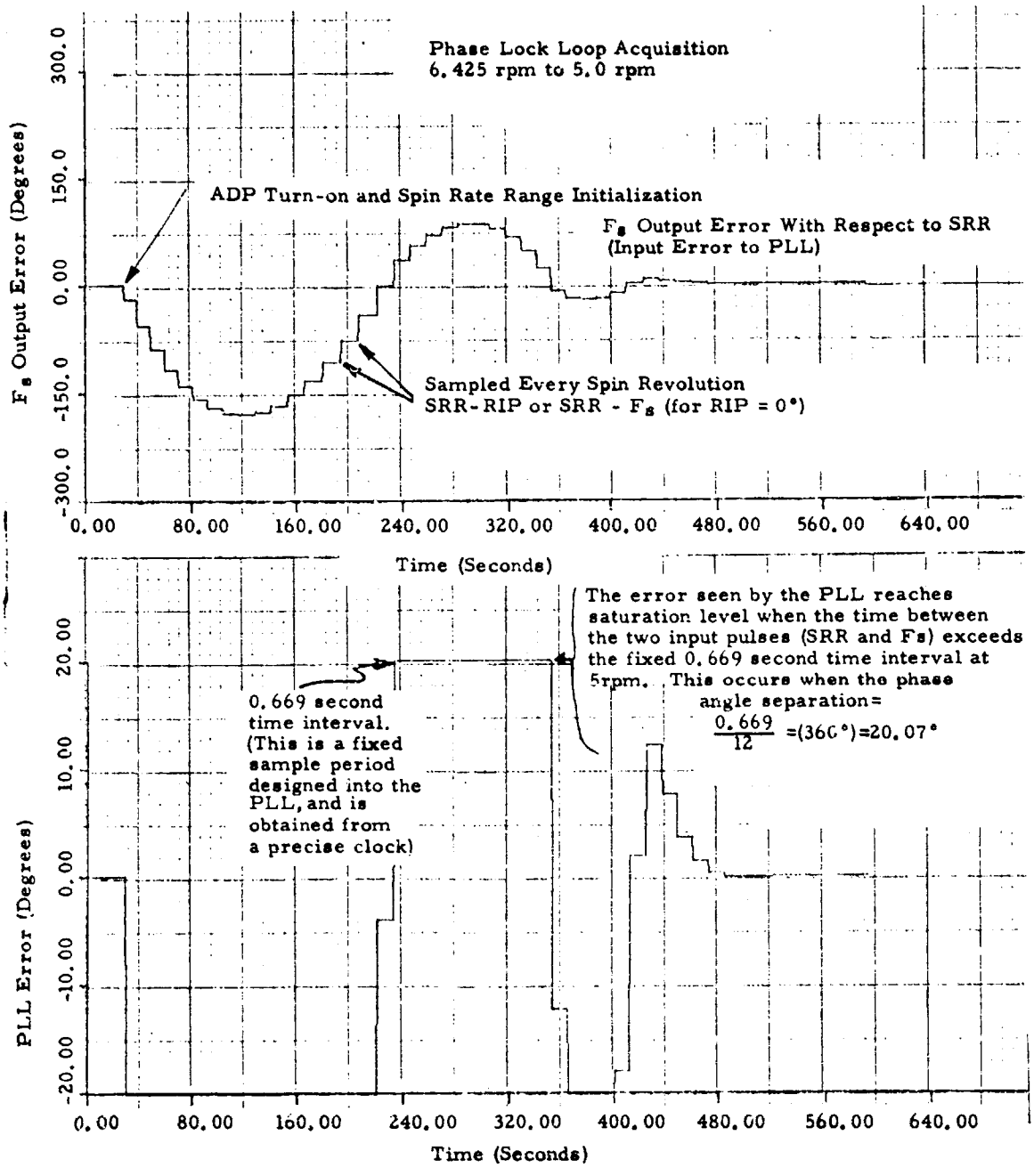


Figure 3.3.3.1-1. Phase Lock Loop Acquisition (5 rpm)

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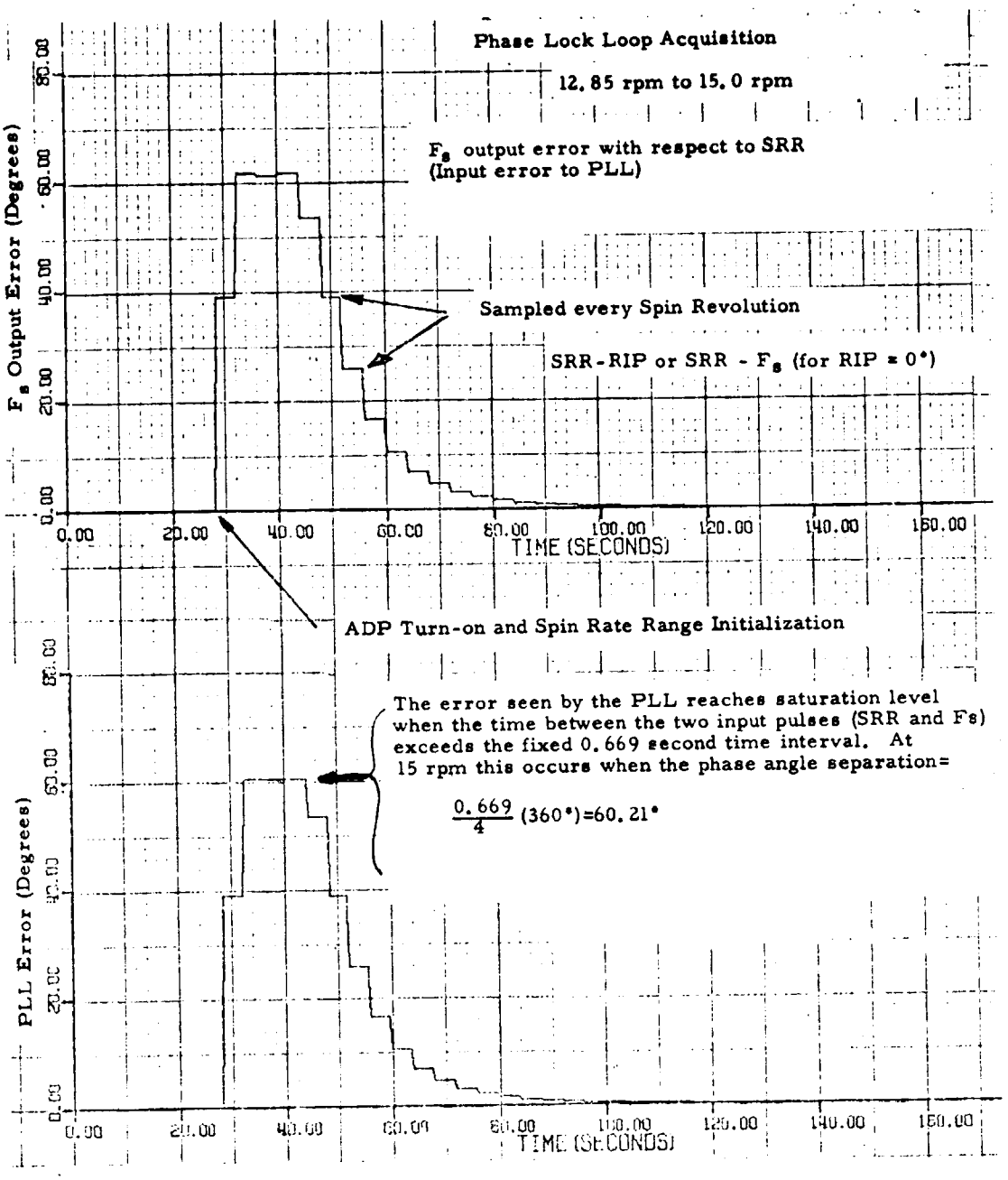
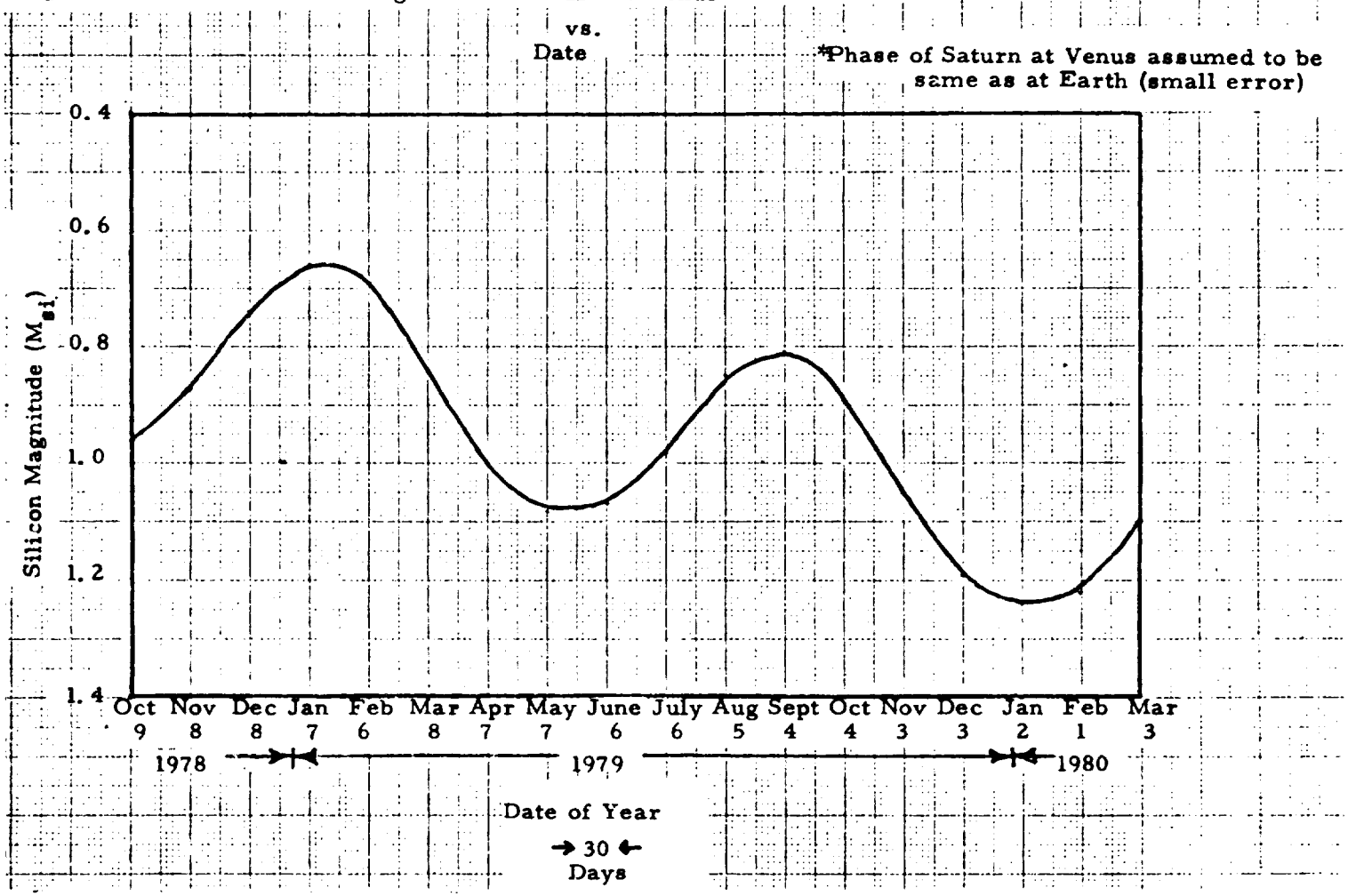


Figure 3.3.3.1-2. Phase Lock Loop Acquisition (15 rpm)

Fioneer Venus Star Sensor (Flight Unit No. 2)

BBRC S/N 4
Hughes S/N 3

Figure 3.3.3.3-1. Silicon Magnitude of Saturn at Venus *



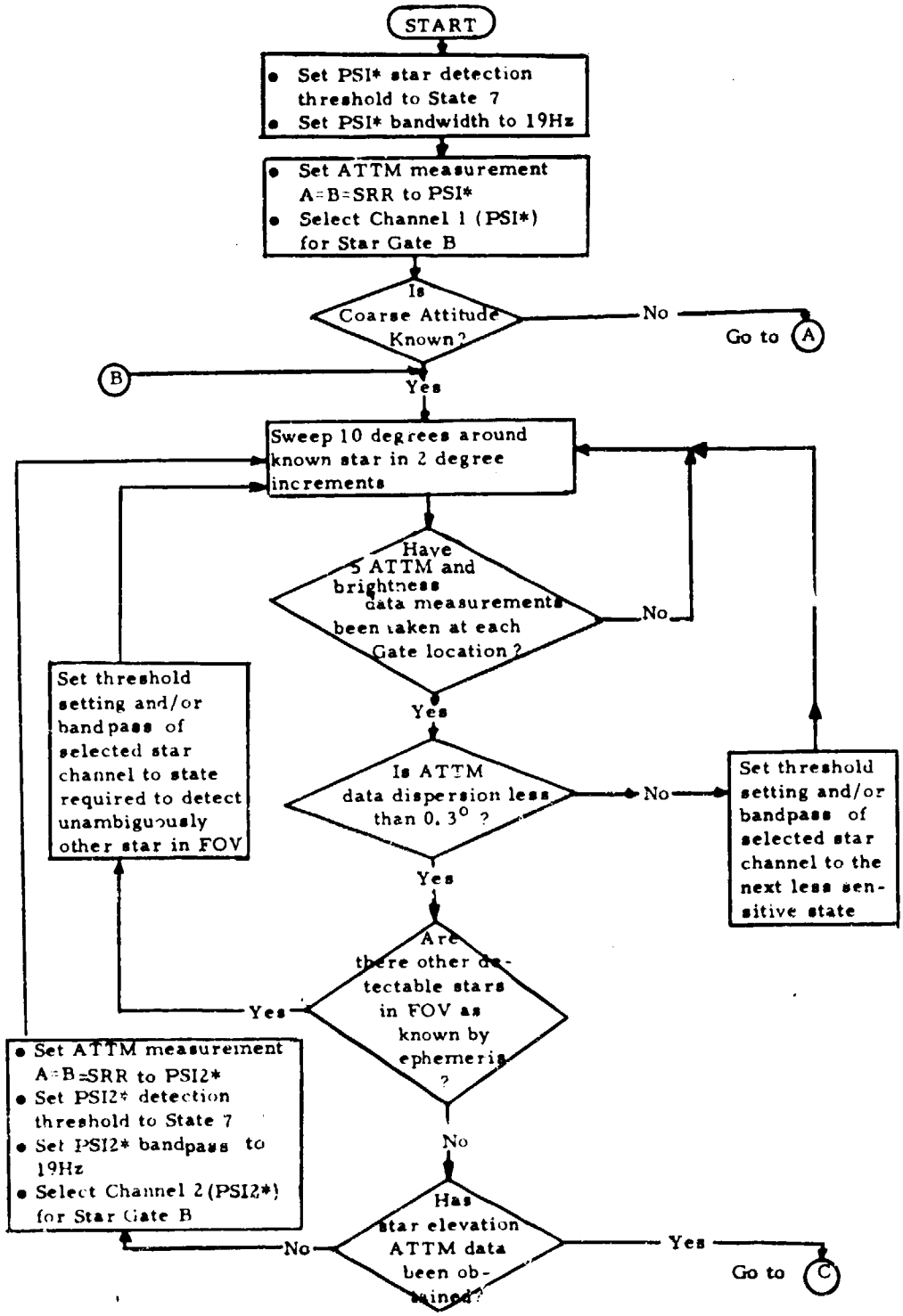


Figure 3.3.3.3-2. Star Mapping Flow Chart

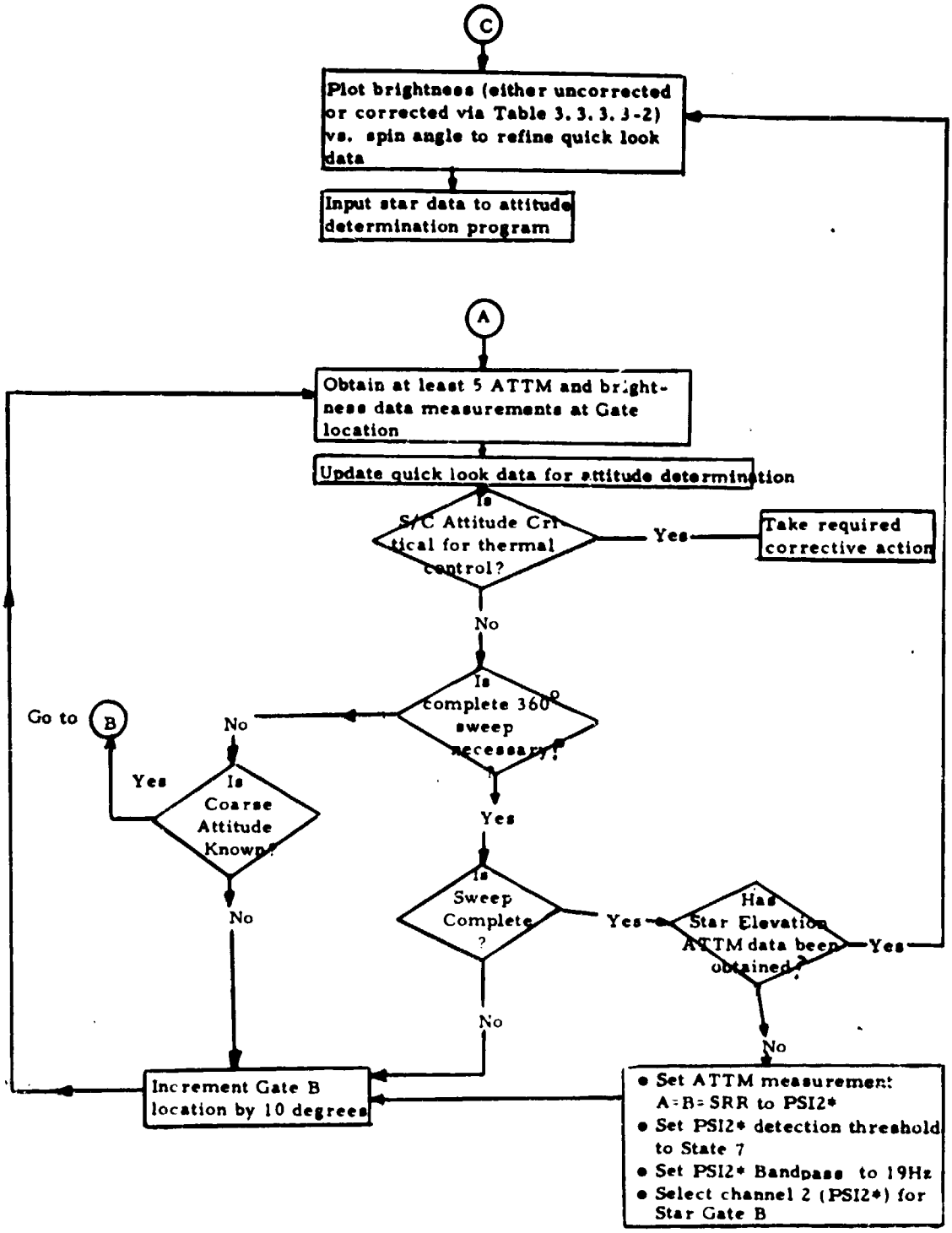


Figure 3.3.3.3-2. Star Mapping Flow Chart (Continued)

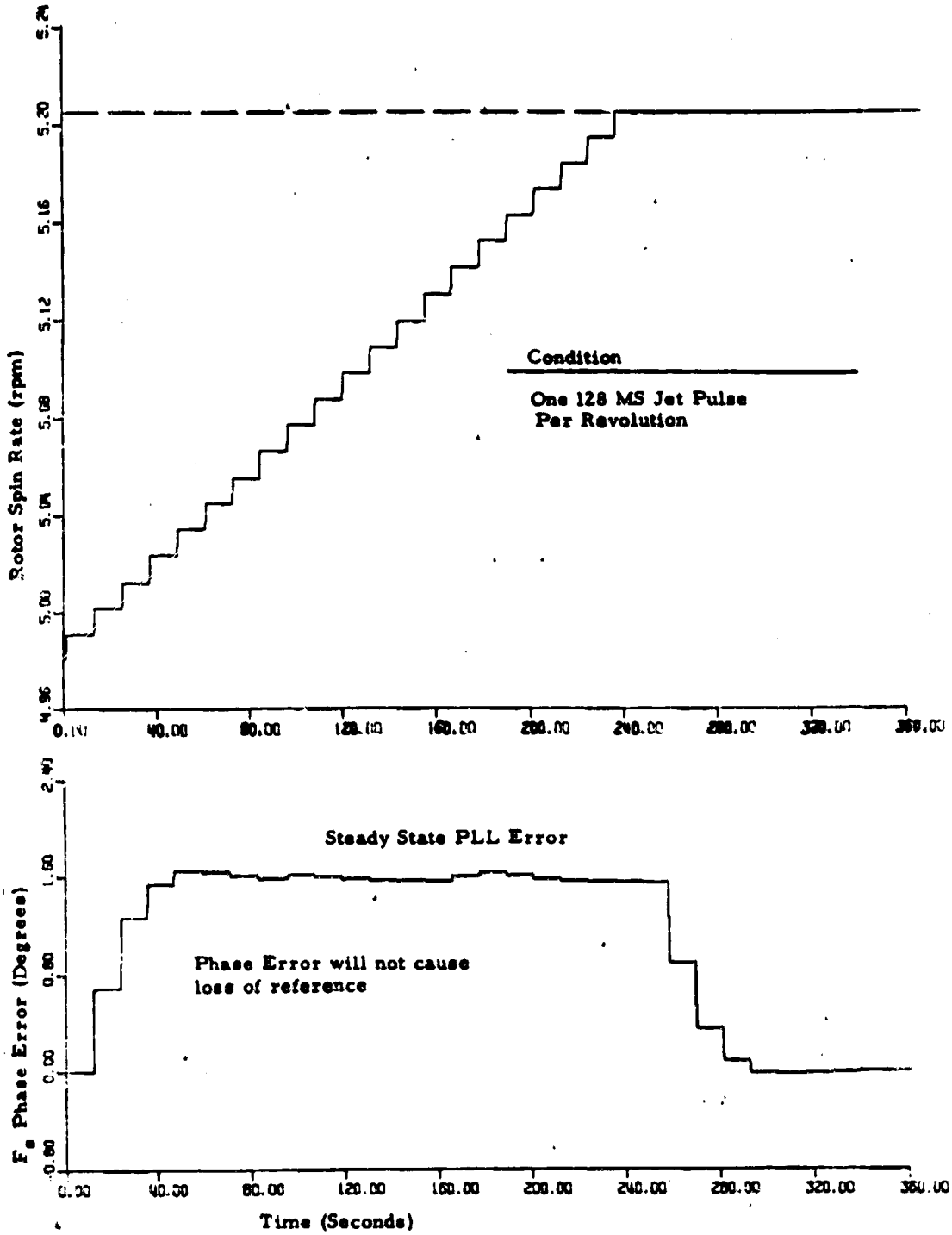


Figure 3.3.3.5-1. F_s Phase Error During Single 128 MS Pulse Per Revolution

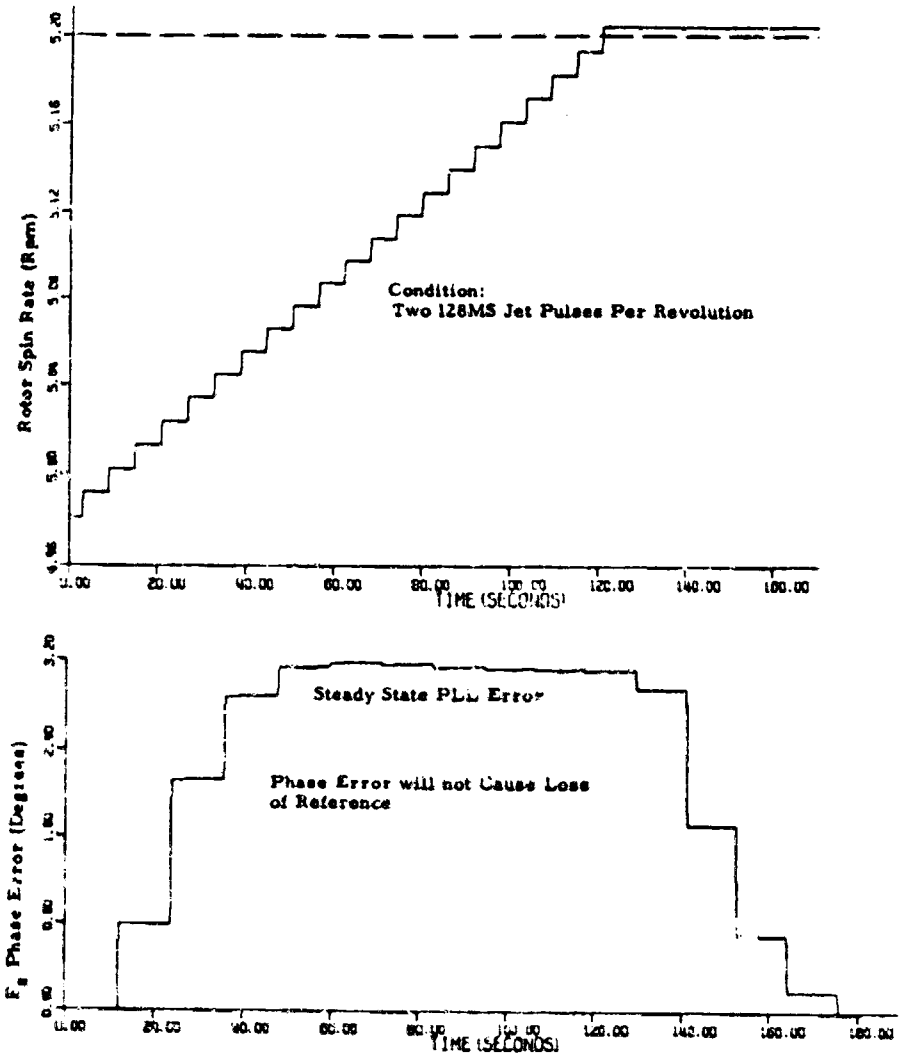


Figure 3.3.3.5-2. F_s Phase Error During Two 128 MS Pulses Per Revolution

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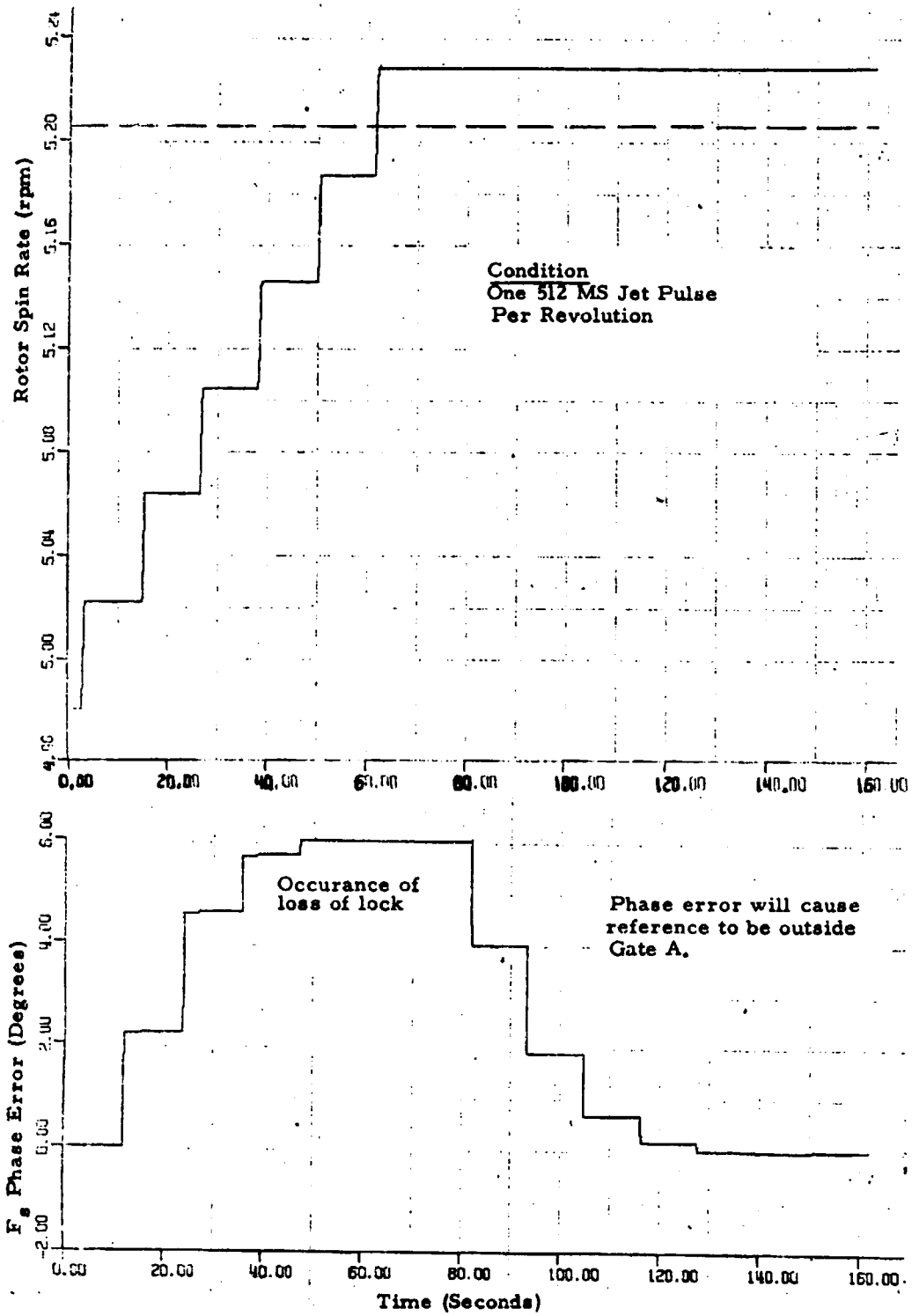


Figure 3.3.3.5-3. F_s Phase Error During Single 512 MS Pulse Per Revolution

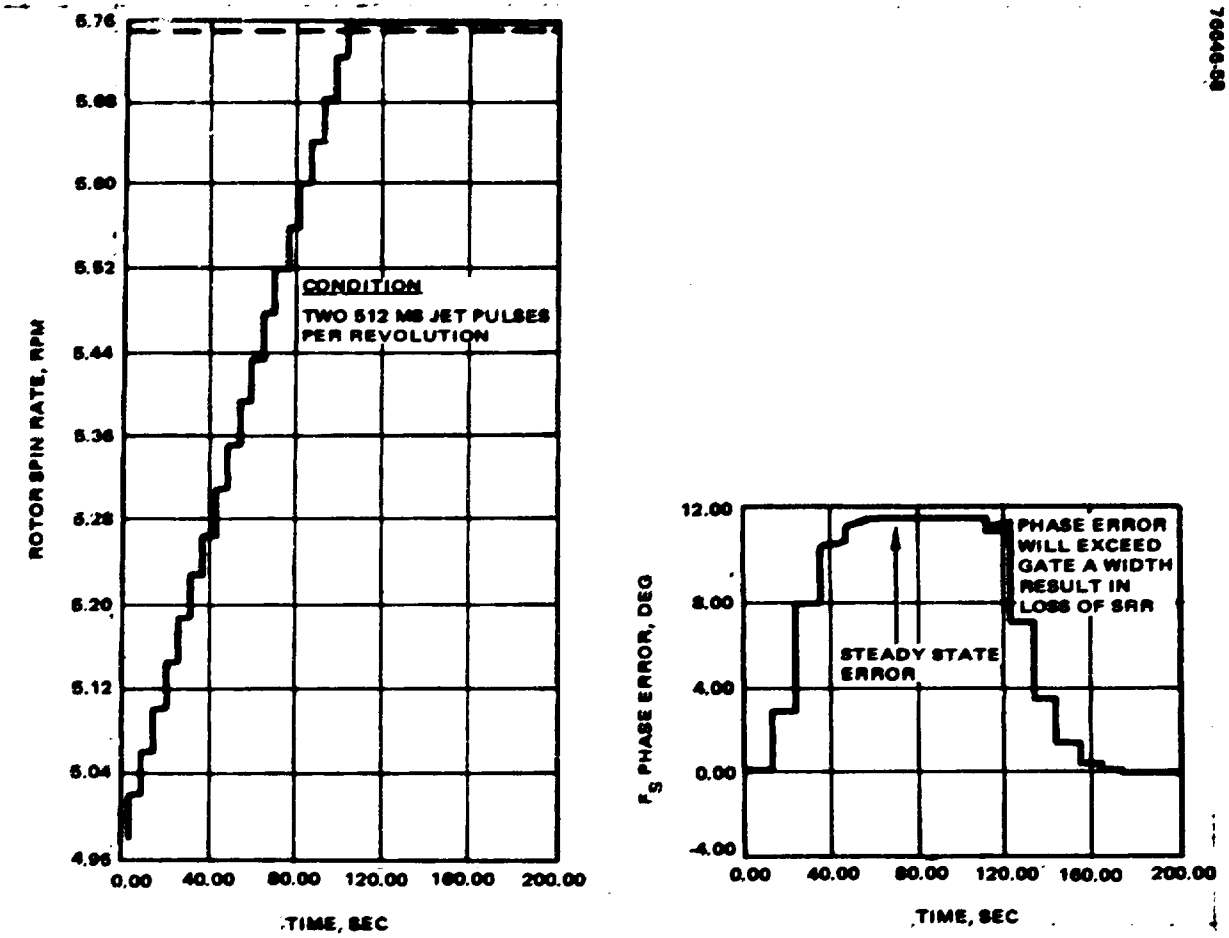


FIGURE 3.3.3.5-4. F_S PHASE ERROR DURING TWO 512 MS PULSES PER REVOLUTION

MEASUREMENT A: SRR TO SRR

MEASUREMENT B: SRR TO MINOR FRAME START

TIME HISTORY

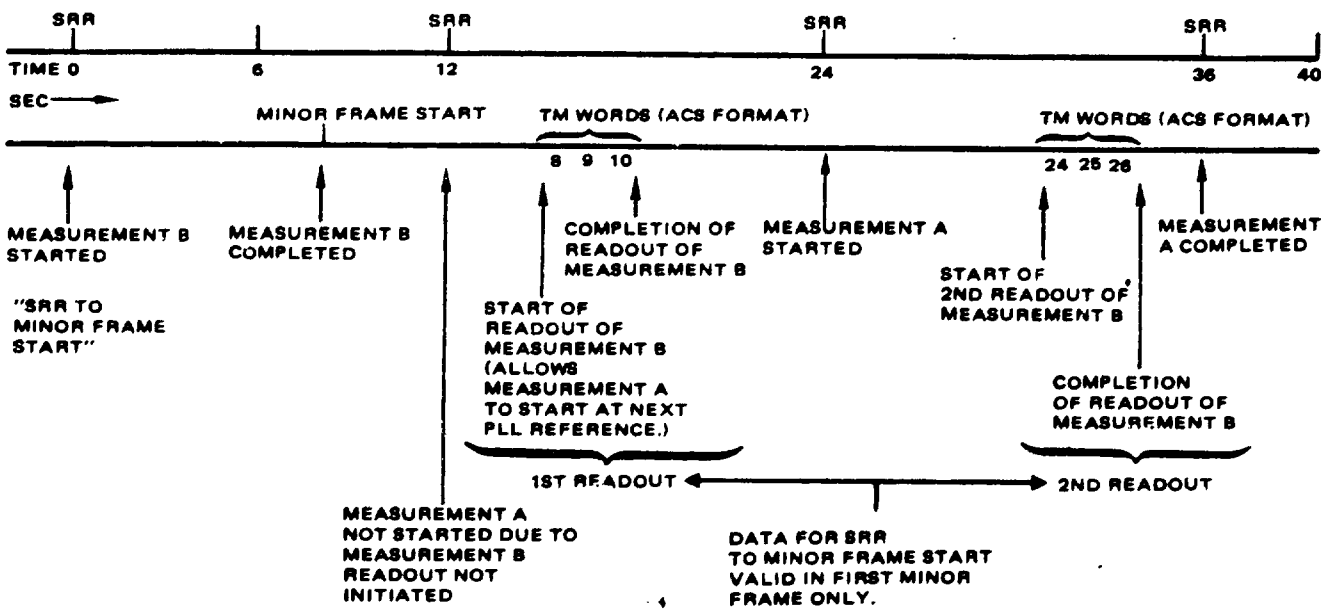
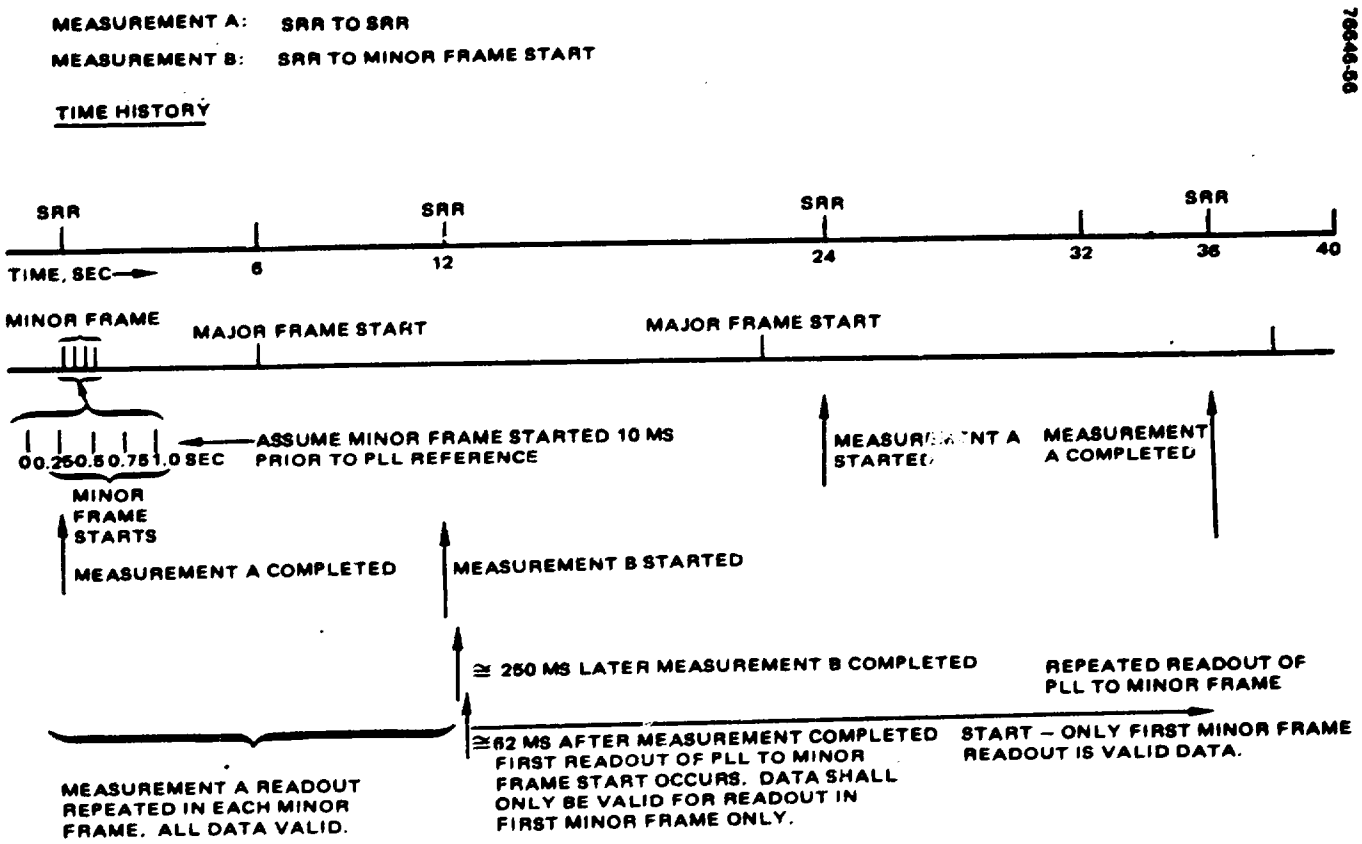


FIGURE 3.3.3.6-1. ATTM TELEMETRY SAMPLED AT 8 BPS IN MINOR FRAME MODE (ACS FORMAT) AT 5 RPM



3.3.3.6-2. ATTM TELEMETRY SAMPLED AT 2048 BPS IN MINOR FRAME MODE (ACS FORMAT) AT 5 RPM

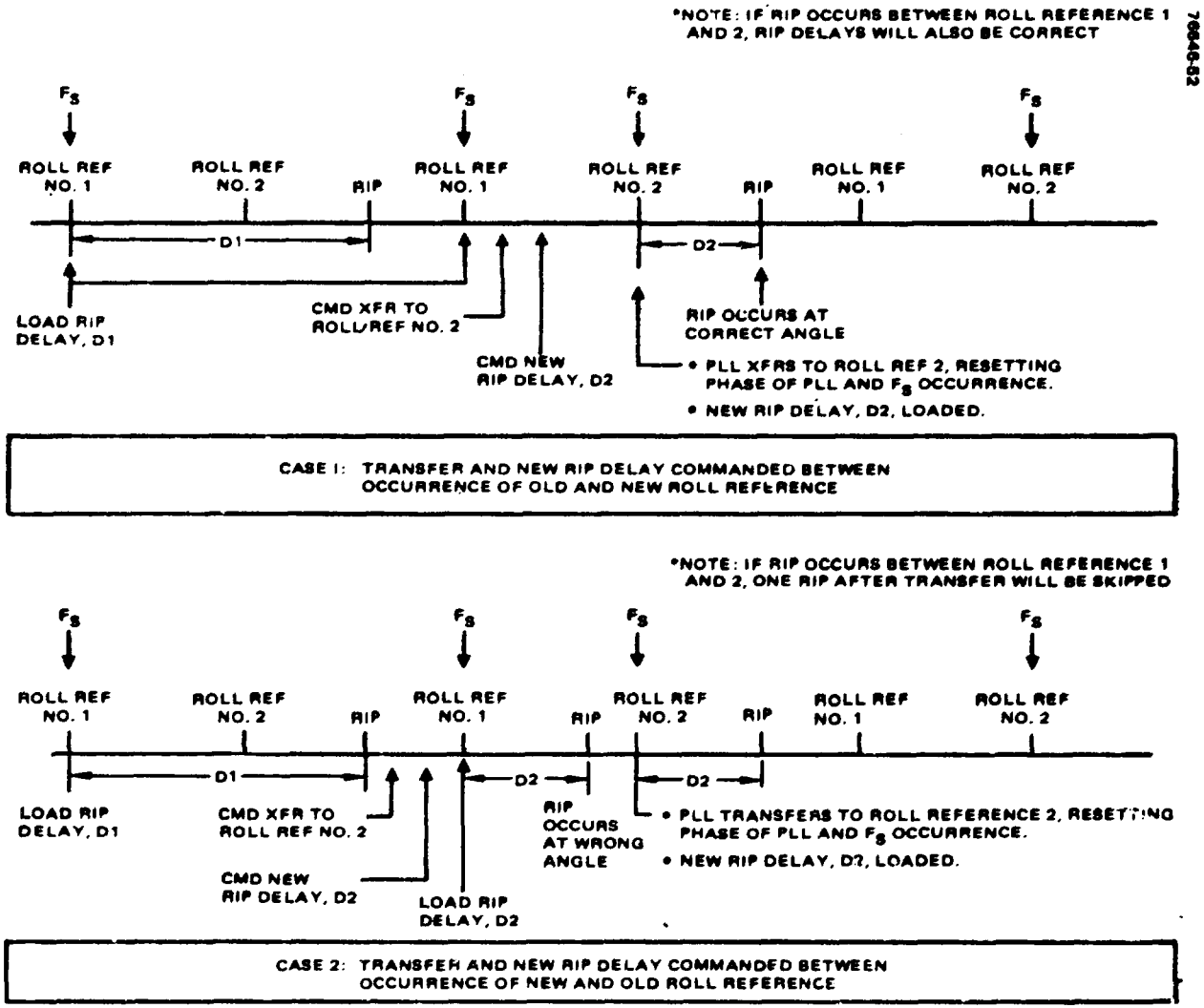


FIGURE 3.3.3.9-1. RIP GENERATION AT ROLL REFERENCE TRANSFER

3.4 PROPULSION SUBSYSTEM

3.4.1 Subsystem Description. The propulsion subsystem provides the propulsive capability to perform all spin rate, velocity and attitude changes required of the Orbiter during its mission. These maneuvers are performed by electrically opening the appropriate latch valves and thruster valves and thus allowing pressurized hydrazine, a monopropellant, to flow to the thruster where it is catalytically decomposed to produce thrust. The subsystem operates in the blowdown mode; i.e., the subsystem volume is constant and the pressure is unregulated so that the subsystem pressure drops as propellant is used. Helium is used as the pressurant gas.

The propulsion subsystem, shown schematically in Figure 3.4.1-1 consists of the following units and components:

- 2 Tanks for propellant storage
- 7 Thruster assemblies for providing thrust (1 lb_f nominal) in specific directions
- 2 Latch valves to isolate the thrusters into two redundant groups
- 4 Filters to prevent thruster valve contamination
- 2 Reservoirs for initial spin-up propellant storage
- 3 Fill/drain valves for subsystem loading and pressurization prior to launch
- 1 Pressure transducer to measure and telemeter subsystem pressure
- Propellant lines to interconnect the various components
- Heaters to maintain propellant temperature greater than 40°F (5° above freezing)
- 13 Temperature sensors to measure and telemeter subsystem temperature (plus 2 on the OIM and 1 on the S&A).

The seven thrusters are separated into two fully redundant groups by the latch valves. Although most normal maneuvers use thrusters from both groups simultaneously, all maneuvers can be performed with only the thrusters in either

group. Each thruster group consists of two radial thrusters and either one or two axial thrusters; thrusters R1, R2 and A6 are in one group, and thrusters R3, R4, A5 and A7 are in the other. The two radial thrusters in each group are aligned so that their thrust vectors are parallel to each other. The plane containing these two thrust vectors is canted in order to pass through the spacecraft center of mass. The radial thrusters of the two different groups are canted through different points corresponding to the spacecraft center of mass during different phases of the mission. For the Orbiter, radial thrusters R3 and R4 are canted through the Orbiter center of meass at the time of the first trajectory correction maneuver while radial thrusters R1 and R2 are canted through the average on orbit center of mass. This design is dictated by the large shift in center of mass due to OIM propellant usage and the requirement to minimize cross coupling. Table 3.4.1-1 and Figure 3.4.1-2 give the thruster locations, thrust directions, moment arms and general thruster uses.

The physical layout of all subsystem units and components near the equipment shelf is shown in Figure 3.4.1-3.

3.4.2 Units Descriptions

3.4.2.1 Thruster Assembly. The thruster assembly shown in Figure 3.4.2.1-1 consists of a thruster valve to control propellant flow and a thrust chamber with catalyst beds and an expansion nozzle. A cross sectional view of the valve is shown in Figure 3.4.2.1-2.

Valve operation is controlled by application of electrical power from the valve drivers in the ADP of the controls subsystem to redundant coils actuating a single torque motor. The opening and closing response times are 15 ±1 ms and 10 ±1 ms, respectively. The valve is normally closed and is capable of pulsed or continuous operation by controlling the duration of the power application. Command selectable pulse on times

TABLE 3.4.1-1

ORBITER THRUSTER GEOMETRY

Thruster	Location	Thrust/Plume Direction (Direction of S/C Motion is Opposite)	Usage
R1	$\theta = 82^{\circ}$ STA = 68.75	$\theta = 52^{\circ}30'$ Canted 4.02° Aft	+ Spin; ΔV & Precession Post OIM Fire
R2	$\theta = 22^{\circ}$ STA = 68.75	$\theta = 52^{\circ}30'$ Canted 4.02° Aft	- Spin; ΔV & Precession Post OIM Fire
R3	$\theta = 262^{\circ}$ STA = 68.75	$\theta = 232^{\circ}30'$ Canted 4.46° Forward	+ Spin; ΔV & Precession Cruise
R4	$\theta = 202^{\circ}$ STA = 68.75	$\theta = 232^{\circ}30'$ Canted 4.46° Forward	- Spin; ΔV & Precession Cruise
A5	$\theta = 105^{\circ}$ R = 46 in.	Forward From Sta 82.4	ΔV; Precession
A6	$\theta = 285^{\circ}$ R = 42 in.	Aft From Sta 63.5	ΔV; Precession
A7	$\theta = 105^{\circ}$ R = 42 in.	Aft From Sta 63.5	ΔV; Precession

of 128 ms and 512 ms are used for Pioneer Venus. The peak power required is 21 watts per valve.

Series redundant propellant shutoff is provided by utilizing dual in-line tungsten carbide poppets and seats. The downstream poppet is actuated by a flapper which is an integral part of the armature; the upstream poppet is mechanically opened by a pin connected to the flapper. A 0.004 inch gap between the pin and the upstream poppet permits positive leak free propellant shutoff by one poppet even if the other poppet is held open by contamination. The interstage cavity between the two valve seats is 0.68 cc; hydrazine trapped in this volume can be subjected to increasing pressure. If so, it relieves back into the system across the upstream poppet which acts as a relief valve. The valve has a cycle life of 350,000 cycles per formal qualification tests; however, Hughes has placed well over one million cycles on a number of the valves without performance degradation. The leakage rate allowable per seat is 2 scc GN_2 per hour for a new valve and 3 scc GN_2 per hour after exposure to hydrazine. Based on analysis and test (Refer to Hughes' IDC 4115.10.2/742, dated 30 September 1976), a GN_2 leak rate of 2 scc per hour is equivalent to less than 10^{-2} gm per year of hydrazine using Knudsen scaling. If effects of kinematic viscosity are considered for a gas leak rate of 2 scc per hour, then the equivalent hydrazine leakage is equal to or less than

$$10^{-1} \frac{\text{gm/yr}}{\text{scc/hr}}$$

The possibility of a closed failure of this valve is remote. The design of the seat and poppet precludes mechanical interference. The independently wound electrical coils are fully redundant in that either coil is capable of opening the valve in case of failure of the other.

The thrust chamber, the second part of the thruster assembly, is a completely mechanical

component that consists of a catalyst chamber where the propellant is decomposed and an expansion nozzle through which the hot gasses are expanded. The thrust chamber is constructed chiefly of cobalt base alloys capable of withstanding the high temperatures associated with hydrazine decomposition. The nozzle has a removable conical extension added to it so that the thruster plume clears the solar panel substrate. With this extension, the nozzle expansion ratio is increased from its normal 40 to 1 ratio to 563 to 1.

Hydrazine flow through the thruster is controlled by a trim orifice. The required orifice size for a given thruster is determined during thruster assembly where water flow tests are made of the injector and candidate orifices. Computation for the orifice pressure drop required to produce one pound of thrust at 200 psia supply pressure takes into account the chamber pressure, the injector and inlet tube assembly pressure drop, the catalyst bed pressure drop and the throat diameter of the nozzle assigned to the thruster. Should hot firing the thruster reveal performance is out of limits, the thruster is re-orificed until the desired performance is achieved.

During the qualification tests of the thruster for Telesat, Canada, 700 cold starts were demonstrated. Presented below is the relation between the mass flow per pulse at high and low supply pressure:

P_s, Psia	On-Time, Ms	Per Pulse Flow, Lb.
300	128	0.00076
300	512	0.00317
75	128	0.00029
75	512	0.00117

Thruster performance varies as a function of propellant supply pressure and, when in the pulse mode, duty cycle and cumulative number of pulses. Performance of the Orbiter thrusters is shown in

Figure 3.4.2.1-3, Figure 3.4.2.1-4, and in Table 3.4.2.1-1.

3.4.2.2 Latch Valve. The two bistable latch valves are used to ensure that propellant is available in each reservoir for spacecraft spinup and to seal off the two redundant thruster groups. The latch valve changes position (open or closed) upon receipt of the appropriate electrical control signal from the ADP of the controls subsystem. The valve contains a single hard seat with an elastomeric poppet. Permanent magnets are used to latch the valve in the desired position after removal of the control signal from either the opening or closing coils of the torque motor. Allowable leakage is 5.0 scc per hour GN_2. Moving parts are isolated from propellant contact by a welded metal torque tube. The valve has been qualified to a cycle life of 1,000 cycles. A cross-sectional view of the valve is shown in Figure 3.4.2.2-1.

The position of the latch valve is indicated by a telemetry status bit (VALV1S and VALV2S) with a "1" indicating the open position.

3.4.2.3 Propellant Supply Components and Units

3.4.2.3.1 Propellant Tank. Each titanium propellant tank consists of a 12.8 inch diameter hemisphere welded to an 86° cone (Figure 3.4.2.3.1-1) and is designed for an operating pressure of 400 psia with a safety factor of four.

Each tank has two outlet ports, with the propellant outlet at the apex of the conical section, and the second part, diametrically opposite the propellant outlet, used to equalize the gas pressure with the other tank in the subsystem. Total volume of the propulsion subsystem is 2265 in^3.

The propellant/pressurant interface within the tank is determined by the centrifugal forces associated with spacecraft rotation during the mission. Complete drainage of the tank is assured by the orientation of the outlet.

TABLE 3.4.2.1-1.
CUMULATIVE TOTAL IMPULSE TABLE

Pulse No.	Cumulative Total Impulse (Lb.-Sec.) at Indicated Duty Cycle, RPM, And Supply Pressure							
Duty Cycle	128/3872	512/3488	128/1872	128/3872	512/3488	128/872	128/11872	512/11488
RPM	15	15	30	15	15	60	5	5
P_a, PSIA	300	300	225	200	200	200	75	75
1	0.08	0.47	0.06	0.06	0.34	0.05	0.03	0.16
2	0.19	1.05	0.14	0.14	0.77	0.13	0.08	0.35
3	0.31	1.66	0.24	0.23	1.21	0.21	0.12	0.55
4	0.44	2.29	0.34	0.32	1.67	0.29	0.16	0.76
5	0.57	2.94	0.45	0.42	2.15	0.39	0.20	0.97
6	0.70	3.59	0.57	0.52	2.63	0.49	0.25	1.19
7	0.84	4.24	0.69	0.62	3.12	0.59	0.29	1.41
8	0.98	4.90	0.82	0.72	3.61	0.69	0.34	1.63
9	1.13	5.56	0.94	0.83	4.11	0.80	0.38	1.86
10	1.27	6.22	1.07	0.94	4.60	0.91	0.43	2.09
15	2.02	9.56	1.72	1.50	7.09	1.46	0.66	3.23
20	2.80	12.91	2.40	2.09	9.61	2.03	0.91	4.37
30	4.39	19.63	3.78	3.29	14.64	3.22	1.41	6.65
50	7.63	33.08	6.62	5.73	24.70	5.65	2.44	11.13
100	15.76	66.82	13.78	11.84	49.93	11.80	4.94	21.76

The nominal tank volume is 1100 cubic inches and may be used in the blowdown mode with up to 35 pounds of propellant per tank. Actual volume of each tank is measured during acceptance test by closely measuring the amount of water it holds at room ambient conditions.

3.4.2.3.2 Propellant Lines. The propellant lines are 0.25 inch O.D. by 0.020 wall titanium tubing. With a minimum burst strength of 25,000 psi and all welded joints, the chance of leakage or breakage of the propellant lines is minimal.

3.4.2.3.3 Propellant Reservoir. The propellant reservoir is a 7.1 cubic inch titanium cylinder that is used to ensure that sufficient propellant is available for initial spin up even if the propellant in the tank has settled away from the tank outlet during zero g coast. One reservoir is used in the feed line for each group of thrusters. After initial spin up, the reservoir serves as part of the propellant line system from the tanks to the thrusters.

3.4.2.3.4 Filters. Four filters, two for each thruster group, are provided to ensure that particulate contamination does not reach the thruster valves where it could cause a failed open condition. The filter, with a 10 micron absolute filter rating is a passive device consisting of electrochemically etched discs mounted concentrically inside the housing. Contaminating particles are trapped and held in the discs.

3.4.2.3.5 Fill/Drain Valves. Three fill/drain valves are provided for ground servicing of the propulsion subsystem. Two valves are used for propellant loading and off loading and one for pressurant loading and off loading. The fill/drain valves, which feature triple redundant seals are closed prior to spacecraft encapsulation and launch. During the mission they serve only as terminations for the lines.

3.4.2.4 Heaters. All elements of the propulsion subsystem except the semicircular run of pressurant line are heated to maintain the

propellant temperature at 40°F or higher. The mechanically redundant components and units which are located downstream of the latch valves have non-redundant heaters while the non-redundant components and units upstream of the latch valves have redundant heaters. The OIM and S&A heaters are redundant and are designed to maintain a minimum temperature of 26° (near earth) and a nominal operating temperature of 35° (at time of Orbit Insertion). The heater design and oeprational characteristics are given in Section 3.2, Thermal Design.

3.4.2.5 Orbit Insertion Motor Assembly

3.4.2.5.1 Orbit Insertion Motor. The orbit insertion motor (OIM) provides the retarding impulse necessary to place the Orbiter into Venus orbit. The OIM, shown in Figure 3.4.2.5.1-1, is a Thiokol TE-M-604 motor with a 24-inch diameter spherical internally insulated titanium alloy case, a 36.7/1 expansion ratio nozzle with a tape wrapped carbon phenolic exit case, a head end mounted PYROGEN igniter, a scalloped eight point star grain of TP-H-3062 propellant and a nozzle throat seal to prevent propellant outgassing by pressurizing the motor to one atmosphere with GN_2. The OIM is mounted inside the open thrust tube with its axis of symmetry along the spacecraft spin axis and the nozzle pointing in the aft (-z) direction.

The OIM is sized to impart a velocity of 3475.42 ft/sec (1059.31 m/sec) to a spacecraft weighing 1250.3 pounds at ignition. The motor is loaded with 394.0 pounds of propellant that has a specific impulse of 286.6 lb_f-sec/lb m. Approximately four pounds of inert material is expended with the propellant which lowers the effective specific impulse to 281.93 lb_f-sec/lb m. The total impulse of 112,191 lb_f-sec is controlled within 0.6%. The burn time of the OIM is approximately 30 seconds at 70°F; burn time decreases with increased temperature.

The OIM is equipped with two commandable redundant heaters to maintain motor temperature

within 26°F to 90°F non-operating and 35°F to 90°F at ignition. There are two temperature sensors (VTHROT and VCASET) on the motor case to measure its temperature. Thermal control of the OIM is discussed more fully in Section 3.2, Thermal Control.

Since orbit insertion occurs when the spacecraft is not in contact with the ground, no telemetry can be monitored during motor firing.

3.4.2.5.2 Safe and Arm Device. The two functions of the safe and arm device (S&A) are to prevent OIM firing during ground operations and to initiate ignition of the OIM upon receipt of the fire command. A cross sectional view of the S&A is shown in Figure 3.4.2.5.2-1.

The safing function of the S&A is accomplished by mechanically and electrically interrupting the ignition train. A small dc motor inside the S&A is used for remote safing and arming from the blockhouse. The S&A can also be manually safed if there is a manfunction. The S&A is armed shortly before launch and remains in the armed condition through the mission.

Orbit insertion motor firing is initiated when the PCU of the command subsystem receives the arm and fire commands and sends a 35 ms five amp pulse to the S&A. The S&A contains two redundant detonators that fire through a crossover network to the output ports of the S&A. Since the S&A is mounted remotely, two explosive transfer assemblies (ETA) are used to connect the S&A output ports to the motor igniter. The crossover network inside the S&A permits either detonator to fire both ETAs thus giving multiple redundancy. The arming and firing of the OIM are discussed more fully in Section 3.6.4, Command Subsystem, and Section 4.2.3, Orbit Insertion Maneuver Sequence.

The physical arrangement of the S&A, ETAs and the OIM is shown in Figure 3.4.2.5.2-2.

Redundant heaters are used to maintain the S&A within the same temperature limits as the OIM. These heaters are controlled by the same commands as the OIM heaters. A temperature sensor (VSAPRT) is provided to monitor the S&A temperature.

3.4.2.6 Engineering Instrumentation

3.4.2.6.1 Pressure Transducer. The pressure transducer is a potentiometer type transducer with a nominal maximum resistance of 5120 ohms. A constant current of one milliamp is supplied to the pressure transducer by the PCM Encoder through the DIU, both of which are part of the data handling subsystem. The voltage drop across the potentiometer (0 to 5.12 vdc) is used as the analog representation of the subsystem pressure (0 to 400 psia). Refer to Appendix A for more information about the telemetry. A cross sectional view of the pressure transducer is shown in Figure 3.4.2.6.1-1.

The main uses of the pressure transducer are in determining the expected thruster performance using the performance versus supply pressure graphs presented in Section 3.4.2.1 and, along with the pressurant temperature sensors, in determining the amount of propellant remaining. The determination of remaining propellant is discussed in Section 3.4.3.2.3.

3.4.2.6.2 Temperature Sensors. Sixteen temperature sensors are used on the Orbiter propulsion subsystem and OIM. Table 3.4.2.6.2-1 lists each telemetry channel and sensor location. One temperature sensor is mounted on each of the seven thruster valve covers. These sensors measure thermal soak back during thruster firing and thus verify thruster operation in the pulse mode and following continuous mode shutdown. Thruster valve temperatures should be between 40°F and 140°F non-operating and up to 225°F during and shortly after operation. One temperature sensor is mounted on the surface of each propellant tank near the top (hemispherical) end of the tank;

TABLE 3.4.2.6.2-1

TEMPERATURE SENSOR LOCATIONS

VTNK1T	Tank 1 temperature, θ = 105°
VTNK2T	Tank 2 temperature, θ = 285°
VLIN1T	Propellant line, TBD
VLIN2T	Propellant line, TBD
VLIN3T	Propellant line, TBD
VLIN4T	Propellant line, TBD
VJET1T	R1 thruster valve cover, θ = 82°
VJET2T	R2 thruster valve cover, θ = 22°
VJET3T	R3 thruster valve cover, θ = 262°
VJET4T	R4 thruster valve cover, θ = 202°
VJET5T	A5 thruster valve cover, θ = 105°
VJET6T	A6 thruster valve cover, θ = 285°
VJET7T	A7 thruster valve cover, θ = 105°
VTHMOT	OIM case near nozzle throat
VCASET	OIM case
VSAFET	S&A case

these sensors are used to indicate pressurant gas temperature. Using this information and the subsystem pressure from the pressure transducer, the propellant remaining can be determined (see Section 3.4.3.2.3). Pressurant temperature should be between 40°F and 140°F.

Four temperature sensors are placed on the propellant lines at locations determined to be representative of the entire subsystem. These locations are shown in Figure 3.4.1-3. Line temperatures should be between 40°F and 140°F.

Two temperature sensors are mounted on the OIM case, one near the nozzle and one near the mount

flange. The S&A has one temperature sensor on its case.

All of the temperature sensors are positive coefficient thermal resistors (thermistors). The tank, line, OIM and S&A sensors are all identical (type 908631-32) with a nominal resistance of 2700 ohms. Because of the higher temperatures of the thruster valves the temperature sensors mounted on these components are slightly different (type 908683-4); their nominal resistance is 3100 ohms.

It should be emphasized that at no time during the mission should any of the propellant temperature sensors read less than 40°F. The freezing point of hydrazine is 35°F; should the hydrazine freeze, catastrophic damage to the propulsion subsystem could occur.

3.4.2.6.3 Latch Valve Status. Each latch valve contains a SPDT status switch that is operated by the valve mechanism. The "valve closed" contact of the switch is grounded and the "valve open" contact is open. A one milliamp constant current is supplied to the center contact of the switch by the PCM Encoder through the DIM both of which are part of the data handling subsystem. When the latch valve is moved to the closed position, the current source is grounded and the input voltage drops below 2.56 vdc. This is interpreted by the data handling subsystem as a "0" bit. When the latch valve is moved to the open position, the current source is ungrounded and the input voltage rises above 2.56 vdc. This is interpreted by the data handling subsystem as a "1" bit.

3.4.2.6.4 Heater Status. Heater status telemetry is identified and discussed in detail in Section 3.8.3.8.

3.4.3 Propulsion Subsystem Operations

3.4.3.1 Initial Conditions. The latch valves and the thruster valves are the only commandable units in the propulsion subsystem. (Propulsion heaters, including those that are commandable, are discussed in Section 3.2, Thermal Design). Initial spacecraft spin up is performed automatically as part of the preprogrammed spacecraft separation sequence (Section 4.2.1). When initial ground station acquisition of the spacecraft occurs sometime after the completion of this sequence, the propulsion subsystem telemetry should indicate that all thruster valves and latch valves are closed.

If operation of the propulsion subsystem is interrupted later in the mission, the thruster valves will return to their normally closed position while the latch valves will remain latched in their position prior to the interruption.

3.4.3.2 Normal Operating Modes

3.4.3.2.1 Latch Valve Usage. The functions of the latch valves are to ensure that propellant is available for initial spin up by trapping it in the reservoirs and to isolate the thruster groups to provide redundancy in the event of a failure. Propellant supply for initial spin up is assured by launching the spacecraft with the latch valves closed and then opening them by the preprogrammed spin up sequence just prior to spin up.

In order to prevent catastrophic loss of propellant in the event of a failure downstream of the latch valves, the latch valves should be closed when the spacecraft is not in communication with the ground. The latch valves should also be closed immediately after any maneuver that uses the radial thrusters in order to prevent large spin speed changes in the event of a thruster valve open failure. During extended maneuvers with the axial thrusters when spacecraft telemetry is being monitored, it is recommended that the latch valves be left open.

This recommendation is based on the non-redundant design of the latch valve.

Prior to any maneuver the appropriate latch valve must be opened; i.e., latch valve 1 must be opened for any maneuver requiring thruster R3, R4, A5 or A7 and latch valve 2 must be opened for any maneuver requiring thrusters R1, R2 and A6. The appropriate latch valve must be open for thrust to be produced and to prevent possible damage to the thruster valve (see Section 3.4.3.4).

Latch valve position status can be determined directly from telemetry as discussed in Section 3.4.2.6.3.

3.4.3.2.2 Thruster Valve Usage. Since the thruster valves control propellant flow to the thrust chamber, they are the main operational elements of the propulsion subsystem. Thrusters are selected as required for the particular maneuver planned (Section 4.3.1), and the controls subsystem supplies the valve control power at the appropriate times in the spacecraft spin period. The thruster valves can be operated in the continuous fire mode as would be done during spin speed changes and axial ΔV maneuvers and in the pulsed mode as would be done during precession and ΔV maneuvers. In the pulse mode, pulse widths of either 128 ms or 512 ms are supplied by the controls subsystem. Table 3.4.3.2.2-1 details the pulse mode duty cycles at the normal spacecraft spin speeds. The 512 ms pulse is not usable (conservatively) at spin rates greater than 20 rpm because of the large angles transversed during the 512 ms pulse width at higher spin rates results in inefficient propellant usage and the duty cycle detector in the Controls Subsystem preclude any pulsed firing. At spin rates below 20 rpm either pulse width can be used; the 128 ms pulse width results in more exact maneuvers while the 512 ms pulse width results in shorter maneuvers.

Thruster valve operation cannot be directly verified in spacecraft telemetry. The first

indirect indication is a decrease in subsystem pressure following the propellant remaining curves given in Figure 3.4.3.2.3-1. A second indirect indication of thruster operation is the thruster valve temperature sensor. During thruster valve operation the temperature increases due to power dissipation in the coils and due to thermal soakback from the hydrazine decomposition. The magnitude and rate of temperature increase depends on initial temperature, duty cycle and propellant supply pressure. In general, the temperature tends to rise slowly; temperature rises of 20°F to 50°F after 10 minutes of pulse mode operation or 100 seconds of continuous operation are typical. After completion of thruster operation, the temperature drops several degrees in the first minute after shutdown and then stabilizes due to thermal soakback. Typical thruster valve temperature profiles are shown in Figures 3.4.3.2.2-1 (pulsed firing) and 3.4.3.2.2-2 (continuous firing).

Continuous operation of a thruster valve can also be detected by its power consumption of 21 watts maximum. The bus loads current is telemetered in the ACS format that will be used during maneuvers. Assuming the normal case where the bus loads is dissipating nominal spacecraft power, the bus loads current should increase during valve operation. The only reason this method cannot be used normally to verify pulse mode operation is the low sampling rate in relation to the pulse on time. Pulsed operations are verifiable via use of 2048 bps rate or use of the programmable format.

The final method of verifying thruster operation is through changes in spacecraft spin rate, attitude and/or velocity.

TABLE 3.4.3.2.2-1

PROPULSION SUBSYSTEM
PULSE MODE DUTY CYCLES

SPACECRAFT RPM	DUTY CYCLE
5	128 ms ON, 11,872 ms OFF 512 ms ON, 11,488 ms OFF
10	128 ms ON, 5872 ms OFF 512 ms ON, 5488 ms OFF
15	128 ms ON, 3872 ms OFF 512 ms ON, 3488 ms OFF
30	128 ms ON, 1872 ms OFF
60	128 ms ON, 872 ms OFF

3.4.3.2.3 Propellant Remaining. The current plan calls for loading 70 pounds of propellant into the Orbiter propellant tanks. The amount of propellant remaining can be determined from the pressure transducer telemetry signal (VTANKP), the pressurant temperature telemetry (VTNK1T and VTNK2T), and the propellant remaining curves given in Figure 3.4.3.2.3-1. Curves are given for the minimum, maximum and nominal temperatures. For other temperatures, the following equation may be used:

$$\text{Propellant Remaining} = 0.0364 \left[2265 - \left(\frac{119{,}673}{P} \times \frac{459.6+T}{529.6} \right) \right]$$

where

P = Tank pressure, psia

T = Tank temperature, °F

3.4.3.3 Abnormal Operating Modes

3.4.3.3.1 Possible Failure Modes. Table 3.4.3.3.1-1 presents a list of general failure modes and corrective actions that are possible for the propulsion subsystem.

3.4.3.3.2 Use of Redundancy. As can be seen from a review of Table 3.4.3.3.1-1, the primary corrective action for a non-catastrophic failure is to use the redundant thruster group. If the failure is anything but a closed failure of a single thruster valve, the appropriate latch valve must be closed thus preventing use of any of the thrusters in that particular group.

In general, the use of only one thruster group causes less efficient use of propellant and more complex maneuvers due to additional cross coupling. Typical maneuvers that can be performed with a single thruster group include:

- Spin speed changes using a single radial thruster for each direction
- Attitude precession maneuvers using a single axial thruster or the radial thruster pair not canted through the c.g. at the time of the maneuver.
- Velocity changes with an axial thruster that may require a large precession maneuver to properly align the thrust vector.

Each of these maneuvers is discussed in detail in Section 4.3.1, (Maneuvers).

3.4.3.4 Operational Restrictions. The following operational restrictions had been determined to be necessary for the longevity and proper operation of the propulsion subsystem.

- The appropriate latch valve must be open before the operation of any thruster valve. If the thruster valve is subjected to continuous power for approximately 60 seconds without propellant flow, permanent damage to the thruster valve coils can result.

TABLE 3.4.3.3.1-1
POSSIBLE FAILURE MODES

Possible Failure Mode	Method of Recognition	Corrective Action
1. Thruster valve fails open.	With no input to thruster valve and L/V open, T/V temperature high; pressure decreasing.	Close appropriate L/V; use redundant thruster group.
2. Failure of non-redundant heater downstream of L/V.	Temperature sensor telemetry.	Close appropriate L/V; use redundant thruster group.
3. Leakage failure downstream of L/V.	Only when L/V open S/S pressure decreases T/V temperature normal.	Close appropriate L/V; use redundant thruster group.
4. Latch valve fails closed.	Telemetry signal; thrusters inoperable.	Use redundant thruster group.
5. Thruster valve fails closed.	T/V temperature remains normal or decreases prematurely. S/S pressure remains normal during "operation" of ADP control.	Use other thrusters.
6. Leakage failure upstream of L/V.	With L/V closed S/S pressure decreases while tank temperature remains constant.	None.
7. Failure of redundant heaters upstream of L/V.	Temperature sensor telemetry.	If possible, maintain temperature by shelf heaters and/or sun angle change.
8. Failure of both thruster group non-redundant heaters.	Temperature sensor telemetry.	Same as 7.
9. OIM/S&A heater failure.	Temperature sensor telemetry.	Use redundant heaters.

- The thruster valve temperature must be 40°F or higher prior to operation to prevent damage to the thruster. This minimum temperature should be maintained automatically by the subsystem heaters.
- The subsystem temperature must be 40°F or higher at all times to prevent propellant freezing which would rupture subsystem components. This minimum temperature should be maintained automatically by the subsystem heaters.
- Prior to opening a latch valve, telemetry should be checked to assure all heaters downstream of the valve are operating. This will determine if it is safe to open the latch valve in the event a heater has failed and eclipses have ruptured a line or component.

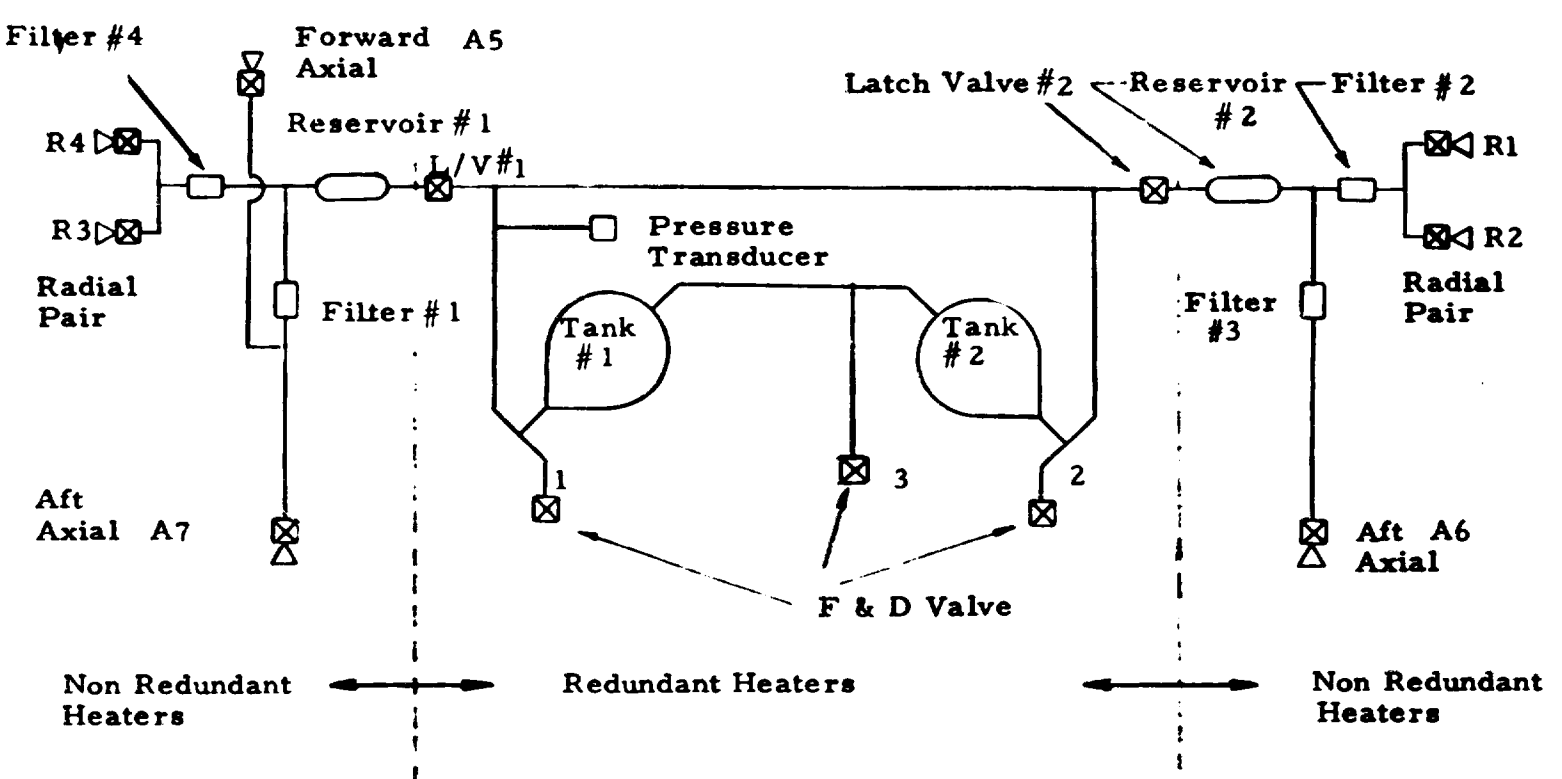
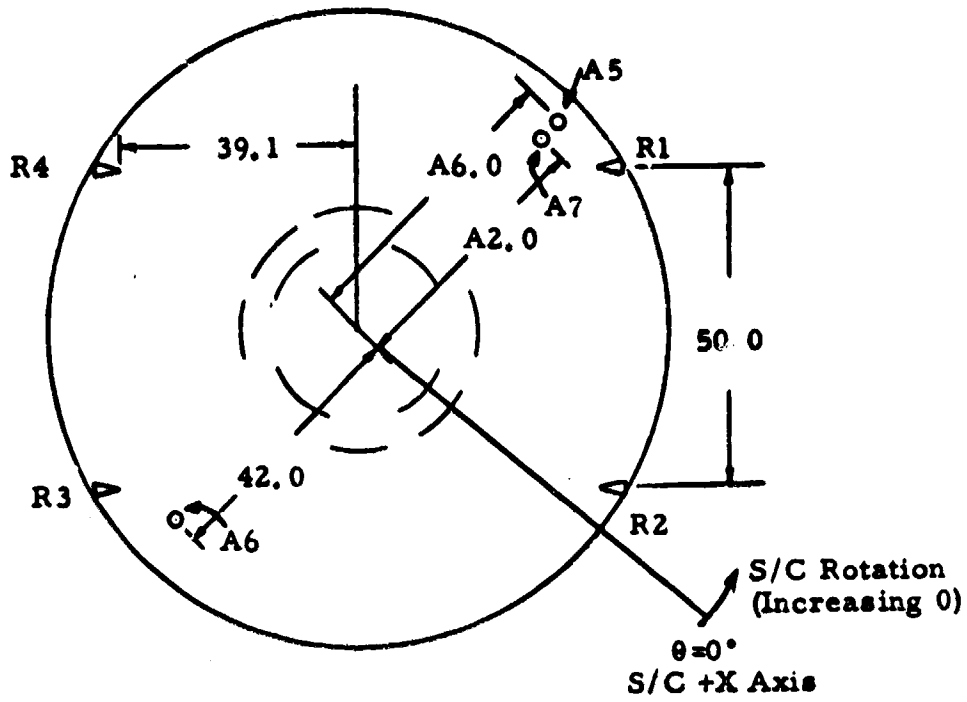


Figure 3.4.1-1. Orbiter Propulsion Subsystem.



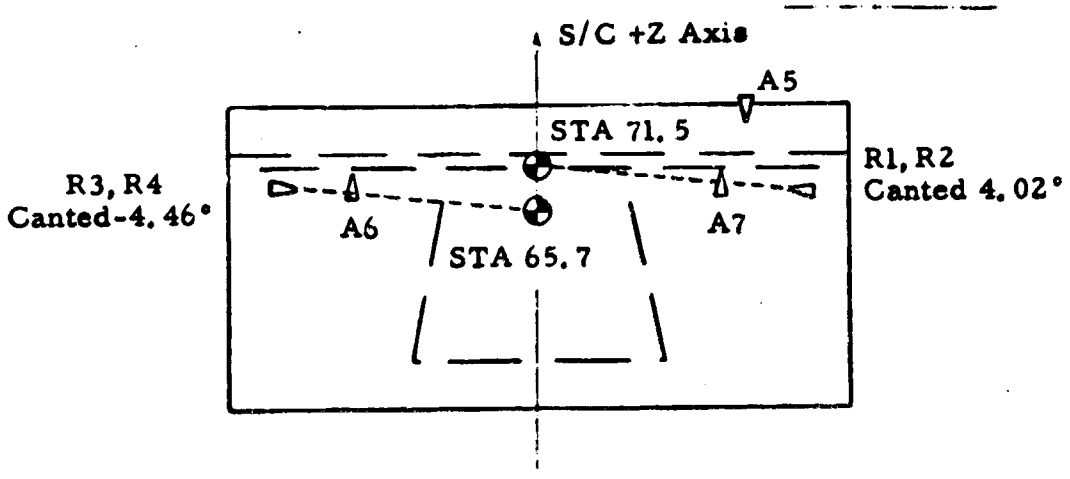


Figure 3.4.1-2. Orbiter Thruster Layout

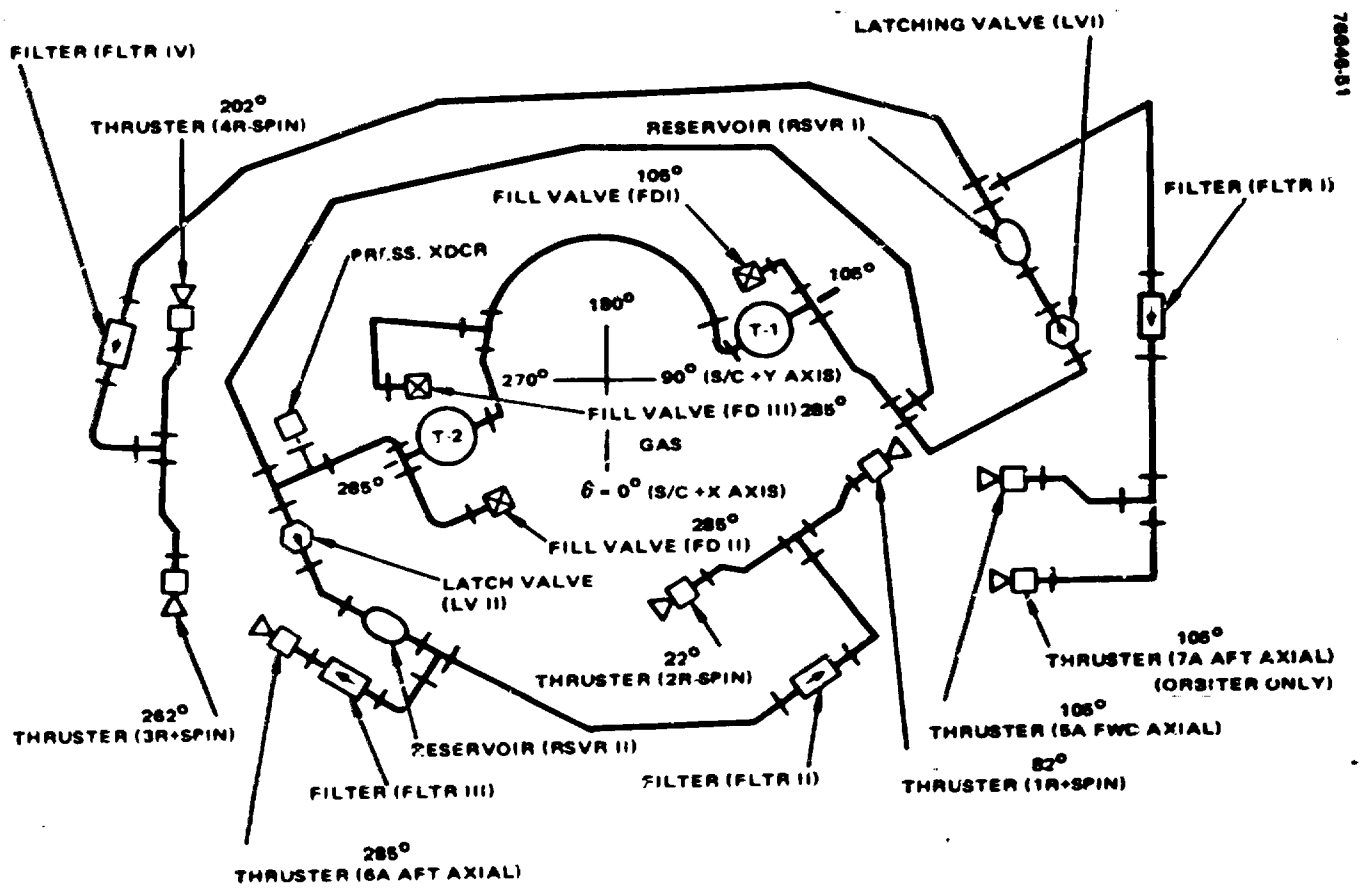


FIGURE 3.4.1-3. PROPULSION SUBSYSTEM LAYOUT

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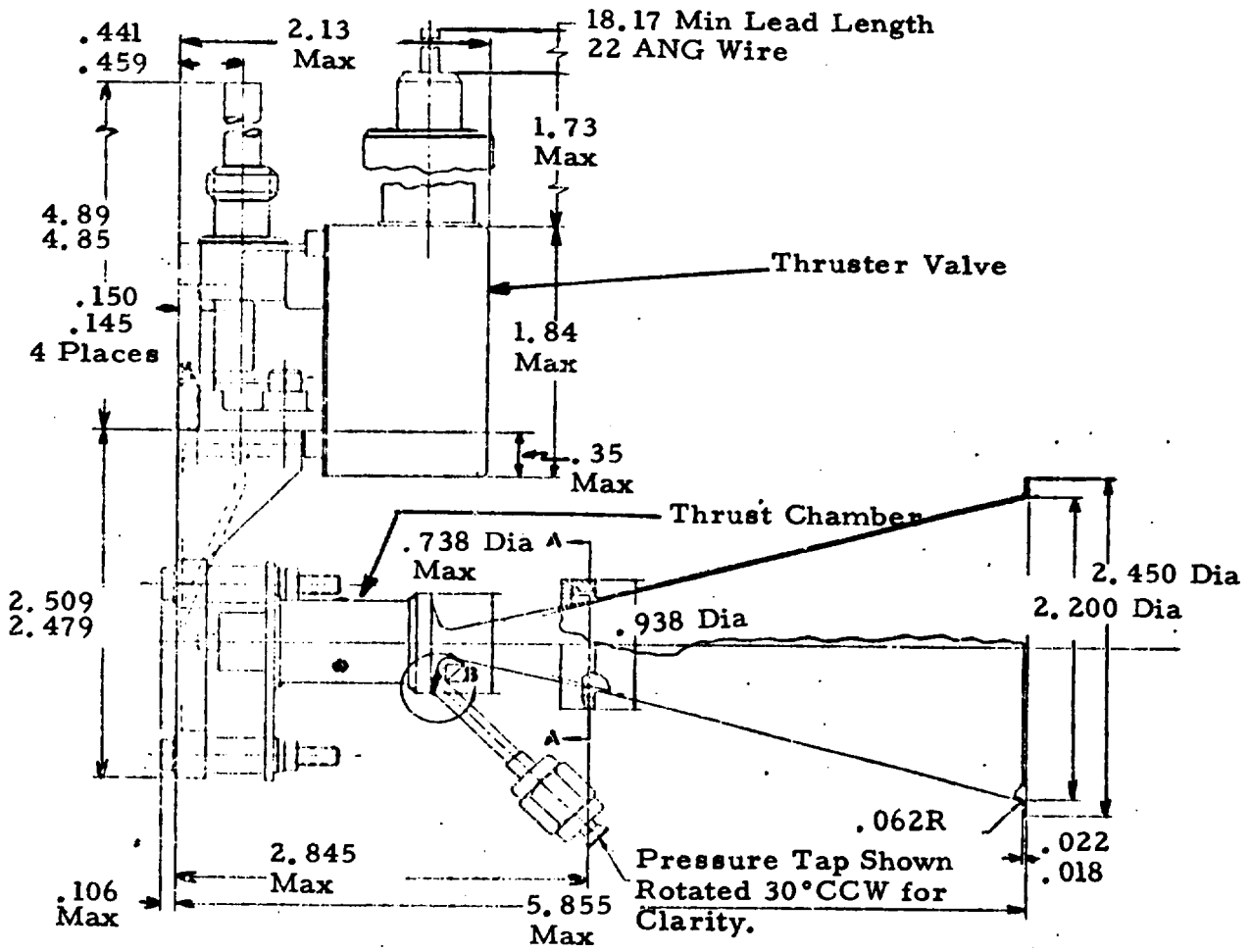


Figure 3.4.2.1-1. Thruster Assembly

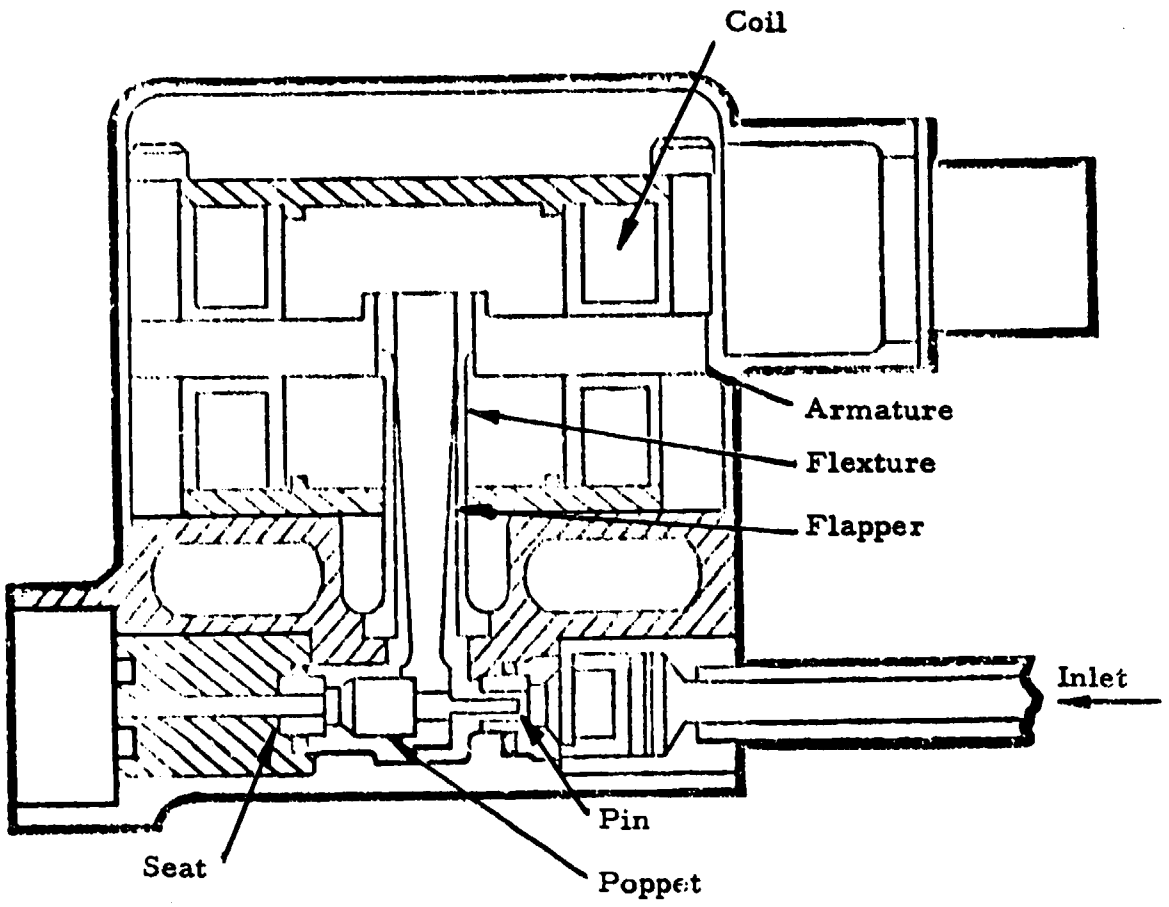


Figure 3.4.2.1-2. Thruster Inlet Valve

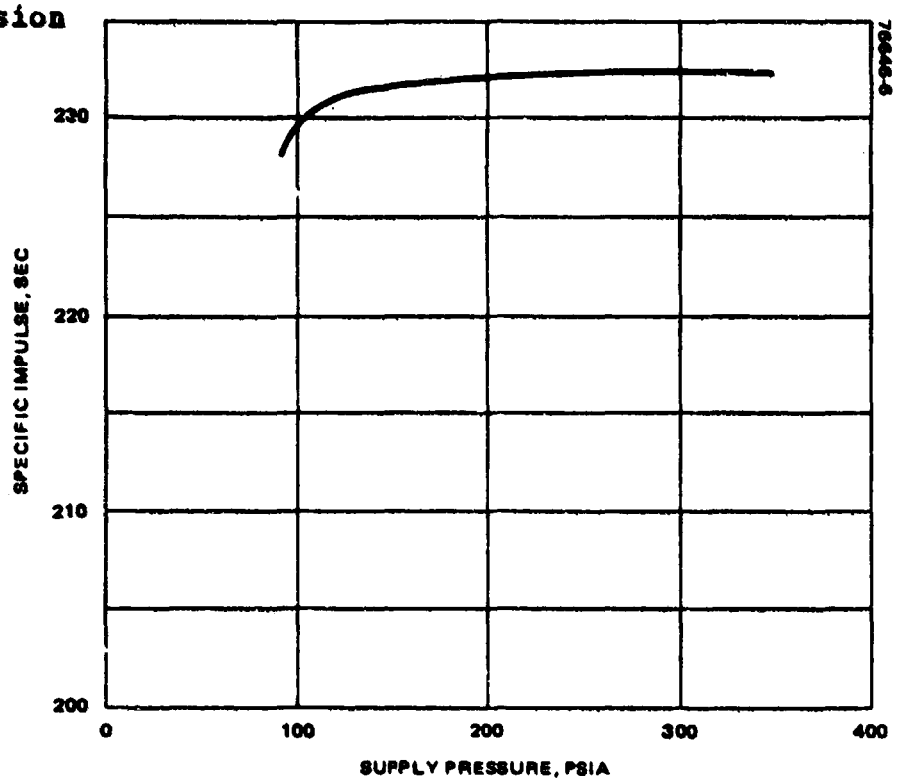


FIGURE 3.4.2.1-3. STEADY STATE SPECIFIC IMPULSE (T = 30 SEC) VERSUS SUPPLY PRESSURE

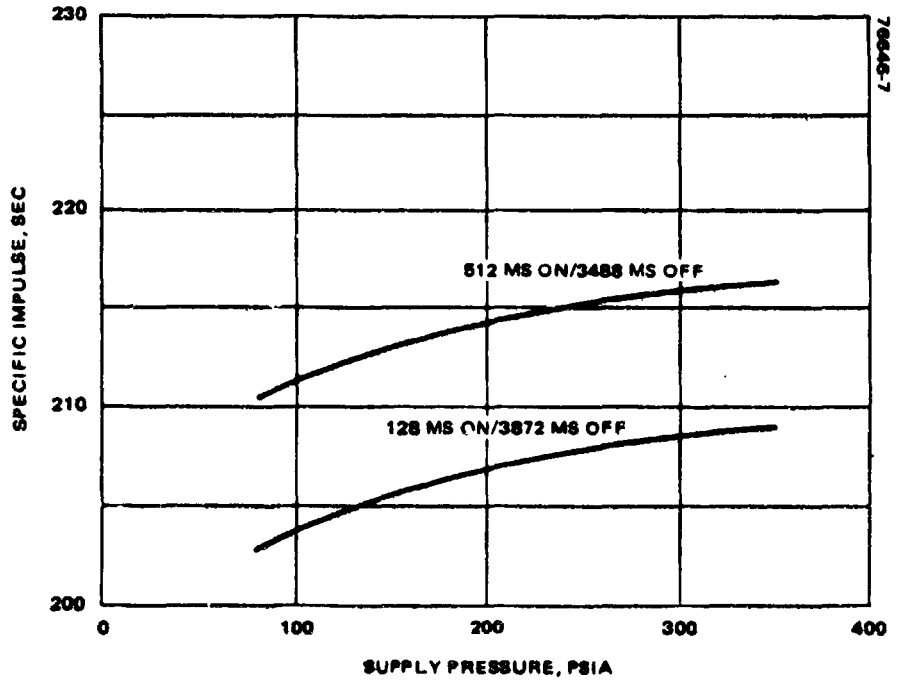


FIGURE 3.4.2.1-4. PULSE MODE SPECIFIC IMPULSE (NOMINAL)

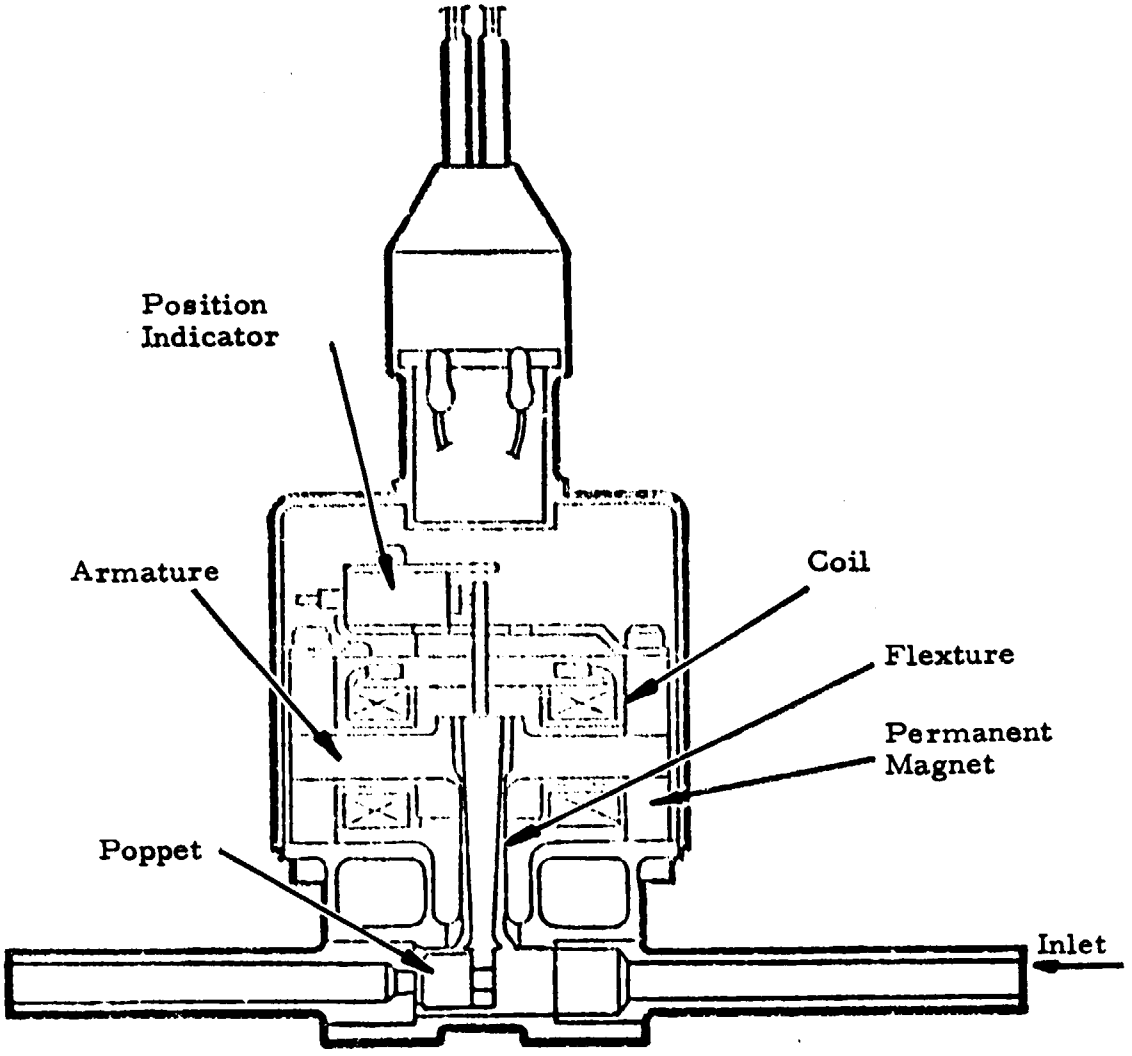


Figure 3.4.2.2-1. Latch Valve

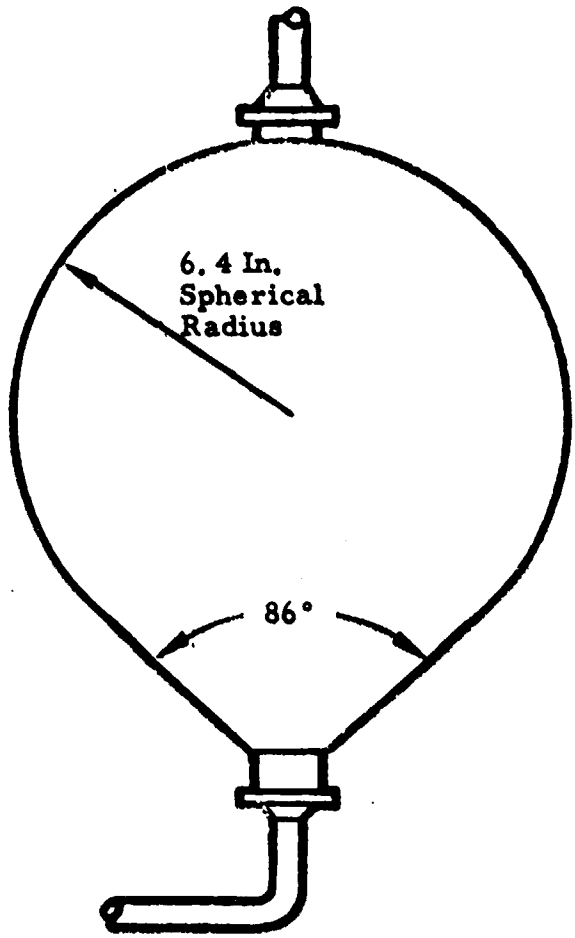


Figure 3.4.2.3.1-1. Hydrazine Tank

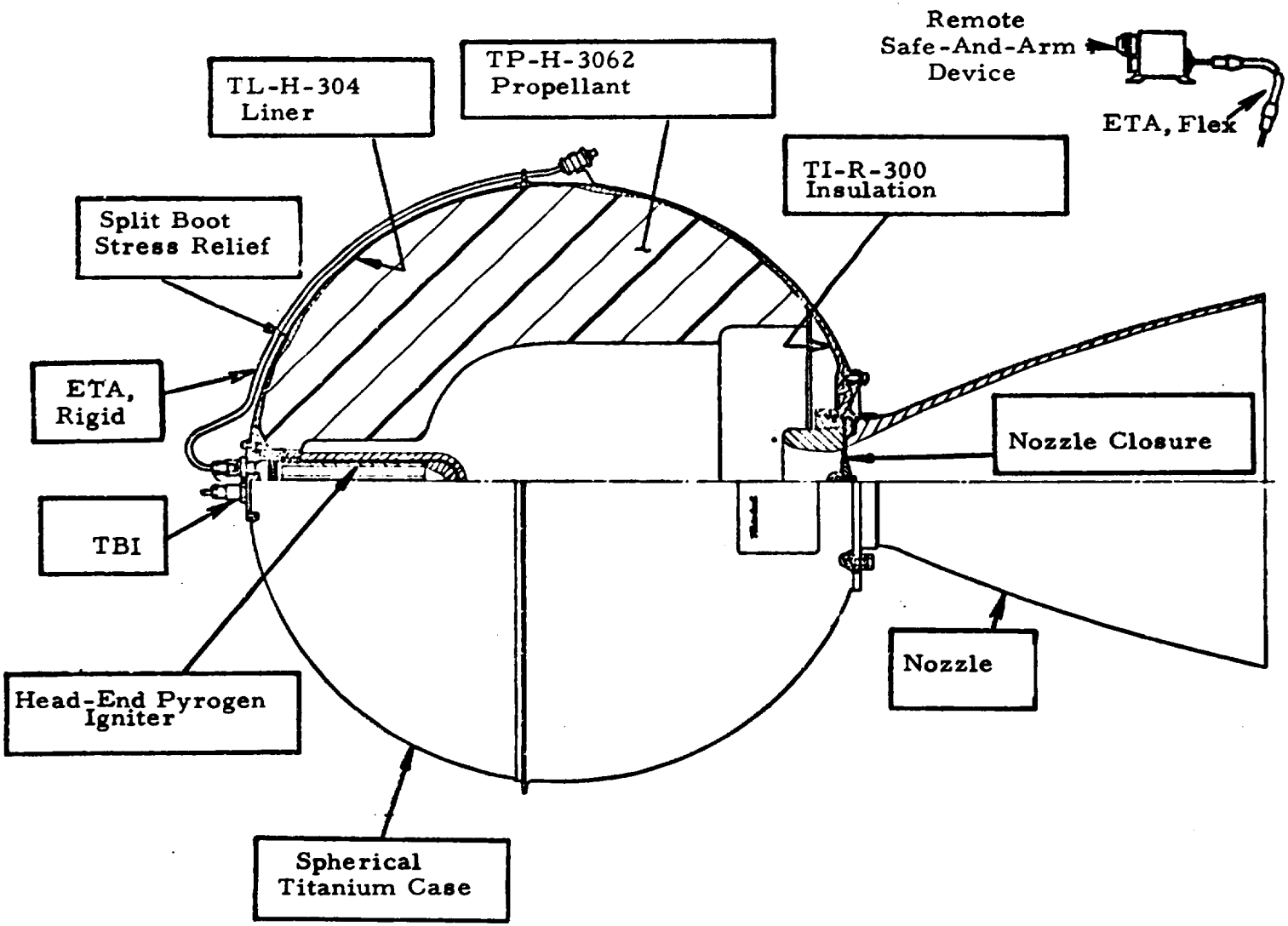


Figure 3.4.2.5.1-1. Orbit Insertion Motor

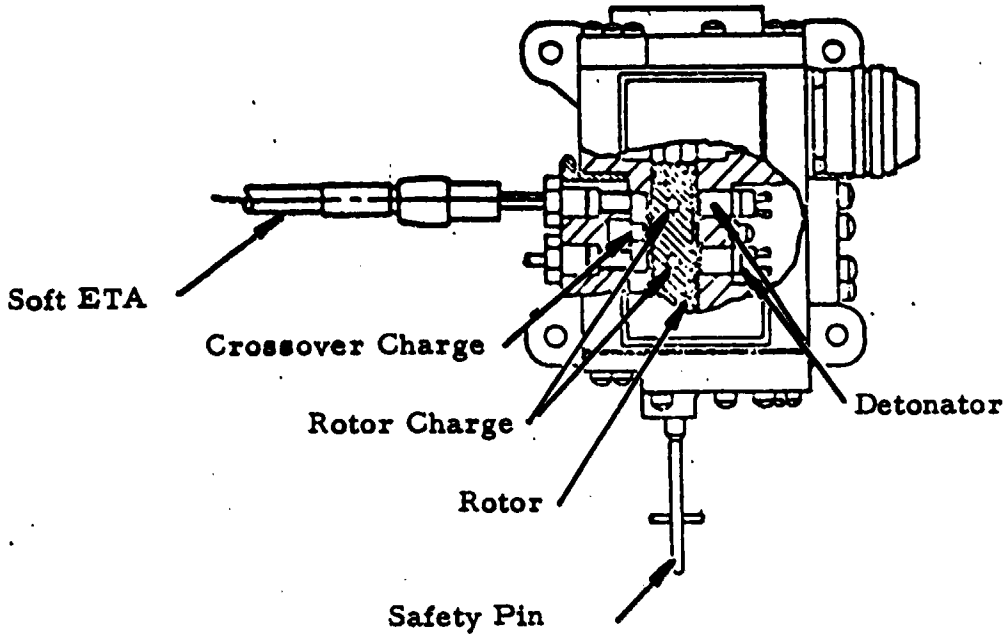


Figure 3.4.2.5.2-1. Safe and Arm Device

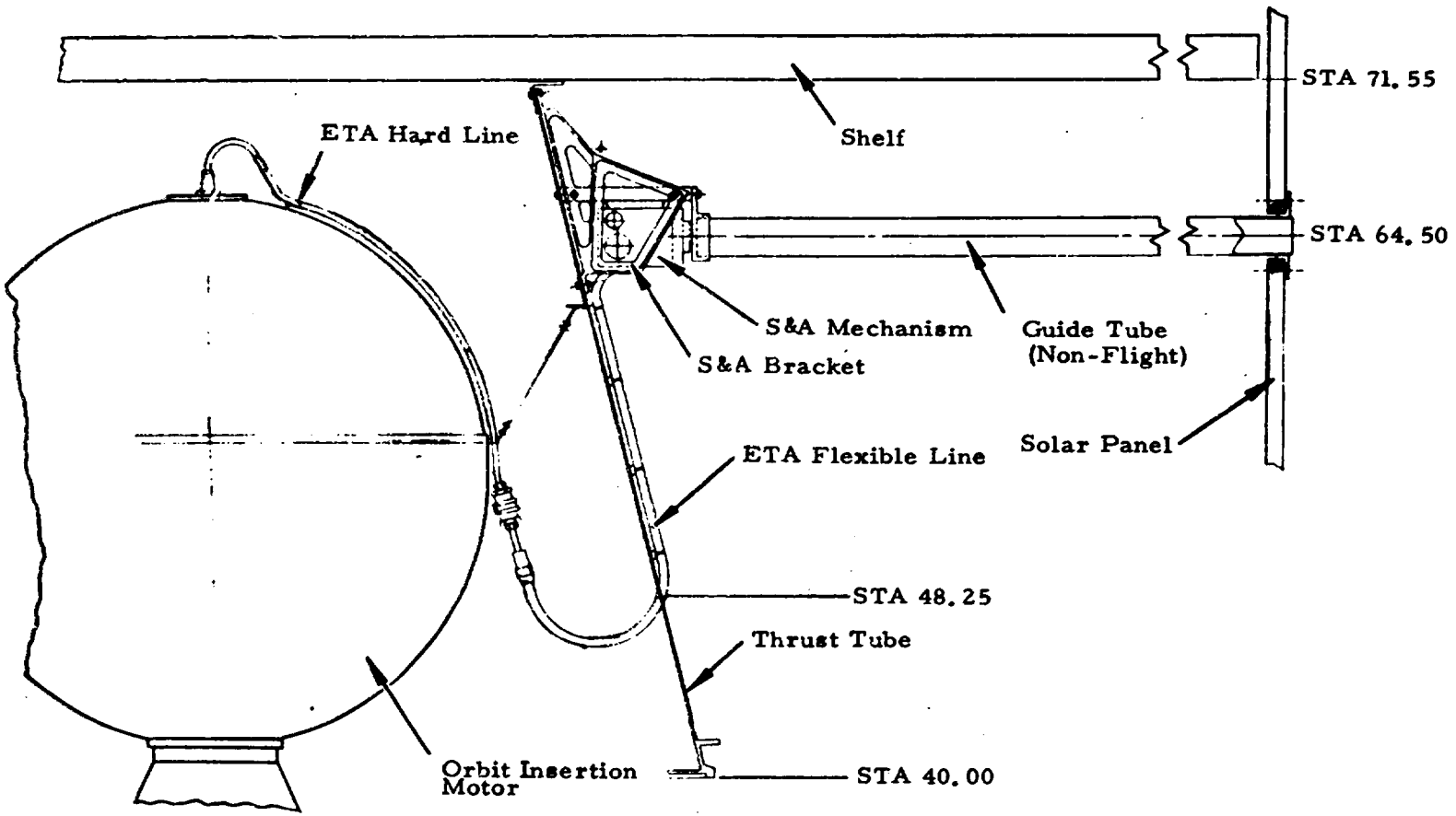


Figure 3.4.2.5.2-2. S&A and ETA Installation

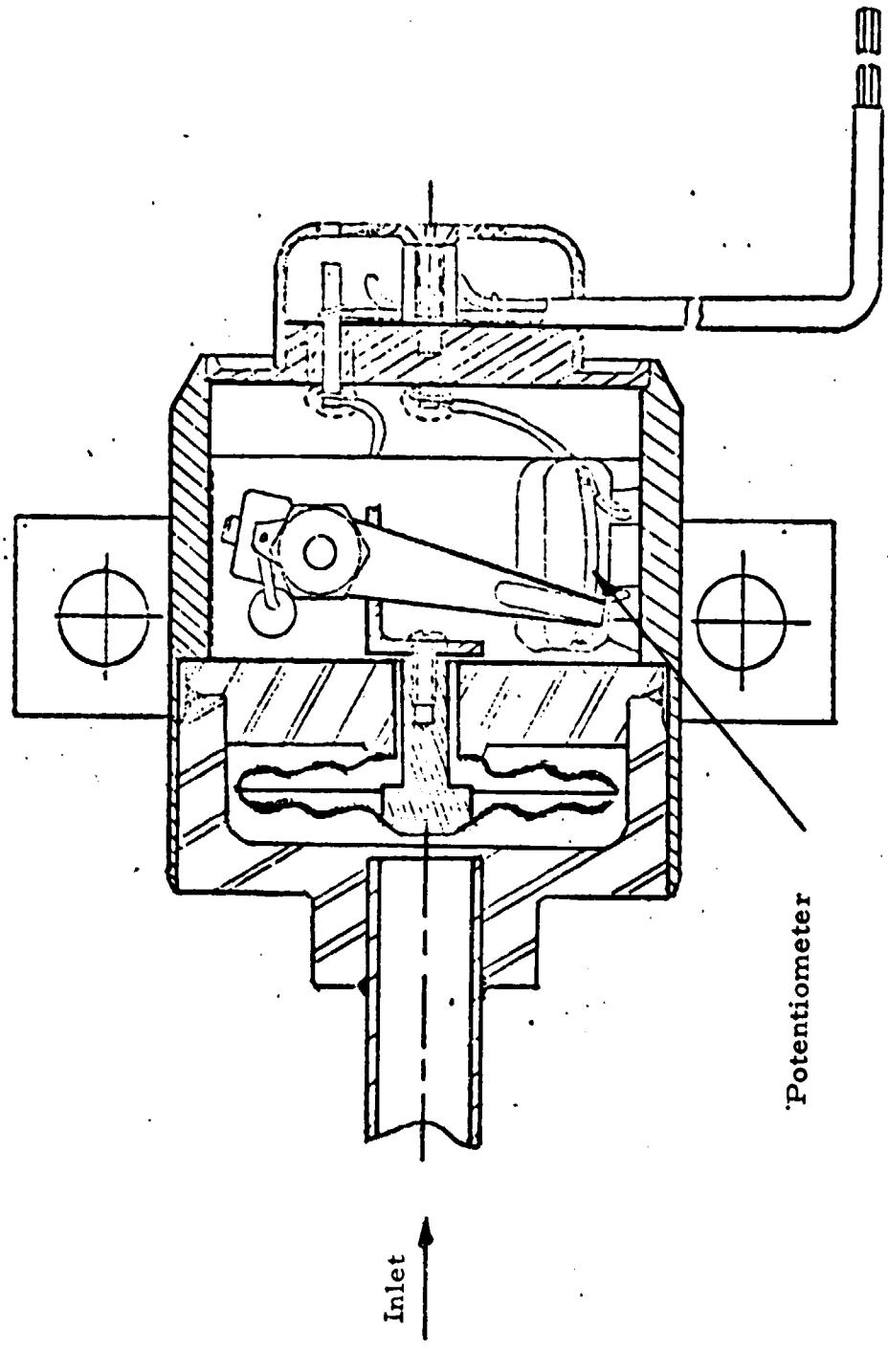


Figure 3.4.2.6.1-1. Pressure Transducer

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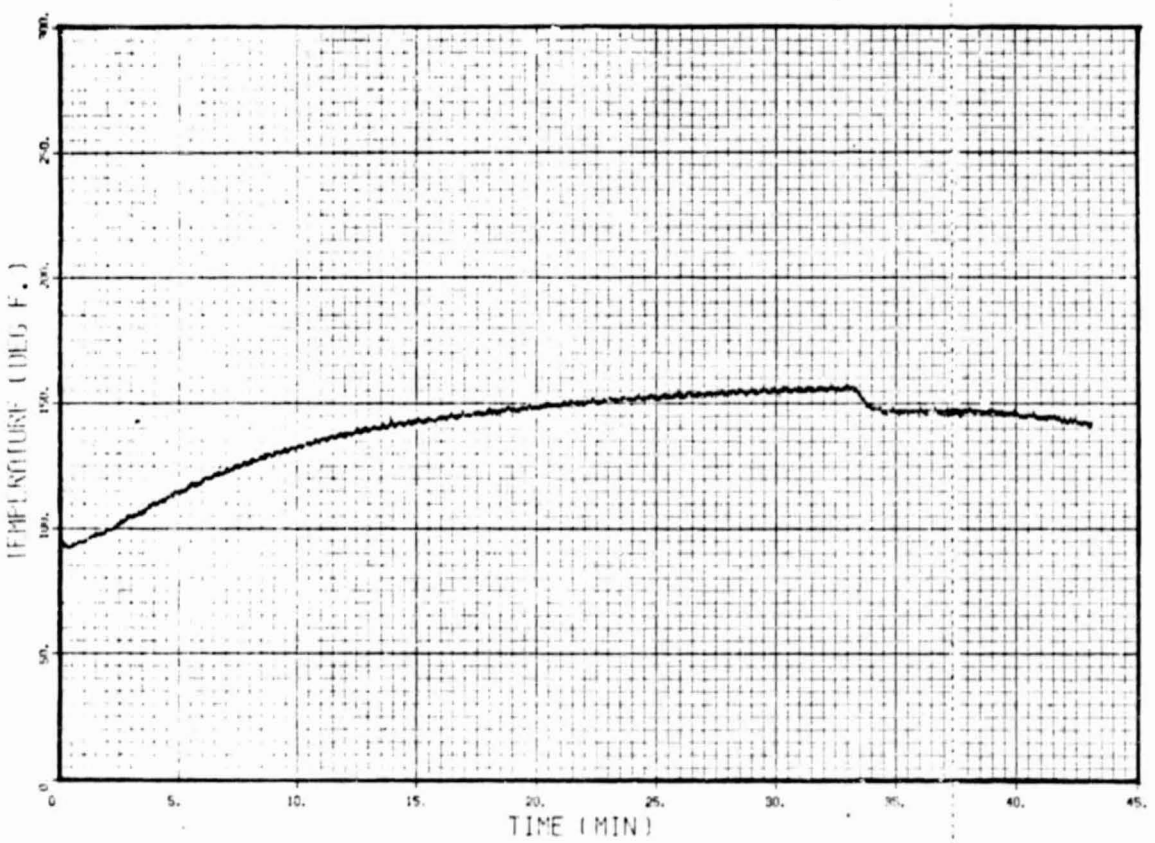


Figure 3.4.3.2.2-1. Typical Thruster Valve Temperature Profile-Pulsed Firing

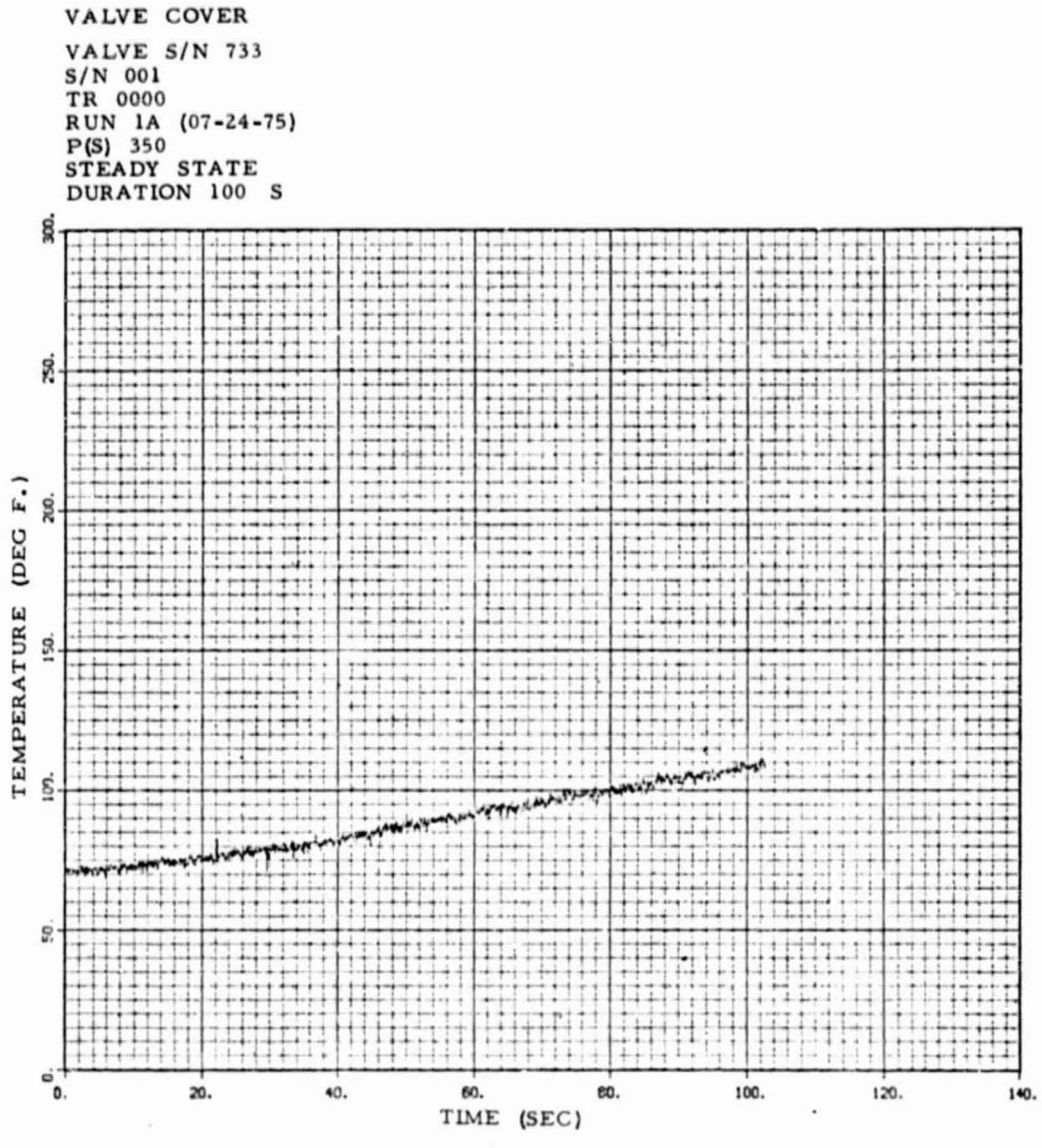


Figure 3.4.3.2.2-2. Typical Thruster Valve Temperature Profile - Continuous Firing

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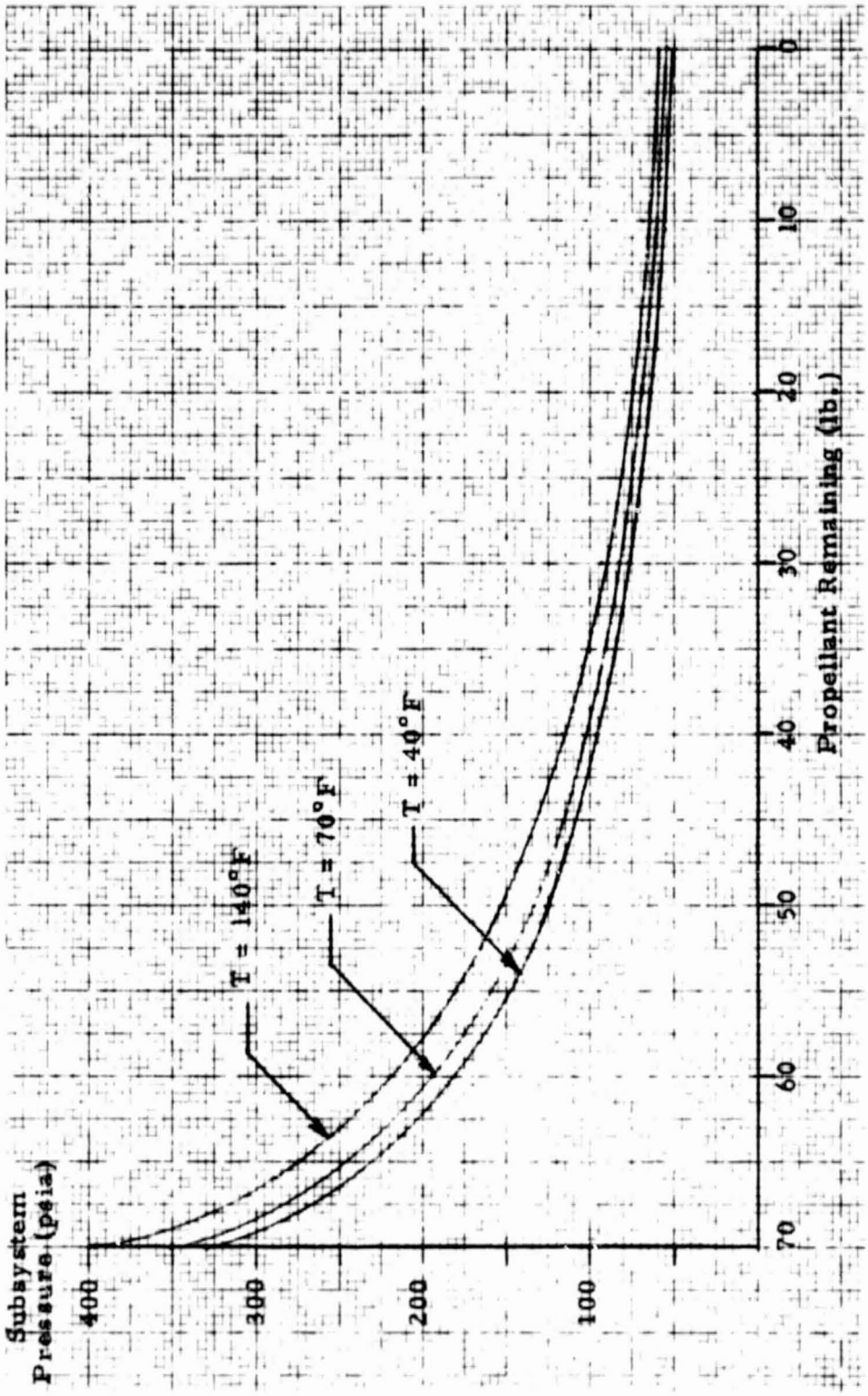


Figure 3.4.3.2.3-1. Orbiter Propellant Remaining

3.5 DATA HANDLING SUBSYSTEM

3.5.1 Subsystem Description. The data handling subsystem contains all the telemetry processing capability on the spacecraft. The subsystem processes analog, serial digital and bilevel telemetry data from the scientific instruments and spacecraft subsystems for transmission to earth. The subsystem provides timing, clock and status signals; data sampling and formatting; convolutional encoding; biphase modulation and data storage for the spacecraft. A functional block diagram for the subsystem is presented in Figure 3.5.1-1 (Appendix C).

The subsystem consists of eight (8) Data Input Modules (DIMs), two (2) redundant PCM Encoders (PCMEs), two (2) redundant Telemetry Processors (TPs), and two (2) redundant Data Storage Units (DSUs). On the spacecraft equipment shelf, these units are arranged as shown in Figure 3.5.1-2. All the data handling subsystem units, except the DIMs, are connected to the switched loads bus. The DIMs receive secondary power from each PCME. Each DIM is power strobed ON just prior to processing a telemetry input, and upon completion, the DIM returns automatically to the standby mode. Discrete and quantitative commands are used to control the functions of all the subsystem units. Figure 3.5.1-3 shows the major cross-strapping between the elements of the subsystem.

The data handling subsystem receives all telemetry data inputs through the eight (8) DIMs and multiplexes these inputs into a PCM/PSK square wave subcarrier that is sent to the spacecraft transmitters. Each DIM is a 32-channel random access commutator with a single input control line and a single Pulse Amplitude Modulated (PAM) output data line. The DIMs provide user-dedicated read envelopes and a common gated read clock to all users of serial digital telemetry; and a 1 milliampere precision current pulse source of a nominal 294 microseconds duration for users of resistive transducers (i.e., conditioned analog channels).

The 1 milliampere precision current pulse source is developed in each PCME and provided on a single line to each DIM. Analog data (conditioned or non-conditioned) is sampled, buffered and gated to the output data line as a single PAM pulse; serial digital data is gated to the output data line as 8-bit serial words; and bilevel data is sampled in parallel 8-bit bytes and also gated to the output data line as 8-bit serial words. The ON/OFF control of each individual DIM is through the PCM ENCODER DIM CONTROL quantitative command, i.e., PCMQ1 for PCME1 and PCMQ2 for PCME2. This command is sent to the PCME in the case of a failure where a DIM does not automatically return to standby after being used to process a telemetry input. The ON/OFF status of each DIM is not telemetered because it can be determined from the sense of the telemetry data. The DIMs provide the standardized telemetry interface with all the scientific instruments and spacecraft subsystem users.

The PCME accepts the PAM analog and serial data from all the DIMs, processes and synchronizes the data and generates a non-return-to-zero-level pulse-code-modulated (NRZ-L/PCM) output bit stream to each TP. The PCME performs an eight bit analog-to-digital (A/D) conversion on the analog data and regenerates the digital data. Information from the TP to select the telemetry data channel required by the selected telemetry format is relayed by the PCME to the DIMs. For routing to channels that require a conditioning current, the PCME provides a precision 1 milliampere current pulse source to each DIM. The PCME receives a quantitative command to control the ON/OFF status of the DIMs, as mentioned above. Commanding a DIM ON enables the PCME data input line and also enables the DIM to be power strobed ON via the control line. A discrete ON/OFF command is also received by the PCME to control the power status of the unit, i.e., PCM ENCODER 1 ON/2 OFF (PCM19 or PCMA9), PCM ENCODER 2 ON/1 OFF (PCM29 or PCMB9) and PCM ENCODERS OFF (PCM1Ø or PCMAØ). The telemetry

ON/OFF status is provided by DPCM1S for PCME1 and DPCM2S for PCME2.

The TP is the main unit in the data handling subsystem. The TP contains the master spacecraft clock that generates all the timing signals required by the subsystem, the scientific instruments and other spacecraft subsystems. The TP provides the 24-bit spacecraft time code, 4096 second clock, clock signals (i.e., 32,768 Hz and 2048 Hz), telemetry synchronization signals (i.e., major frame rate, minor frame rate and telemetry word rate), and the bit rate status signals. A data minor frame consists of 64 telemetry words consisting of eight bits each (512 bits total). There are 64 minor frames in a major frame. A subcommutated format also consists of 64 telemetry words, except that each word in the subcommutated format appears only once in a unique word position in the minor frame. Therefore, it takes one major frame to complete a subcommutated format. The Orbiter spacecraft has assigned to it: 13 pre-programmed minor frame formats, one (1) programmable minor frame format and three (3) pre-programmed subcommutated formats, as identified in Table 3.5.1-1. The telemetry data words required in these formats are sequentially addressed by the TP and are sent from the DIMs through the PCME. The TP receives the telemetry data by addressing the PCME, and conditions it for transmission to the spacecraft communications subsystem, or, by command, to the data memory, or both.

The data processing in the TP includes convolutionally encoded or uncoded data and biphase modulation of the data onto the 16,384 Hz square wave subcarrier. The functions of the TPs are configured by the TELEMETRY PROCESSOR CONTROL quantitative command, i.e., TPCQ1 for TP1 and TPCQA for TP2. Minor frame format words 3 and 8 (i.e., PSU status) give the telemetry status of the TPs.

TABLE 3.5.1-1

ORBITER TELEMETRY FORMATS

Pre-Programmed Minor Frame Formats:

1. Orbiter Engineering
2. ACS
3. Command Memory Readout
4. Playback (Partial Data Memory Readout and Remainder for Pre-Programmed Telemetry)
5. Data Memory Readout
6. Launch/Cruise
7. Periapsis A
8. Periapsis B
9. Periapsis C
10. Periapsis D
11. Periapsis E
12. Apoapsis A
13. Apoapsis B

Pre-Programmed Subcommutated Formats:

1. Subcom A
2. Subcom B
3. Subcom D

Programmable Minor Frame Format:

"Programmable"

The Data Memory, consisting of two DSUs, is used to store telemetry data on the spacecraft. Each DSU is a magnetic core memory having a data storage capacity of 524,288 bits. Each of the two (2) DSUs interface with, and is controlled by, either of the two (2) TPs. The data stored is not lost during power failures, shut-down, or read-out. Internally, each memory is organized to read-out all 524,288 bits in a FIFO (first-in, first-out) mode. The ON/OFF control of the DSUs

is provided by commands DSU1 ON (DSU19 or DSUA9), DSU2 ON (DSU29 or DSUB9) and DSU1 and 2 OFF (DSU1Ø or DSUAØ). Telemetry ON/OFF status is given by MINOR FRAME OVERHEAD TM Word 8, bit 3 for DSU1 and TM Word 8, bit 2 for DSU2.

3.5.2 Unit Descriptions. The following sections provide detailed descriptions of the units within the data handling subsystem.

3.5.2.1 Data Input Module. The DIM receives analog, serial digital and bilevel data from the scientific instruments and spacecraft subsystems and routes these data to the TP via the PCME. The DIM receives control signals, the 1 milliampere precision current pulse source and power from the PCME. The DIM provides a read clock and read envelopes to users of serial digital data; and relays the precision current pulse source to users of conditioned analog and bilevel channels. The main components of the DIM are the multiplexer, Manchester decoder and control logic, read envelope and clock generator, and power control. In the following sections, these components are described.

3.5.2.1.1 Multiplexer. The multiplexer receives the 32 telemetry channel inputs from the users. Each input is protected against damage from overvoltages as large as +50 volts and -30 volts. The multiplexer is implemented by using eight (8) 4-channel input multiplexers which are controlled by two (2) 4-channel switches that each provide a one-out-of-four decode. Current switches are also contained in the multiplexer that operate synchronously with the multiplexer input switches. The purpose of the current switches is to commutate the 1 milliampere precision current pulse source into the selected multiplexer channel. Six (6) precision current pulse source inputs are provided, with each input allocated to a unique 4-channel input multiplexer.

3.5.2.1.2 Manchester Decoder and Control Logic. The Manchester decoder in the DIM receives a 10-bit Manchester coded channel address word formatted by the PCME. It decodes the address word and

generates a two-phase bit clock and NRZ serial data. These signals are provided to the control logic. The control logic provides the proper coding to the multiplexer to select the desired channel by selecting the 4-channel multiplexer and channel within the multiplexer. Programming pins are used in the unit connector to choose the appropriate distribution of analog, serial digital and bilevel channels for the DIM. The control logic provides the read clock and read envelope outputs to the users from the mode (i.e., programming) selected for the DIM and the channel address word.

3.5.2.1.3 Read Envelope and Clock Generator. The read envelope and clock generator circuit provides the control signals required for serial digital data transfer from the users. The circuit consists of a one-out-of-sixteen decoder matrix that generates the sixteen read envelope signals and a clock amplifier that provides buffering of the gated read clock signal from the control logic. The read envelope signals are generated in synchronization with the sampling of each 8-bit serial digital word. Up to sixteen serial data channels may be accommodated by wiring the data line to the appropriate input channel (i.e., channels 0 through 15).

3.5.2.1.4 Power Control. The DIM is turned on and off by a variable width pulse on the control line from the PCME. Pulses of 2 milliseconds and 230 microseconds will turn the DIM ON and a 36-millisecond pulse will turn OFF the DIM. The DIM is power-strobed as a means of minimizing power dissipation during idle or non-interrogate times. Power for the unit is provided by +15 volt and -15 volt regulated outputs and ground from the PCME that are inputted to power switches. The power switches apply the +15 volts, -15 volts, and ground to the internal electronics when an address word is detected on the control line. When there are no further pulses on the control line, the Manchester decoder senses this and sends a signal to the power control circuitry to return the DIM to the standby condition.

3.5.2.2 PCM Encoder. The PCME accepts PAM analog and digital data from the DIMs and formulates an NRZ-L/PCM (Non-Return to Zero level/Pulse Code Modulation) output data stream to the TPs. The main functional components of the PCME are the multiplexer, comparator and A/D converter, Manchester decoder and associated logic, DIM decoder, and power supply. These functions are described in the following sections.

3.5.2.2.1 Multiplexer. The multiplexer receives the PAM data input from each of the eight (8) DIMs along with a reference ground. The multiplexer consists of a differential commutator that differentially multiplexes the PAM data and reference ground from each DIM to the comparator. A distributor commutates a precision 1.000 milliampere ±0.5 percent reference current (294 microseconds) to the selected DIM for signal conditioning of transducer resistive inputs. The precision current source is outputted from the multiplexer to each DIM on a dedicated line.

3.5.2.2.2 Comparator and A/D Converter. The comparator consists of a unity gain differential amplifier followed by a precision voltage comparison circuit. The output of the comparator goes to the A/D converter. The differential input from the multiplexer is converted into a single-ended signal by the differential amplifier in the comparator. The treatment of the data, as analog or digital, is controlled by the A/D bit in a 20-bit instruction and address word supplied by the TP. For analog inputs, the comparator compares the input signal to a known reference. The reference is provided by a digital-to-analog feedback circuit in the A/D converter consisting of a binary voltage weighted resistor ladder, switches and a precision -5.120 vdc reference supply. The output of the comparator is encoded into 8-bit digital words by the A/D converter which employs a successive approximation technique. To process digital data, the register and switches are forced to half scale for the entire word time so that the ladder is generating a constant +2.56 volt reference. Thus, the A/D converter functions as a fixed threshold detector

for making one-zero decisions on the incoming digital data.

3.5.2.2.3 Manchester Decoder and Associated Logic. The Manchester decoder receives a 20-bit instruction and address word from the TP for three (3) times in succession. The decoder generates a coherent bit clock from the incoming 20-bit words, that provides timing for the bit counter circuit. In addition, the coherent bit clock enables the serial transfer of the instruction and address portion (minus 8-bit sync code) of the 20-bit word into a 12-bit shift register. The 11 instruction and address bits (i.e., A/D bit, 5 DIM address bits and 5 channel address bits) of the 20-bit word constituting the first interrogate subcycle are parallel transferred into an 11-bit holding register by the 12th bit in the shift register. The 5-bit channel address is parallel transferred during the 2nd interrogate subcycle into the address generator where it is converted into a 10-bit word required by the DIMs consisting of a 5-bit sync followed by the channel address.

3.5.2.2.4 DIM Decoder. The DIM decoder provides the channel address information to each DIM on a dedicated output. The DIM decoder circuit receives parallel DIM address bits from the holding register to select the DIM output line, the serial 10-bit channel address word from the address generator to select the DIM channel and power control commands from the power control circuit to turn on and off the DIMs. The ON/OFF control pulses to the DIMs go out on the same lines as the 10-bit channel address word. The power control circuit receives the PCM ENCODER DIM CONTROL quantitative command (PCMQ1 for PCME1 and PCMQ2 for PCME2) and provides a pulse width modulated technique to control the DIMs. Command execution is synchronized to the beginning of the minor frame.

3.5.2.2.5 Power Supply. The PCME has an internal power supply that operates from the +28 volt switched loads bus and is commandable ON and OFF. A series regulator provides regulation of the input

bus voltage and an electric conversion unit (ECU) generates all the secondary voltages required by the PCME, including the power supplied to the DIMs. The PCME power supplies are commanded ON and OFF by three discrete commands, i.e., PCM ENCODER 1 ON/2 OFF (PCM19 or PCMA9), PCM ENCODER 2 ON/1 OFF (PCM29 or PCMB9), and PCM ENCODERS OFF (PCM1Ø or PCMAØ). The ON/OFF status is provided to telemetry as DPCM1S for PCME1 and DPCM2S for PCME2.

3.5.2.2.6 Internal Command Response. The PCME receives both quantitative and discrete commands from the COMs. The following sections describe the internal response of the PCME to the different commands received.

3.5.2.2.6.1 PCM ENCODER DIM CONTROL Quantitative Command. Power control for the DIMs is processed by the DIM power control circuit in the PCME. The 16-bit quantitative command (i.e., bits 25 through 40 of the uplink command format) (PCMQ1 for PCME1 or PCMQ2 for PCME2) is stored in a 16-bit shift register, and the verification pulse at the end of the quantitative command is ignored. Each pair of bits in a given command word contain the power ON/OFF information for a DIM. Power control for the DIMs is implemented using a pulse width modulation technique. Command execution is synchronized to the beginning of a minor frame by using the second negative transition of the minor frame signal following receipt of a command to trigger three (3) one-shots whose periods represent the power ON/OFF states of a DIM. The proper pulse is gated to the DIM address line as a function of the information contained in the command word. A code of "01" or "11" will generate a 36-millisecond pulse which will turn a DIM OFF. A code of "00" or "10" will turn a DIM ON; a "00" code generates a 230 microsecond pulse, and a "10" code generates a 2 millisecond pulse.

3.5.2.2.6.2 PCM Encoder ON Discrete Command. The PCM Encoder ON command (i.e., PCM19 or PCMA9 for PCME 1; PCME29 or PCMB9 for PCME 2) is a discrete command routed through an input buffer to the

latch which controls the PCME power. This command sets the latch; a voltage is fed back from the regulator to keep the latch set and PCME power ON. The ON command to a PCME simultaneously turns OFF the other PCME.

3.5.2.2.6.3 PCM Encoder OFF Discrete Command. The PCM Encoder OFF command (i.e., PCM1Ø or PCMAØ) is a discrete command routed to an input buffer and to the latch which controls the PCME power. This command will reset the latch and thus turn PCME power OFF. The OFF command turns off both PCMEs at the same time.

3.5.2.3 Telemetry Processor. The TP controls all functions of the data handling subsystem. The TP requests telemetry data from the DIMs through the PCME, organizes the data into telemetry formats and transmits a PCM/PSK subcarrier to the spacecraft transmitters. The TP also interfaces with and controls the two (2) DSUs to provide data storage on the Orbiter spacecraft. All spacecraft timing is derived from the master clock contained in the TP.

The major functional components of the TP are timing, input interface, output interface, controller and power supply. In the following sections, each of these components are described.

3.5.2.3.1 Timing. A 2^{20} (1,048,576) Hz crystal oscillator with a 50 ppm stability is provided in the TP as a timing reference for the scientific instruments, spacecraft subsystems and TP. The crystal oscillator is used as the frequency standard for the timing block. A countdown chain in the timing block provides buffered 32,768 Hz, 2048 Hz and word rate signals to users.

The operating telemetry bit rate of the TP is selected by sending the TELEMETRY PROCESSOR CONTROL quantitative command (i.e., TPCQ1 for TP1 or TPCQA for TP2). Bits 29 through 32 of the uplink command format are coded to select the desired bit rate, as defined in PC-455 (Reference: Paragraph 1.5.1). This 4-bit code is sent to the timing block from the controller

which receives it from the command input logic (part of input interface). A change to the bit rate is implemented at the beginning of the next minor frame following receipt of the quantitative command. The timing block also transmits a signal twice the selected bit rate to the controller.

The countdown chain provides a word rate start signal to the controller to start the processing of a telemetry word. The controller completes the processing of one word in less than 3.9 milliseconds and goes into an idle state with the control logic strobed off to conserve power. The controller is reactivated by a new word rate start signal from the timing block. The telemetry word retrieved from a DIM during the current processing cycle is transmitted during the next telemetry word time.

The bit rate status signals (i.e., four (4) output signals giving the logic coding for the operating bit rate) are sent to the scientific instrument users from the timing block.

3.5.2.3.2 Input Interface. The input interface component consists of the command input logic block, the PCM input logic block and the DSU status input logic block. All input information is buffered through standard input buffers and all serial inputs are stored in shift registers which are interrogated by the controller.

The command input logic block receives the 16-bit (i.e., bits 25 through 40 of the uplink command format) TELEMETRY PROCESSOR CONTROL quantitative command (TPCQ1 or TPCQA) from the COM (i.e., COM 3 for TP1 and COM 4 for TP2), and stores it in a shift register. After the command has been completely received by the TP, a flip-flop is set in the command input logic block by the quantitative command verification pulse. The output of the flip-flop is sent to the controller. During internal word time zero (i.e., first word of internal minor frame to be transmitted in next telemetry word time as a TP output), the controller senses the state of the

flip-flop and transfers the contents of the shift register into the controller's Random Access Memory (RAM). As a result, the command will be acted upon in the next telemetry word time, which is the time of the first word of the minor frame outputted by the TP. If two (2) or more quantitative commands are sent in the period of a minor frame, only the last transmitted command will be processed at the beginning of the next minor frame.

The PCM input logic block receives PCM data inputs from whichever of the two (2) PCMEs is ON. The PCM interrogate logic block sends a telemetry channel address to the PCME and as a result, 8 bits of data are transferred to the PCM input logic block. The returning data is clocked in using a 32,768 Hz clock generated within the PCM interrogate logic block. The input data is stored in a shift register in the PCM input logic block within one telemetry word time slot, and is transferred to the data output logic block for output at the selected bit rate during the next telmetry word time slot.

The DSU status input logic block receives the rollover pulse and ON/OFF status from each of the two DSUs. The rollover pulse is a signal which indicates that the address counter in that DSU has transitioned from the maximum address to address zero. The rollover pulses are stored in flip-flops and the ON/OFF status of each DSU is sensed as a logic 1 or 0 input. The status is transferred to the controller upon command from the controller.

3.5.2.3.3 Output Interface. The output interface component contains the PCM interrogate logic block, the DSU interface logic block and the data output logic block. All outputs are bufferred through standard output buffers. The real time outputs of the data output logic block to the transponders which are required to be telemetry word rate synchronous, consist of a two (2) register pair located within the block. These are: 1) an internal register that is updated by the controller, and 2) an external register that

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receives the contents of the internal register by a synchronous transfer using the telemetry word rate clock.

Primary timing and control of the PCME is accomplished by the Manchester coded (biphase) address and instruction format provided by the PCM interrogate logic block. The address and instruction format consists of a 20-bit word sent at 2^{16} (65,536) bits per second. This 20-bit word contains an 8-bit sync pattern plus 1 bit used to shift the remaining 11 bits into a holding register. The first data bit instructs the PCME as to whether the requested telemetry data is analog or digital, the next five (5) data bits select the DIM that is to be interrogated and the last five (5) bits represent the channel address of the selected DIM. This 20-bit word is sent three (3) times in order to provide the proper timing for the PCME.

Data to be stored in either DSU is stored during the same telemetry word time slot that it is received from the PCME. Data is sent serially to the DSU from the DSU interface logic block and is accompanied by a store bit clock and a store word clock. Four (4) control lines are sent to the DSUs from the DSU interface logic block to control the DSU mode, i.e., ENABLE DSU1, ENABLE DSU 2, STORE/READ, and RESET.

The data output logic block is responsible for synchronizing the output data with the selected telemetry bit and word rates. Data to be sent to the transponder is loaded into a temporary internal shift register during the telemetry word time slot it is received from the PCME. At the start of the next telemetry word time slot, it is transferred into an external shift register which is serially clocked at the selected bit rate. The data may be convolutionally encoded, sent directly to the biphase modulator, or commanded off by use of the coding within the TP CONTROL quantitative command. When the data is convolutionally encoded, the convolutional encoder is reset after the last bit of the third sync word (i.e., 24th bit of the minor frame) is

shifted into the encoder. Data out of the convolutional encoder is clocked into the biphase modulator at 16,384 Hz in order to provide coherence with the subcarrier. After the data is biphase modulated, it is buffered to the spacecraft interface levels and sent to the transponder. The data output logic block also provides the 4096 second clock output to the command processor and the major and minor frame rate signals to the scientific instruments and spacecraft subsystems, as seen in Figure 3.5.1-1.

3.5.2.3.4 Controller. The controller is the core of the TP. Telemetry minor frame and subcommutated formats and control signals are stored within Programmable Read Only Memories (PROMs) contained in the controller. The timing, input interface and output interface functions are linked to the controller by a data bus over which data and control signals are sent. The controller also performs arithmetic and logic manipulations with data and stores pertinent data in the RAM.

Upon TP power turn-on (via TLM19 or TLMA9 for TP1; TLM29 or TLMB9 for TP2), the controller executes an initialization routine. The commandable functions initialize to the states shown in Table 3.5.2.3.4-1. After initialization, the controller goes into the idle state and waits for the telemetry word rate start signal from the timing block. Upon receipt of the telemetry word rate start signal, contents of internal registers are transferred to the external data registers that interface with units outside the TP. This assures that the output interface will be synchronous with the selected telemetry word rate. The spacecraft time code update is the first routine executed after receipt of the telemetry word rate start signal. The time code is incremented as a function of bit rate, i.e., at the highest bit rate (2048 Hz), the increment is 1/256 second and at the lowest bit rate (8 Hz), the increment is 1 second.

TABLE 3.5.2.3.4-1

TP INITIAL CONDITIONS

PARAMETER	INITIAL CONDITIONS
Bit Rate	16 Bits Per Second
Frame Format	Orbiter Engineering Format
DSN Control Logic State	Standby (per Reference: Paragraph 1.5.20)
Convolutional Encoder	ON
Data	ON
Subcarrier	ON
Time Code Word	4 Minutes 16 Seconds prior to Full Scale Clock Reading
Minor Frame Word	63
Minor Frame Number	63

If a new command has been received by the command input logic block and the internal telemetry word time slot is telemetry word zero, the command is transferred onto the data bus and stored within the RAM for later access. If, however, the Direct Memory Access (DMA) bit is set in the TP CONTROL quantitative command, the command is processed as a direct memory access command, i.e., the command is being used to configure the programmable format. At this point, the internal registers that control bit rate select and TP output functions are loaded so that the external registers can be loaded at the next telemetry word time slot, which will be telemetry word zero of the next minor frame. Also at this point, the

DSU control bits are masked out and stored in a specific RAM location for later use by the DSU control routine. If the current telemetry word is the last telemetry word of a major frame, the three (3) most significant time code words (i.e., three 8-bit words) are sampled and stored for later readout. Next, the various bits which represent minor frame rate, major frame rate, and the 4096 second rate are masked out of the counters which are stored in the RAM and are transferred via the data bus to the data output logic block.

The controller forms a 10 bit address for the minor frame format memory composed of 4 bits for the commanded format type and 6 bits for the contents of the telemetry word counter. A 12-bit word is then retrieved from the minor frame format memory. Four (4) bits of this word serve to classify this word into one of six types.

First, the "normal" type, which is the most prevalent, signifies that the remaining 8 bits are a DIM address and are to be sent to the PCME. The returning commutated telemetry word is placed in an internal register which will be transferred to an external output register at the next telemetry word time slot.

The second type of format word is the subcommutated word. Within the space that would normally be the DIM address bits, there are two (2) bits to indicate the subcom type. These two (2) bits are connected to the 6 bits of minor frame count to form an 8-bit address for the subcom format memory which is similar to the minor frame format memory. The 12-bit word retrieved from the subcom format memory is treated similarly to the 12-bit word retrieved from the minor frame format memory.

The third type of format word is the internal word. Five (5) of the eight (8) remaining bits are used to indicate an internal status word stored in the RAM to be outputted during the next telemetry word time slot.

The fourth type of format word is the sync word where the actual sync code is stored in the remaining 8 bits of the 12-bit format word.

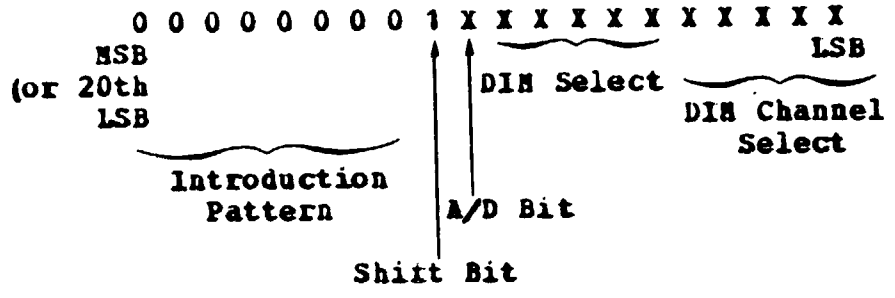
The fifth format word type is the programmable format word which contains a scratch-pad memory address used to fetch the programmed DIM and channel number.

The sixth format word type is the RAM-read word. When this word is encountered, the minor frame count is used as an address to fetch a word from the scratch-pad and insert it into the data stream.

3.5.2.3.5 Data Transfer

3.5.2.3.5.1 DIM to Telemetry Processor. The Telemetry Processor calls for data from a DIM by first initiating an interrogate cycle within the PCM Encoder. This interrogate cycle takes place entirely within the "busy" time segment of controller activity shown in Figure 3.5.2.3-1. The telemetry processor sends a 20-bit Manchester coded address from its PCM interrogate logic to the PCM Encoder's Manchester decoder circuitry. The 20-bit code is sent three successive times to create three subcycles as seen in Figure 3.5.2.3-2. However, the 20-bit address code is not usually the same for the first subcycle as it is for the second and third subcycles.

The first 20-bit address code consists of an 8-bit sync pattern, plus one bit to shift the remaining 11 data bits into a holding register:



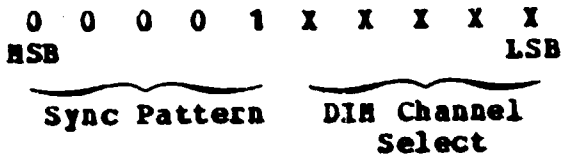
The 11 data bits consist of an A/D bit which indicates the type of telemetry data required, i.e., analog or digital; 5 DIM select bits (only 3 of the 5 are relevant to the Pioneer Venus Program which uses 8 DIMs); and 5 DIM channel select bits. The second and third 20-bit address codes differ from the first code only in that the DIM channel select bits are always coded as channel 16 (10000). All other bits are identical. In the event that channel 16 of a given DIM is being interrogated, then all bits of all three 20-bit address codes will be the same.

The second and third 20-bit address codes are generated for timing purposes only. Their DIM channel select bits are coded to channel 16 because channel 16 (as well as channels 17 through 31) doesn't create a read envelope which could cause a loss of synchronization, if a read envelope had already been generated during the first subcycle.

In the first interrogate subcycle, the PCM encoder's Manchester decoder circuitry receives the 20-bit address code serially shifted with MSB first. As the address code bits enter the Manchester decoder circuitry, they are used to create clocking signals. The Manchester decoder circuitry accumulates the data into a 12-bit shift register. The first 8 bits, i.e., the logical zeroes (sync pattern) pass straight through the register. The 12th LSB bit of the code, i.e., the logic "1" following the sync pattern, triggers the parallel transfer of the following 11 data bits to an 11-bit holding register. Upon data transfer, the 12-bit shift register is immediately reset.

Instantly after the 11-bit holding register receives the data, the 5 DIM select bits and the 5 DIM channel select bits are decoded by the PCM encoder's DIM decoder circuitry and address generator circuitry, respectively. This is possible because the contents of the 11-bit holding register, excluding the A/D bit, are hard-wired to the DIM decoder circuitry and the address generator circuitry.

The DIH decoder uses 3 of the 5 DIH select bits to activate an address line to the selected DIH. The address generator uses the 5 DIH channel select bits to encode a 10-bit address code. The 10-bit address code consists of a 5-bit sync pattern followed by the 5 DIH channel select bits, MSB first:



The address generator also receives a delayed 5-bit DIH select code for the DIH previously interrogated. This delayed 5-bit code is used to inhibit or enable the sending of a fill address which is generated in the address generator. The 5-bit DIH select code comes from a 6-bit holding register within the PCM encoder. However, the 5 DIH select bits from the 6-bit holding register and the fill address generated by the address generator are of minor importance with respect to the Pioneer Venus Program. In the Pioneer Venus Program, the fill address is derived directly from the 2nd 20-bit address code of the iterrogation cycle. It should be noted that the A/D bit is one of the delayed 6 bits from the 6-bit holding register. The A/D bit must be delayed for one subcycle period to properly instruct the PCM encoder's A/D converter circuitry.

At the midway point of the LSB of the 1st 20-bit interrogate subcycle (as seen in Figure 3.5.2.3-2), the first 10-bit address code generated by the address generator is serially shifted (MSB first) from the PCM encoder's DIH decoder to the selected Manchester decoder circuitry of the DIH.

In the 2nd interrogate subcycle, the PCM encoder's Manchester decoder circuitry receives the 2nd 20-bit address and generates clocking signals. The data bits are processed in the same

manner as in the 1st interrogate subcycle; but the 10-bit address code that is generated by the address generator will serve the purpose of the fill address.

At the MSB of the 3rd 20-bit interrogate subcycle, the fill address leaves the PCM encoder's DIM decoder and goes to the DIM's Manchester decoder. The fill address is used by the DIM to create clocking signals which will allow an 8-bit word of data to be clocked from the DIM to the PCM encoder.

If conditioned data is being sampled, a 1.000 mA precision current output (see Figure 3.5.2.3-2) is also sent from the PCM encoder to the selected DIM within 4 microseconds after the fill address leaves the PCM encoder. If serial digital data is being sampled, then the 50% rise time point of the read envelope (see Figure 3.5.2.3-3) would occur within the same 4 microseconds time span as the leading edge of the precision current output pulse. The read envelope and read clock (also in Figure 3.5.2.3-3) are sent to users by the DIM to transfer data into the DIM.

Within 24 microseconds after the 2nd MSB of the fill address leaves the PCM encoder, an 8-bit telemetry data word is clocked from the DIM (see Figure 3.5.2.3-2) to the PCM encoder.

The PCM encoder receives the PAM or digital data from the DIM, and regenerates the data in non-return-to-zero pulse code modulated (NRZ-L/PCM) form. The PCM encoder uses clocking signals generated from the 3rd 20-bit interrogate subcycle to serially shift the 8-bit telemetry data word (MSB first) to the telemetry processor (see Figure 3.5.2.3-2).

3.5.2.3.5.2 Telemetry Processor to DSU. Telemetry data received by the telemetry processor is routed to either the transponder or to the DSU or both by command. Data which is routed to the DSU is stored in the DSU within the same telemetry word time that it is received. The data is clocked into an internal shift register by the trailing

edge of the store bit clocks (see Figure 3.5.2.3-4). After eight store bit pulses, the trailing edge of the store word clock is used to activate an internal memory write cycle.

Data which is routed to the transponder is first stored in an internal register of the telemetry processor's data output logic. At the next telemetry word time, the data is transferred to an external register which interfaces directly with the transponder.

3.5.2.3.6 Power Supply. The power supply generates the voltage forms required by the internal TP functions. The power supply provides current limiting, undervoltage detection and shutdown, overvoltage detection and shutdown, negative voltage input clamping, and bus fusing. Power strobing is used to conserve energy. In addition, the power supply provides isolation between the bus return and the signal returns, and buffers the discrete input commands that turn the TP on and off. The TP power supplies receive three (3) discrete commands to control the ON/OFF status of the units, i.e., TP1 ON/2 OFF (TLM19 or TLMA9), TP2 ON/1 OFF (TLM29 or TLMB9) and TPs OFF (TLM1Ø or TLMAØ). The ON/OFF telemetry status of the power supplies is given by DTLM1S for TP1 and DTLM2S for TP2.

3.5.2.3.7 Internal Command Response. The TP receives both quantitative and discrete commands from the COMs to control the configuration and operation of the unit. The following sections describe the internal response of the TP to the different commands received.

3.5.2.3.7.1 TELEMETRY PROCESSOR CONTROL Quantitative Command. The 16-bit quantitative command (i.e., bits 25 through 40 of the uplink command format (TPCQ1 for TP1; TPCQA for TP2) used to configure the TP is stored in a temporary holding register (i.e., external register) in the command input logic block. After the command has been received by the command input logic, a flip-flop is set by the verification pulse. The controller senses the state of the flip-flop during the internal word zero input time and transfer the contents of the temporary holding register into the controller's RAM via the data bus. As a result, the latest TP CONTROL quantitative command will be acted upon at the start of the next minor frame. Information transfer is then accomplished via the data bus to an internal register by a synchronous transfer using the word rate signal. The information bits are decoded and used to control their specified functions.

Bit 36 of the 16-bit quantitative command is used to determine whether a direct memory access (DMA) operation is to take place. If the DMA bit is a logic "1", the TP CONTROL quantitative command is used to configure the programmable format. A logic "0" indicates a normal command. A programmable format command is loaded directly into the RAM, and contains information regarding DIM number, DIM channel, and data type of the telemetry signal(s) to be sampled. The structure of the 16 bits of data for the programmable format command is given in PC-455 (Reference: Paragraph 1.5.1).

The normal command consists of 15 information bits and one spare bit (Bit 37). Bits 25 through 28 are decoded to provide format selection, and bits 29 through 32 are decoded to provide bit rate selection. Bit 38 controls the subcarrier ON/OFF state; bits 39 and 40 control the Convolutional Encoder/Data modes. The status of these three functions are not provided in telemetry. The other three bits of the quantitative command (bits 33 through 35) are decoded by the DSU control logic to produce DSU control commands. Three of the commands are

standby commands (000, 001, 111). The standby commands are used to interrupt DSU read or store operations without affecting the address counter of the DSU. To reset the address counter to zero, the Standby Reset (110) command is used. Four other commands (Read DSU1 "010", Read DSU 2 "011", Store DSU1 "100", Store DSU 2 "101") are sent to configure the DSU to various operational states (see Figure 3.5.3.6-1, DSU State Diagram).

3.5.2.3.7.2 Telemetry Processor ON Discrete Command. The Telemetry Processor ON command (i.e., TLM19 or TLMA9 for TP1; TLM29 or TLMB9 for TP2) is a discrete command routed to the input of a latch which controls the TP power. This command sets the latch; a voltage is fed back from the regulator to keep the latch set and the TP power ON. The ON command to a TP is also used to simultaneously turn off the other TP.

3.5.2.3.7.3 Telemetry Processor OFF Discrete Command. The Telemetry Processor OFF command (i.e., TLM10 or TLMA0) is a discrete command routed through an input buffer to the latch which controls the TP power. This command will reset the latch and thus command the TP power OFF. This OFF command simultaneously turns off both TPs.

3.5.2.4 Data Storage Unit. Each DSU interfaces with the TPs to provide a data storage capability for the Orbiter spacecraft. Control of the read and write modes of the DSU are provided by the TP through the TP CONTROL quantitative command (i.e., TPCQ1 for TP1 and TPCQA for TP2). The DSU receives a store bit clock, store word clock, DSU enable and DSU mode signals, and data from the TP, and transmits an address rollover signal and ON/OFF status to the TP. DIMs 3 and 5 provide read clocks and read envelopes to read the contents of the DSUs. The major components of the DSU are timing and control logic, address counter, core stack, and power supply. These components are described in the following sections.

3.5.2.4.1 Timing and Control Logic. The timing and control logic block receives the store bit clock, store

word clock, DSU enable and DSU mode signals from whichever TP is powered ON. These control signals start the operating modes in the DSU, and once started, these modes are completed automatically by the timing and control logic circuitry. The store mode is initiated when the DSU enable and DSU mode signals are both in the logic 1 states. For the read mode, the DSU enable signal must be in the logic "1" state and DSU mode signal must be in the logic "0" state.

To store data, the timing and control logic enables the input data to an 8-bit shift register which is clocked in by the store bit clock. After eight store bit clock pulses, the trailing edge of the store word clock is used to initiate an internal memory store cycle. During this time, the address in the address counter is decoded to select the memory word location, i.e.,, one of 256 X-drive lines and one of 256 Y-drive lines are selected. The timing and control logic circuitry then applies drive line currents to the X- and Y-drive lines. The X- and Y-drive currents cause readout (i.e., clearing) of all cores in the selected word location. At the completion of the clear cycle, the data is read into the memory word slot through a high-speed register.

For the read operation, a memory read cycle is initiated with the leading edge of the read envelope. Like the store mode, the timing and control logic circuitry controls the X- and Y-drive currents to the core stack and the sensing and inhibit functions. Early in the read cycle, the high-speed register is reset (i.e., cleared). The address in the counter is then decoded causing the correct X- and Y-drive lines to be selected and energized. The eight cores located at the intersection of the selected X- and Y-drive lines are interrogated and the resulting signals appear on the sense lines at the input to the sense amplifiers. The output of the sense amplifiers is temporarily stored in the high speed register. The data is then restored in the memory and transferred to the output data

register where it is read into the DIMs via the read clocks.

3.5.2.4.2 Address Counter. The address counter is a 16-bit register that controls the read and write addresses to the memory core stack. The counter can be reset to zero, i.e., set to the first address location, by sending the TP CONTROL quantitative command (Standby Reset "110") to the TP. At the end of a store or read operation, the address counter increments by one and continues counting until the counter is filled and goes from the maximum address back to the first address. At that time, an address rollover pulse is transmitted, indicating to the TP controller that some action is required, such as sending a DSU disable or a DSU off signal to the DSU, or sending a DSU on signal and/or DSU enable to the other DSU. For the read operation, the trailing edge of the read envelope causes the address counter to advance one count.

3.5.2.4.3 Core Stack. The core stack is organized in the conventional 3-D, 3-wire, coincident current destructive readout (DRO) configuration. The basic magnetic storage element is a miniature (18-mil OD) temperature - independent ferrite core. The memory has a storage capacity of 65,536 words of 8-bits each, and has two operating modes, clear/write (store) and read/restore (read). In either the store or read mode, the 65,536 words in the memory are accessed sequentially under the control of an address counter at the maximum in-flight rate of one 8-bit word every 3,906 microseconds. No memory operations are performed during power-up or power-down, so that stored data will be retained during power sequencing.

3.5.2.4.4 Power Supply. The DSU uses the unregulated +28 Vdc switched loads bus for operating power. This power supply is a commandable ON/OFF supply. The discrete commands used to control the DSU power supplies are as follows: DSU1 ON (DSU19 or DSUA9); DSU2 ON (DSU29 or DSUB9); and both DSU1 OFF and 2 OFF (DSU1Ø or DSUAØ). Telemetry status is given by Word 8 of the telemetry formats

containing the DSU status word. The power supply provides power to timing and control, address counter and circuits required for the ready condition. In order to reduce power, power switching circuits are provided which keep the memory in a normally off (low power) state, except for the very short period (10 microseconds out of 3,900 microseconds) when the memory is activated by a store or read command. The memory contents will not be altered when power is turned on or off.

3.5.2.4.5 Internal Command Response. The DSU receives discrete commands from the COMs to control the ON/OFF state of the DSUs. The following sections describe the internal response of the DSUs to the discrete commands received.

3.5.2.4.5.1 Data Storage Unit On Discrete Command. The DSU ON command (i.e., DSU19 or DSUA9 for DSU1; DSU29 or DSUB9 for DSU 2) is a discrete command routed to the input of a latch which controls the DSU power. This command sets the latch; a voltage is fed back from the regulator to keep the latch set and DSU power ON.

3.5.2.4.5.2 Data Storage Units OFF Discrete Command. The DSUs OFF command (i.e., DSU1Ø or DSUAØ) is a discrete command routed through an input buffer to a switch in the input of the latch. This command will reset the latch and thus command the DSUs power OFF. The OFF commands turns the power off to both DSUs simultaneously.

3.5.3 Operational Description. The following sections describe in detail the operational characteristics of the data handling subsystem.

3.5.3.1 Programmable Format Operations. The TP contains one (1) minor frame format that can be programmed in-flight. Up to eight different analog, serial digital or bilevel words can be inserted into the programmable format by use of the TP CONTROL quantitative command. The primary purpose of this format is to increase the sample rate on single or multiple telemetry channels for spacecraft diagnostic purposes. However, it is

also useful for receiving selected telemetry channels during special mission maneuvers and/or events. The following sections describe the organization of the programmable format, the method for loading it, the verification of the format contents and other operational considerations.

3.5.3.1.1 Format Organization. The programmable format is organized into the standard 64 word minor frame format (see Figure 3.5.3.1.1-1) used by the other 15 minor frame formats on the Multiprobe and Orbiter spacecrafts. The first eight words of the 64 words are pre-programmed and cannot be changed. The first seven words (i.e., words 0 to 6) are standard to all 16 minor frame formats and the definition of these words is given in PC-454 (Reference: Paragraph 1.5.2). Word seven is unique to the programmable format. This word contains the full contents of the Random Access Memory (RAM) in the TP. The RAM contains 32 entries consisting of 8-bit words of information; eight of these entries are used to store the DIM address information associated with the eight programmed telemetry words. Section 3.5.3.1.3 will describe the contents of the RAM in detail.

The remaining 56 words in the format are used to program the selected telemetry words. These 56 words can be considered to contain a mini-minor frame that is eight words long and repeated seven times. That is, words 8 to 15, 16 to 23, 24 to 31, etc., all contain identically the same 8 word telemetry format. Each mini-minor frame can be programmed into any combination of telemetry words addressable through a DIM. Internal words, such as spacecraft time code (dynamic and static), format and bit rate status and minor frame count, cannot be selected for the programmable format. The eight words in the mini-minor frame can be all the same, all different, or any intermediate combination. The method for programming the format is explained in the next section.

Figure 3.5.3.1.1-2 shows an example of a programmable format. Bus Voltage Limiter Current

(PLINTI) has been assigned to alternate words in the 8 word format for the mini-minor frame (words 8, 10, 12, 14, 16, 18, etc., to word 62). ATTITUDE MEASUREMENT for ADP1 (ATTM1X) has been assigned to every eighth word, starting with word 9. ADP status has been assigned to every fourth word, starting with word 11. Finally, PSI* Brightness (A*1BRM) has been assigned to every eighth word, starting with word 13.

3.5.3.1.2 Programming and Starting the Format. The programmable format is configured by the use of those versions of the TP CONTROL quantitative command that are identified as the PROGRAMMABLE FORMAT commands in PC-455 (TPCQ2 and TPCQ3 for TP1; TPCQB and TPCQC for TP2). To initially set up the format, it takes nine (9) transmissions of the quantitative command to the TP. The contents of each command is stored into a different location in the RAM to be used later by the TP in the execution of the format. The order in which the commands are sent is not important because each command contains the RAM address into which the format information is stored.

The structure of the PROGRAMMABLE FORMAT versions of the TP CONTROL quantitative command are given in PC-455 (Reference: Paragraph 1.5.1), but will also be discussed here. Bits 25 through 40 of the uplink command format are used for all quantitative command data. For the PROGRAMMABLE FORMAT commands, bits 33 through 35 are not used and can either be logic "0" or logic "1" bits, but bit 36 must always be logic "1". Bit 36 is used to select TP CONTROL ("0") or PROGRAMMABLE FORMAT ("1") versions of the TP CONTROL command. The RAM addresses (i.e., 23 through 31) are defined in bits 37 through 40 and bits 25 through 32 are used in two different ways, depending on which type of PROGRAMMABLE FORMAT command is sent to the RAM.

The first type of command (TPCQ3 or TPCQC) is used to define whether the selected channels are analog (conditioned or unconditioned) or digital (serial digital or an 8-bit bilevel word). This command must be sent to RAM address 23, and the

contents of bits 25 through 32 must define whether the 8 programmed telemetry words are analog or digital. The eight command bits (i.e., 25 through 32) are assigned as follows to the eight telemetry words; bit 25 - word 3, bit 26 - word 4, bit 27 - word 5, bit 28 - word 6, bit 29 - word 7, bit 30 - word 0, bit 31 - word 1 and bit 32 - word 2. A logic "1" denotes an analog telemetry word and a logic "0" denotes a digital telemetry word. In summary, bits 25 through 32 define the type of telemetry word, bits 33 through 35 are not used, bit 36 must be a logic "1" and bits 36 through 40 must be binary 23 where bit 36 is the MSB, to properly transmit this first type of command.

The second type of command (TPCQ2 or TPCQB) is used to load the DIM address information associated with the selected telemetry words into the RAM. RAM addresses 24 through 31 are used to store the binary code for the DIM and the DIM channel that is assigned to the telemetry words to be programmed into the format. Programmed words 0 through 7 are assigned to RAM addresses 24 through 31 respectively. Bits 25 through 27 define the DIM and bits 28 through 32 define the DIM channel. Bits 25 and 28 are the MSBs for the two codes. Therefore, telemetry words 0 through 7 of the mini-minor frame are stored into RAM addresses 24 through 31 respectively. After all the RAM locations are filled and a change is required to a DIM or DIM channel address, only a single command with the corrected DIM information and its associated RAM address has to be sent to the TP, provided the new telemetry word is the same as the old telemetry word it replaces regarding analog or digital categories (otherwise, the corrected first command must precede this single command). The contents of the RAM are permanent for all time that power is applied to the TP.

Once RAM locations 23 through 31 are loaded with the correct information, the programmable format can be selected by using the TP CONTROL version of the TP CONTROL quantitative command. The code for the programmable format is "0011" and it must

be inserted in bit locations 25 through 28 of the uplink command format. The other codes in the quantitative command for format, bit rate, DSU control, etc., must be for the current status of the TP if only the format is being changed to the programmable format. The structure for the TP CONTROL version is given in PC-455 (Reference: Paragraph 1.5.1). The programmable format starts at the next minor frame after the format select command is sent.

3.5.3.1.3 Format Verification. The DIM address information loaded into the RAM locations, and consequently the telemetry channels programmed into the format, can only be verified through the use of the format itself. Minor frame telemetry word 7 contains a complete readout of the 32 RAM locations. The first entry in the RAM (location 0) is read out in minor frame 0 of the major frame and all the following RAM locations are read out in the following minor frames. Since there are 64 minor frames in a major frame, the complete RAM readout is accomplished twice in each major frame.

Figure 3.5.3.1.3-1 defines the contents of all RAM locations. Starting with minor frame 24 of the major frame, the eight programmed telemetry words can be verified. If an error is detected, a new command can be sent to the RAM to correct the contents. The command can be sent while the programmable format is in operation.

3.5.3.1.4 Operational Constraints and Considerations. There are a number of constraints and considerations when operating the programmable format. These are as follows:

(a) RAM locations 23 through 31 are random when the TP is first powered on. Therefore, the programmable format should not be commanded into use before loading the desired telemetry words.
(b) RAM locations 0 through 15 are inaccessable by command, so they cannot inadvertently be altered. However, locations 16 thourgh 31 can be accessed

by command. Location 16 contains the most significant word of the dynamic time code and therefore, care should be exercised to prevent this word from being changed by sending a command to this location. Locations 17 through 22 are spares but could be used to store information because they are addressable and are read out in the telemetry format.

(c) All commands to the TP when loading the RAM should be spaced by one minor frame. If two commands are received in the space of one minor frame, the first command will be ignored.

(d) The contents of the RAM are permanent while the TP is powered. Consequently, the programmable format can be used again after it has been programmed and other minor frame formats have been used in the interim.

3.5.3.2 Telemetry Format and Bit Rate Operations. There are 16 minor frame formats and 13 bit rates that can be commanded into operation on either the Orbiter or Multiprobe spacecrafts. Detailed information about format layout and overhead words for each format is contained in PC-454 (Reference: Paragraph 1.5.2). Either spacecraft will respond to a command containing any telemetry format and any bit rate. However, commonality between the two spacecrafts exists totally only for the bit rates. There are five (5) formats that operate properly on the Multiprobe and 14 on the Orbiter. Consequently, only three (3) formats are common, i.e., the ACS, Command Memory Readout and Programmable formats.

If a telemetry format designed for the Multiprobe is commanded into operation on the Orbiter, the TP will address the DIS channels in accordance with the Multiprobe pre-programmed format. As a result, the format will contain data that is meaningless to the ground software which is programmed to receive known telemetry parameters in each word of the commanded format. However, the data can be made useful, if necessary, by

using PC-454 (Reference: Paragraph 1.5.2) to compare the differences between the DIM assignments on the two spacecrafts. Therefore, caution should be exercised when commanding telemetry formats to insure that the correct formats are used on each spacecraft.

Information regarding TM data times is shown in Table 3.5.3.2-1.

The formats and bit rates are commanded into operation through the TP CONTROL quantitative command, (TPCQ1 or TPCQA, as explained in PC-455 (Reference: Paragraph 1.5.1). Bits 25 through 28 of the uplink command format are used to code the telemetry formats and bits 29 through 32 are used for the telemetry bit rates. Telemetry verification is through word 3 of all 16 telemetry minor frame formats. The first four (4) bits of this 8-bit word gives the status of the format selected and the remaining four (4) bits gives the status of the bit rate selected. The 4-bit codes in this format word are identical to the codes used to command the format and/or bit rate into operation.

One constraint on the use of this command is that two different quantitative commands should not be sent to the TP in the space of one minor frame time. Since the commands are loaded into a shift register when received and held until the next minor frame, only the last command will be executed.

A further consideration on the use of this command is that bits 33 through 35 control the operation of the DSUs, and there is not a "no action" code for the DSU control logic. This means that the current operating mode of the DSU must be known when changing formats or bit rates. If both DSUs are off, there is no concern, but if either or both are on, the code corresponding to the operating mode of the DSUs must be used in the quantitative command to prevent a possible costly mode change when changing bit rates and formats.

TABLE 3.5.3.2-1

PIONEER VENUS TELEMETRY DATA TIMES

Bit Rate (bps)	Minor Frame Period (sec)	Major Frame Period	Minor Frame Word Rate (bps)	Orbiter Memory Storage Time	Orbiter Memory Readout Time	
					Playback Format	Data Memory Readout Format
4096	1/8	8sec	64	4min16sec	6min50sec	4min58sec
2048	1/4	16sec	32	8min32sec	13min39sec	9min56sec
1024	1/2	32sec	16	17min4sec	27min18sec	19min52sec
682-2/3	3/4	48sec	10-2/3	25min36sec	40min57sec	29min48sec
512	1	1min 4sec	8	34min 8sec	54min37sec	39min44sec
341-1/3	1-1/2	1min36sec	5-1/3	51min12sec	1hr21min54sec	59min35sec
256	2	2min 8sec	4	1hr 8min16sec	1hr49min14sec	1hr19min27sec
170-2/3	3	3min12sec	2-2/3	1hr42min24sec	2hr43min48sec	1hr59min10sec
128	4	4min16sec	2	2hr16min32sec	3hr38min28sec	2hr38min53sec
64	8	8min32sec	1	4hr33min 4sec	7hr16min56sec	5hr17min45sec
32	16	17min 4sec	1/2	9hr 6min 8sec	14hr33min52sec	10hr35min30sec
16	32	34min 8sec	1/4	18hr12min16sec	29hr 7min44sec	21hr11min00sec
8	64	1hr 8min16sec	1/8	36hr24min32sec	58hr15min28sec	42hr22min00sec

Data Storage Capacity = 1,048,576 bits.

Playback Format = 40 Stored Data Words.

Data Memory Readout Format = 55 Stored Data Words

Another consideration is that the status of the Convolutional Encoder and Data control bits (bits 39 and 40 of the uplink command word) must be inserted in the quantitative command to insure a mode change does not take place, such as commanding OFF the data output of the TP (Reference: Paragraph 1.5.17).

3.5.3.3 PCM Encoder Operations. The PCMEs are turned ON and OFF by three discrete commands, i.e., PCM ENCODER 1 ON/2 OFF (PCM19 or PCMA9), PCM ENCODER 2 ON/1 OFF (PCM29 or PCMB9) and PCM ENCODERS OFF (PCM1Ø or PCMAØ). When one PCM is commanded ON, the other PCM is simultaneously commanded OFF. This prevents the telemetry data from being scrambled by two PCMEs responding to the instructions from the operating TP.

At initial spacecraft turnon, a Telemetry Processor should be turned ON before the PCM Encoder is turned ON. If, subsequently, the TP is turned OFF and then ON, or the TPs are switched, a DIM ON status should be re-established by either a DIM configuration command, or by turning the PCM Encoder OFF and ON again to establish initial conditions (Reference: Paragraph 1.5.17).

3.5.3.4 Spacecraft Time Code Operations. The spacecraft time code consists of a 24-bit code (DCLOK1, DCLOK2 and DCLOK3) that is inserted into the first three 8-bit words of Subcom A of all the 16 minor frame formats. The time code has a resolution of 125 milliseconds and can count time for 24.27 days. At the beginning of each major frame, the time code is updated and frozen for the entire major frame. The major frame, as it exits the TP, is delayed 30.5 microseconds relative to Universal time. The accuracy of the time code is a function of the 50 ppm stability of the TP master oscillator.

The spacecraft time code also appears in the ACS, Engineering and Command Memory Readout formats. The 8 MSBs of the time code (DCLOK1) appears in the Command Memory Readout format, the 8 LSBs (DCLOK3) appear in the ACS format and the 16 LSBs

(DCLOK2 and DCLOK3) are in the Engineering format. Originally, these time code entries were intended to be the dynamic time code that is updated on a telemetry word basis by the TP, but through a programming error, these time code words are the same as those in Subcom A. Therefore, it should be recognized that these time code words in the minor frame do not provide any additional time code information.

3.5.3.5 4096 Second Clock Operations. The TP generates a squarewave clock with a period of 4096 seconds that is used as an accurate timing reference by the command processor to start the command memory. When the Stored Command Logic (SCL) within the COMMAND PROCESSOR is commanded into the TIMED START mode, the SCL waits until the next one-to-zero transition of the 4096 second clock and then starts processing the contents of the command memory.

The 4096 second clock is derived in the TP from the 24-bit time code that is inserted into the first three words of Subcom A of all the minor frame formats. The second MSB of DCLOK2 is used as the clock, i.e., the bit that changes state every 2048 seconds. The time code is updated at the beginning of each major frame. Therefore, the occurrence of the 4096 second clock can be resolved to the time of one major frame without any special procedures. At 2048 bps the resolution is 16 seconds. To improve this resolution to one minor frame time, a command can be inserted into the command memory that changes the state or mode of a parameter telemetered in the minor frame. By doing this, the resolution can be improved to 250 milliseconds at 2048 bps.

There are three errors associated with the use of the 4096 second clock that are inherent in the way the clock is implemented in the TP and used by the SCL. The errors are the 125 millisecond resolution in starting the SCL, the bit rate dependance of the clock output in that the 24-bit time code is updated on a word basis, and the stability of the command processor which effects the execution of events when time delays are

programmed in the command memory. These errors are discussed in detail in Section 3.6.3.2.3.2.

3.5.3.6 DSU Operations. The two (2) DSUs are controlled by the TP through the TP CONTROL quantitative command. The DSU state diagram is given in Figure 3.5.3.6-1. There are six (6) states in the diagram that are commandable by the TP quantitative command; one state (i.e., standby) has three commands associated with it. There are no restrictions on commanding from one state to another state. The new state will commence at the beginning of the next minor frame after receipt of the state change command. The state diagram shows, by arrows, all the automatic state changes; i.e., without commands. Also shown in Figure 3.5.3.6-1 are the command codes required in the TP CONTROL quantitative command to obtain the desired state.

The following sections describe the procedures to start, store into and read out of the DSU, and other constraints and/or considerations for using the units.

3.5.3.6.1 Starting the DSU. The control of the two (2) DSUs is through the internal logic in the TP and the TP CONTROL quantitative command. However, the ON/OFF control of each DSU is accomplished by discrete commands to the DSUs. Therefore, the minimum preparation required to store or read with the DSUs, is to have one TP operational and command on one DSU. The TPs are controlled by the following discrete commands; TP1 ON/2 OFF (TLH19 or TLHA9), TP2 ON/1 OFF (TLH29 or TLHB9) and TPs OFF (TLH1Ø or TLHAØ). One TP must be on in order to have a telemetry downlink. The ON/OFF control of the DSUs is through three (3) discrete commands; i.e., DSU1 ON (DSU19 or DSUA9), DSU2 ON (DSU29 or DSUB9), and both DSU1 OFF and 2 OFF (DSU1Ø or DSUAØ). When a DSU is commanded on, it provides an indication to the TP that it is on and can be used for reading or storing.

When a TP is first powered on, it initializes the DSU control logic into the standby state. In

this state the DSU is in the standby mode and the memory address pointer may be pointing at any memory cell location. To set the pointer to the first memory cell of each DSU, the Standby Reset command must be executed. There is no telemetry status of the memory address pointer.

After one or both of the DSUs are commanded on, the read or store status can be entered by sending the appropriate command code in the TP CONTROL quantitative command. If a transfer is made from one TP to the other when the first TP is in any other state but Standby Reset (DSU state 6), any read or store operation in progress will continue until the completion of the full telemetry word and then stop. The new TP will begin operating in the Standby state (DSU state 1). The address counter will be unaffected by the transfer to the alternate TP, so that it would be easy to renew the store or read operation at the point left off when the transfer was made to the new TP.

The read or store operation is started by inserting the correct command code in the TP CONTROL version of the TP CONTROL quantitative command (TCPQ1 or TPCQA). To read, code "010" (for DSU 1) or "011" (for DSU 2) must be placed in bit locations 33, 34 and 35 of the uplink command format. Likewise for the store operation, code "100" (for DSU 1) or "101" (for DSU 2) is required in these locations of the uplink command format. The read or store operation will begin at the start of the next minor frame after receipt of the command. The other 13 bits in the data portion of (TPCQ1 or TPCQA) must be set to the correct codes corresponding to the present status of the TP (i.e., bit rate, format, etc.).

3.5.3.6.2 Store Operation. The store operation can begin from Standby Reset (DSU state 6), standby (DSU states 0, 1, 7) or either of the two Read states (DSU states 2, 3). To begin storing at the beginning of either (or both) DSUs, the TP must be in the Standby Reset state. If both DSUs are to be used, the two DSUs must be commanded ON and

the Store 1 (100) command sent to the operational TP. When DSU 1 is completely filled, it will automatically transition to DSU 2 and continue storing. Upon completion of storing in DSU 2, the TP will revert to the Standby state. If storing is started in DSU 2 instead of DSU 1, the same result occurs upon completion of storing in DSU2: the TP reverts to the standby state, and no storing takes place in DSU 1. If only one DSU is to be used for storing, the selected DSU must be commanded ON and the appropriate command code sent to the TP (i.e., "100" for DSU 1 or "101" for DSU 2). A store command sent to a DSU that is OFF will cause the TP to return to the Standby state. The telemetry status for the DSU states is given in Word 8 of all the Orbiter telemetry formats, except the Command Memory Readout and ACS formats. The telemetry code corresponds to the binary number for the DSU states.

When the DSU is storing and a read command is sent to the TP, the store operation will terminate and the read operation will begin with the next address location in the DSU being asked to read. For example, if DSU 1 is full and storing is taking place in DSU 2, and a Read 1 command is sent, DSU 2 will stop storing and the read operation will begin with address 0 of DSU 1. When a DSU fills to capacity, the address counter resets to zero. If instead, a Read 2 command is sent, DSU 2 will stop storing and, at the very next address, the read operation will begin. Therefore, the old data in the remainder of the DSU 2 will be read out.

Any data that can be telemetered via any TM format can also be stored.

Storing can be accomplished in contiguous segments as long as no read operation takes place between the store operations. The address pointer is not telemetered, and there is no way to return to the address where storing last stopped unless no read operation has taken place. If a read operation takes place after a store operation on the same DSU, the address pointer

will be moved and the next store operation will not be contiguous with the former.

The Telemetry Processor can be commanded into the store mode while simultaneously operating in either the Playback or Data Memory Readout formats. However, the telemetry output will be all zeroes (Reference: Paragraph 1.5.17).

3.5.3.6.3 Read Operation. The read operation is analogous to the store operation. From Figure 3.5.3.6-1, the DSU state diagram, it can be observed that the conditions for automatic transitions for the read states are identical to those for the automatic transitions for the store states. As for the store operation, the read operation should begin by commanding the DSU control logic into the Standby Reset state (DSU state 6) and both DSUs on, if the full contents are to be read out. Again, the status of the DSUs is given in word 8 of the aforementioned minor frame formats.

The main difference with the read operation is that one of the two DSU readout minor frame formats, i.e., Playback or Data Memory Readout, must be commanded into operation in order to read the contents of the memory. The Data Memory Readout format is used exclusively for reading out the DSU memory but the Playback format contains 15 telemetry samples of scientific instrument data interspersed with the DSU readout data. Before the format is selected, the Read 1 command, "010", in the TP CONTROL quantitative command, must be sent to place the DSU control logic in the Read 1 state (DSU state 2). When the desired format is selected, the DIMs will transmit read envelopes and read clocks to the DSUs under control of the TP to read out the contents of the memory. The format can be selected before placing the DSU control logic in the Read 1 state. For this situation, the telemetry format will contain all zeroes until the read state is commanded on.

The same conditions for automatic transfer, from one read state to another or to standby, exists for the read operation as well as the store

operation. Also, contiguous readout of segments of the memory can be accomplished from a single DSU if a store operation is not performed in the middle. The rollover pulse for the read operation is an indication that the full contents of each individual DSU has been read out. Readout does not erase the contents of either DSU.

3.5.3.6.4 Operational Constraints and Considerations. The operational constraints and considerations for the DSU are given as follows:

(a) The address counter does not appear in telemetry. The only control of the address counter, is through the Standby Reset command or the DSU ON/OFF discrete commands. Commanding the DSUs ON will reset the counters to zero.

(b) All states in the DSU state diagram can be commanded directly by command. Therefore, it is possible to store over old data that is still needed by inadvertently sending the store command to a DSU in the middle of a read operation.

(c) All DSU commands must be spaced by a minimum of one minor frame or the first command will be ignored.

(d) The Playback or Data Memory Readout telemetry formats must be commanded into operation to read the contents of the memories.

(e) There is not a "no action" state for the DSU. Therefore, the current operating mode must be placed in the quantitative command when changing formats, bit rates, etc.

(f) The DSU can store without the telemetry subcarrier ON and with data commanded OFF. This permits storing real time scientific instrument data while the subcarrier signal is commanded off and/or data is OFF to satisfy requirements for occulation experiments or other events.

(g) Commanding the DSUs into Store and the TP into the Playback or Data Memory Readout formats will cause the telemetry formats to be stored in the DSUs with zeros for telemetry words. The old data in the DSUs will be lost because of this store operation.

(h) It has been learned from pre-launch system test that the DSU cannot be commanded ON after being turned OFF in less than 0.5 second; i.e., the time delay between the DSU OFF command and DSU ON command must be ≥0.5 second.

3.5.4 Command Response. Table 3.5.4-1 presents the command responses for the data handling subsystem. The table lists every command that directly effects the subsystem and the telemetry indication that verifies the proper execution of the command. The mnemonics for each of the command and telemetry parameters are also included.

TABLE 3.5.4-1

COMMAND RESPONSES FOR DATA HANDLING SUBSYSTEM

COMMAND			TELEMETRY VERIFICATION	
Mnemonic	Title	Response	Mnemonic	Indication and/or Remarks
Telemetry Processor Control (TPCQ1 for TP1; TPCQA for TP2) FMT = XXXX (Format Select)	Orbiter Engr (0000) Bus Engr (0001) Bus Entry (0010) Programmable (0011) ACS (0100) Command Memory Readout (0101) Playback (0110) Data Memory Readout (0111) Launch/ Cruise (1000) Periapsis A (1001) Periapsis B (1010) Periapsis C (1011) Periapsis D (1100) Periapsis E (1101) Apoapsis A (1110) Apoapsis B (1111)	The format select command selects one of the mission selectable formats. Execution of the selected format change is minor frame synchronous.	Internal Word 3	Format: Orbiter Engineering (0000) Bus Engineering (0001) Bus Entry (0010) Programmable (0011) ACS (0100) Command Memory Readout (0101) Playback (0110) Data Memory Readout (0111) Launch Cruise (1000) Periapsis A (1001) Periapsis B (1010) Periapsis C (1011) Periapsis D (1100) Periapsis E (1101) Apoapsis A (1110) Apoapsis B (1111)
Telemetry Processor Control (TPCQ1 or TPCQA) BR = XXXX (Bit Rate Select)	8 bps (0000) 16 bps (0001) 32 bps (0010) 64 bps (0011) 128 bps (0100) 170 2/3 bps (0101) 256 bps (0110) 341 2/3 bps (0111) 512 bps (1000) 682 2/3 bps (1001) 1024 bps (1010) 2048 bps (1011) 4096 bps (11xx) (where x = don't care bits.)	The bit rate select command selects one of 13 mission selectable bit rates. Execution of the selected bit rate change is minor frame synchronous.	Internal Word 3	Bit Rate: 8 bps (0000) 16 bps (0001) 32 bps (0010) 64 bps (0011) 128 bps (0100) 170 2/3 bps (0101) 256 bps (0110) 341 2/3 bps (0111) 512 bps (1000) 682 2/3 bps (1001) 1024 bps (1010) 2048 bps (1011) 4096 bps (11xx) (where X = don't care bits equal to that which was commanded).

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TABLE 3.5.4-1 (Continued)

COMMAND			TELEMETRY VERIFICATION	
Mnemonic	Title	Response	Mnemonic	Indication and/or Remarks
Telemetry Processor Control (TPCQ1 or TPCQA) DSU = XXXX (DSU Control)	Standby Reset (110)	Execution of all DSU control commands are minor frame synchronous. The standby reset command resets the address counter to zero and places the DSU in Standby.	Internal Word 8	State: Standby Reset (110)
	Standby (000, 001, 111)	The standby command places the DSU in Standby.	"	State: Standby (000, 001, 111)
	Read DSU 1 (010)	The Read DSU 1 command places the DSU in the read state. A Data Memory Readout or Playback format will then readout the DSU, beginning with the memory word slot to which the address counter is set.	"	State: Read DSU 1 (010)
	Read DSU 2 (011)	Same as above, except DSU 2.	"	State: Read DSU 2 (011)
	Store DSU 1 (100)	The Store DSU 1 command will store data into DSU1, beginning with the memory word slot to which the address counter is set.	"	State: Store DSU 1 (100)
	Store DSU 2 (101)	Same as above, except DSU 2.	"	State: Store DSU 2 (101)

TABLE 3.5.4-1 (Continued)

COMMAND			TELEMETRY VERIFICATION	
Mnemonic	Title	Response	Mnemonic	Indication and/or Remarks
Telemetry Processor Control (TPCQ1 or TPCQA) SUBC = E/D (Subcarrier Control)	Subcarrier ON/OFF ON: 1 OFF: 0	The subcarrier ON/OFF command controls the state of the subcarrier output.	N/A	No telemetry indication; check RF carrier for absence or presence of subcarrier.
Telemetry Processor Control (TPCQ1 or TPCQA) CONV = E/D, Data = E/D Conv. Enc. Data	Convolutional Encoder/Data - Unencoded Data (11 or 10) Encoded: 01 Data OFF: 00.	The convolutional encoder bypass/encode command bypasses or encodes the telemetry data and controls the modulation on the subcarrier prior to transmission.	N/A	No telemetry indication; proper operation with or without convolutional decoder. If data has been commanded OFF, loss of frame sync lock occurs.
Telemetry Processor Control (TPCQ2 or TPCQB) DIM = 0 - 7 (Progr. Format; DIM Select)	DIM Select (0 to 7) (000 to 111)	Execution of all programmable Mode Select commands are minor frame synchronous.	N/A	Same as commanded code; appears in RAM playback in minor frame format word 7.
	DIM Channel Select (0 to 31) (00000 to 11111)	The DIM and channel select commands select one of eight telemetry signal to be dwelled on. Any combination of up to eight telemetry words can be selected.	N/A	Same as commanded code; appears in RAM playback in minor frame format word 7.
Telemetry Processor Control (TPCQ2 or TPCQB) RAM = 24 - 31 (Progr. Format; RAM Address).	RAM Address Location (24 to 31) (11000 to 11111)	Addresses for Programmable words 0 to 7.	N/A	No telemetry indication; verification of data above verifies correct address location.

TABLE 3.5.4-1 (Continued)

COMMAND			TELEMETRY VERIFICATION	
Mnemonic	Title	Response	Mnemonic	Indication and/or Remarks
Telemetry Processor Control (TPCQ3 or TPCQC) RXXX=D/A (Progr. Format: Digit. or Analog Select)	Analog/Digital Data Analog: 1 Digital: 0	This command specifies whether each of the 8 programmable words are analog or digital.	N/A	Same as commanded code; appears in RAM playback in minor frame word 7.
	RAM Address Location (23) 10111	Address for Analog/Digital data specification.	N/A	No Telemetry indication; verification of data above verifies correct address location.

TABLE 3.5.4-1 (Continued)

COMMAND			TELEMETRY VERIFICATION	
Mnemonic	Title	Response	Mnemonic	Indication and/or Remarks
PCM19 PCMA9	PCM Encoder 1 ON/2 OFF	Discrete command which will turn PCM Encoder 1 ON and PCM Encoder 2 OFF.	DPCM1S DPCM2S	PCM Encoder 1 ON/OFF: ON (Logic 1) PCM Encoder 2 ON/OFF: OFF (Logic 0)
PCM29 PCMB9	PCM Encoder 2 ON/1 OFF	Discrete command which will turn PCM Encoder 2 ON and PCM Encoder 1 OFF.	DPCM2S DPCM1S	PCM Encoder 2 ON/OFF: ON (Logic 1) PCM Encoder 1 ON/OFF: OFF (Logic 0)
PCM10 PCMA0	PCM Encoders OFF	Discrete command which will turn both PCM Encoders OFF.	N/A	Loss of all Downlink Telemetry.
PCM Encoder DIM Control PCMQ1 PCMQ2	DIM 0 DIM 1 DIM 2 DIM 3 DIM 4 DIM 5 DIM 6 DIM 7 Coding: ON: 00 or 10 OFF: 01 or 11	Quantitative command used to turn off a failed DIM. In normal operation, all DIMs are automatically turned on two major frames after receipt of the PCM Encoder ON command.	N/A	No DIM telemetry status; correct telemetry from all DIMs in the ON condition. Telemetry from OFF DIMs is indeterminate.
TLM19 TLMA9	Telemetry Processor 1 ON/2 OFF	Discrete command which will turn Telemetry Processor 1 ON and Telemetry Processor 2OFF	DTLM1S DTLM2S	Telemetry Processor 1 ON/OFF: ON (Logic 1) Telemetry Processor 2 ON/OFF: OFF (Logic 0)
TLM29 TLMB9	Telemetry Processor 2 ON/OFF	Discrete command which will turn Telemetry Processor 2 ON and Telemetry Processor 1 OFF.	DTLM2S DTLM1S	Telemetry Processor 1 ON/OFF: OFF (Logic 0) Telemetry Processor 2 ON/OFF: ON (logic 1)
TLM10 TLMA0	Telemetry Processors OFF	Discrete command which will turn both Telemetry Processors OFF.	N/A	Loss of all downlink telemetry.

TABLE 3.5.4-1 (Continued)

COMMAND			TELEMETRY VERIFICATION	
Mnemonic	Title	Response	Mnemonic	Indication and/or Remarks
DSU19 DSUA9	DSU 1 ON	Discrete command which will turn DSU 1 ON.	Internal word 8	DSU 1 Status: ON (Logic 1)
DSU29 DSUB9	DSU 2 ON	Discrete command which will turn DSU 2 ON.	"	DSU 2 Status: ON (Logic 1)
DSU10 DSU A0	DSU 1 and 2 OFF	Discrete command which will turn both DSU's OFF.	"	DSU 1 Status: OFF (Logic 0) DSU 2 Status: OFF (Logic 0)

This Figure is a Foldout.

See APPENDIX C

Figure 3.5.1-1.

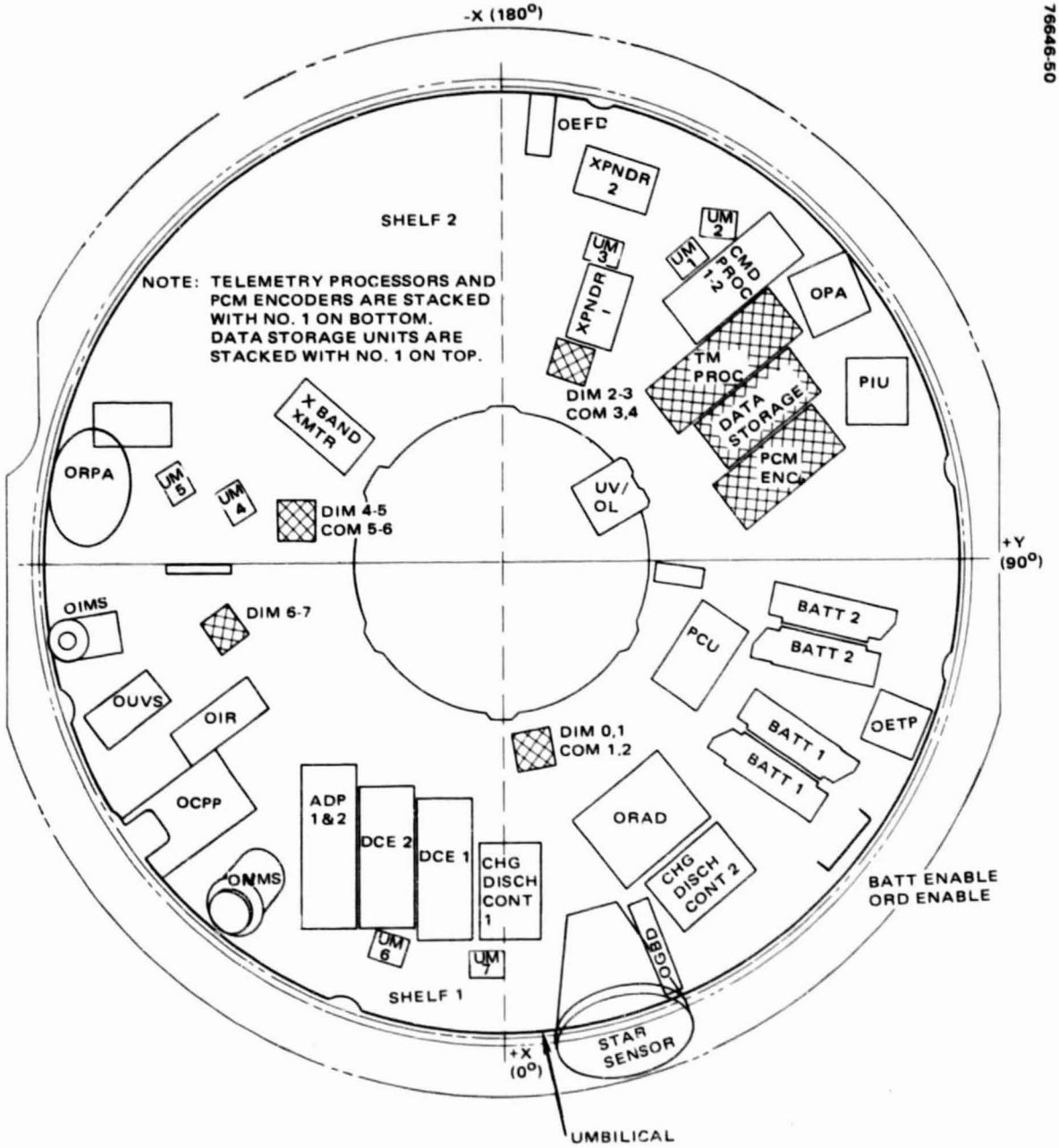


FIGURE 3.5.1-2. SHELF LAYOUT FOR DATA HANDLING SUBSYSTEM

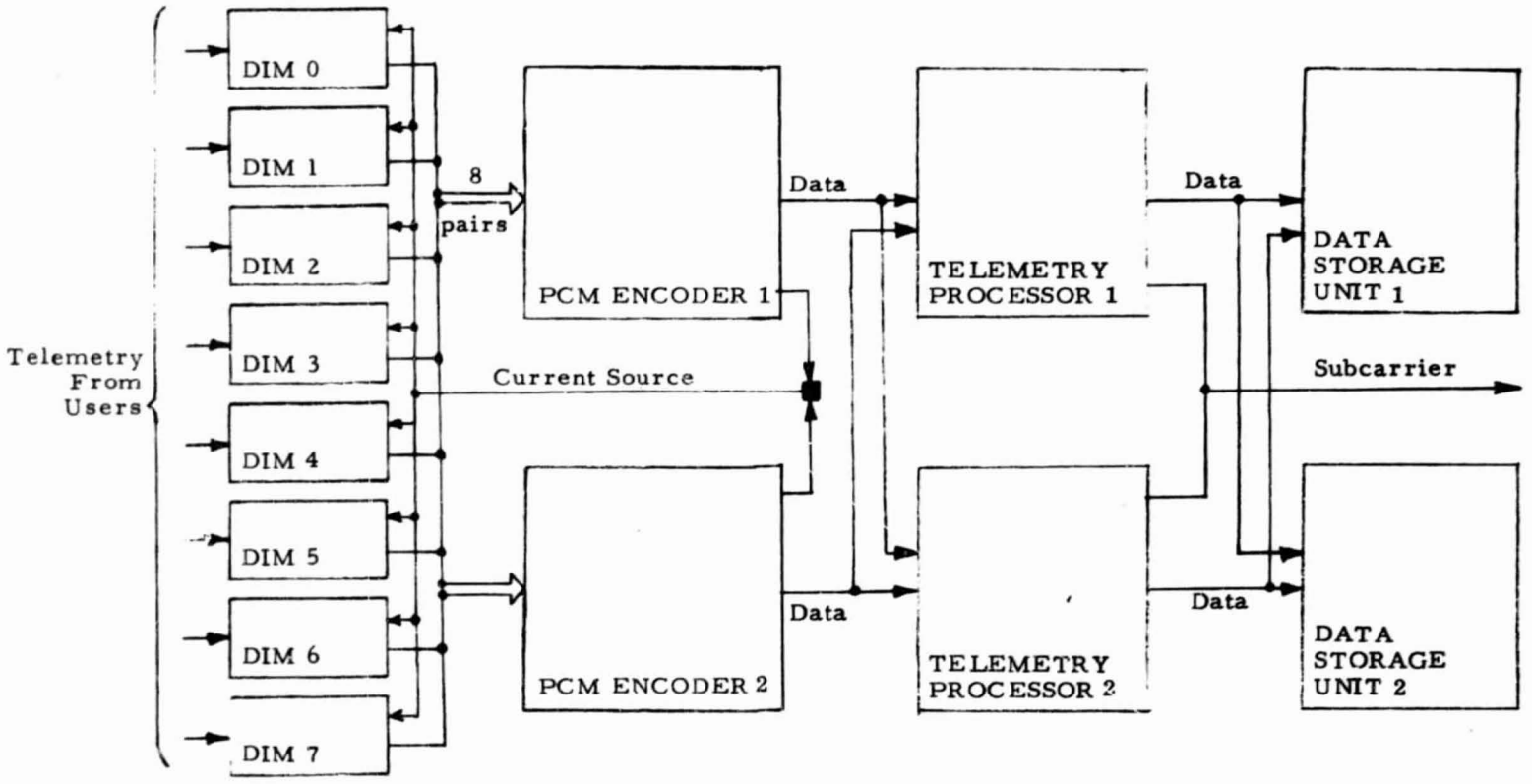


Figure 3.5.1-3. Orbiter Data Handling Subsystem Redundancy Block Diagram

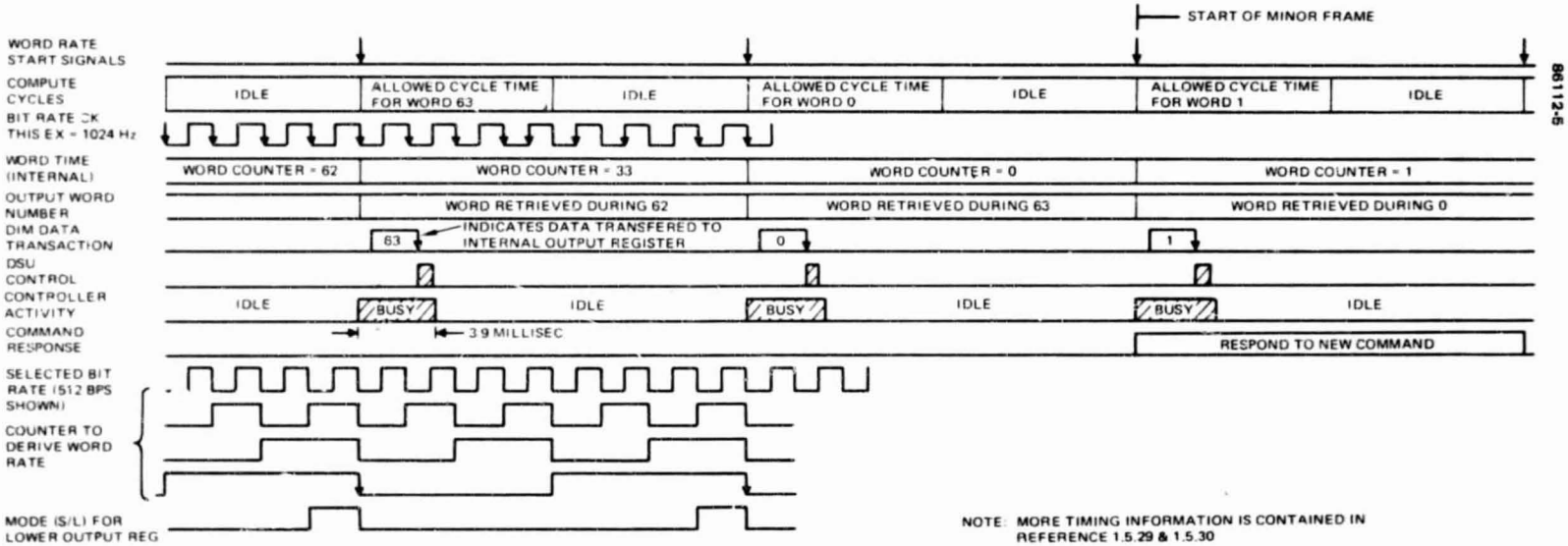


Figure 3.5.2.3-1. Telemetry Processor Timing

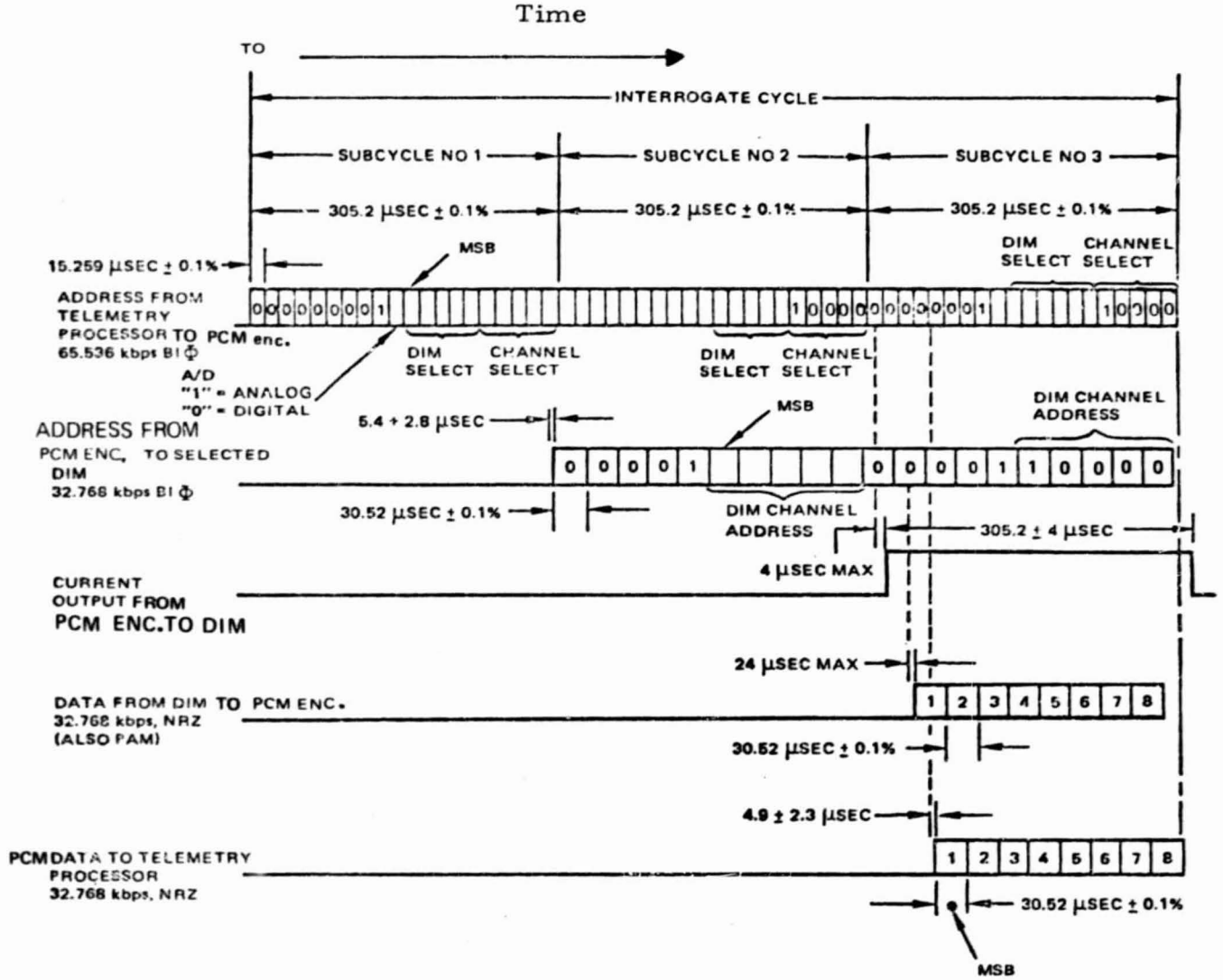
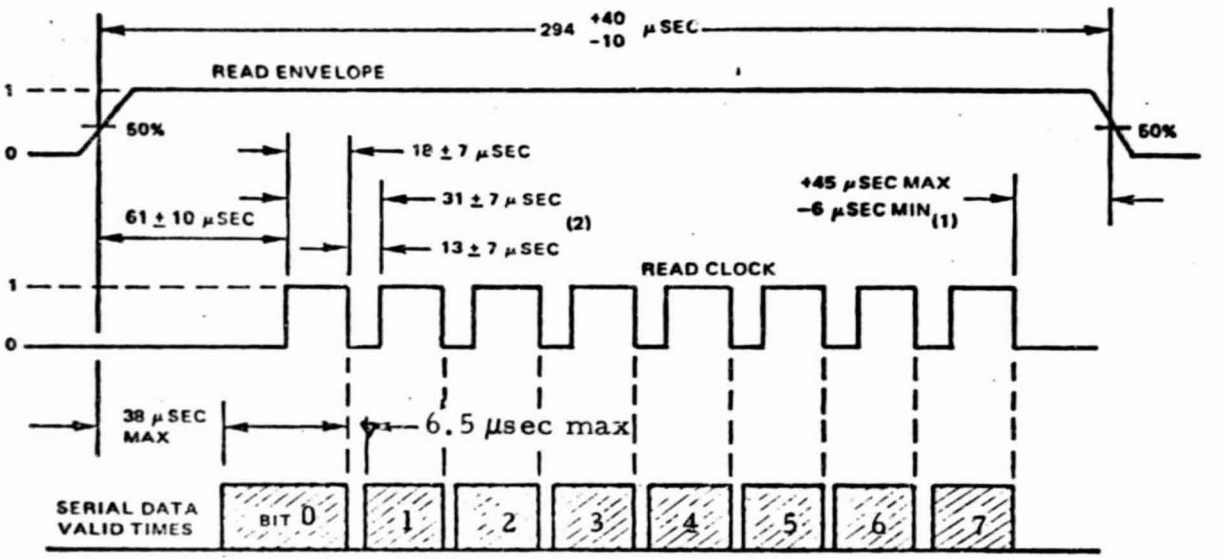


Figure 3.5.2.3-2. PCM Encoder Timing Diagram



NOTES: (1) READ ENVELOPE FALL TIME SHALL NOT EXCEED 1 μSEC FOR THE -6 μSEC TOLERANCE CONDITION

(2) TOLERANCES ARE NOT ACCUMULATIVE FOR ADJACENT CLOCK PULSES

Figure 3.5.2.3-3. DIM Serial Digital Timing Diagram

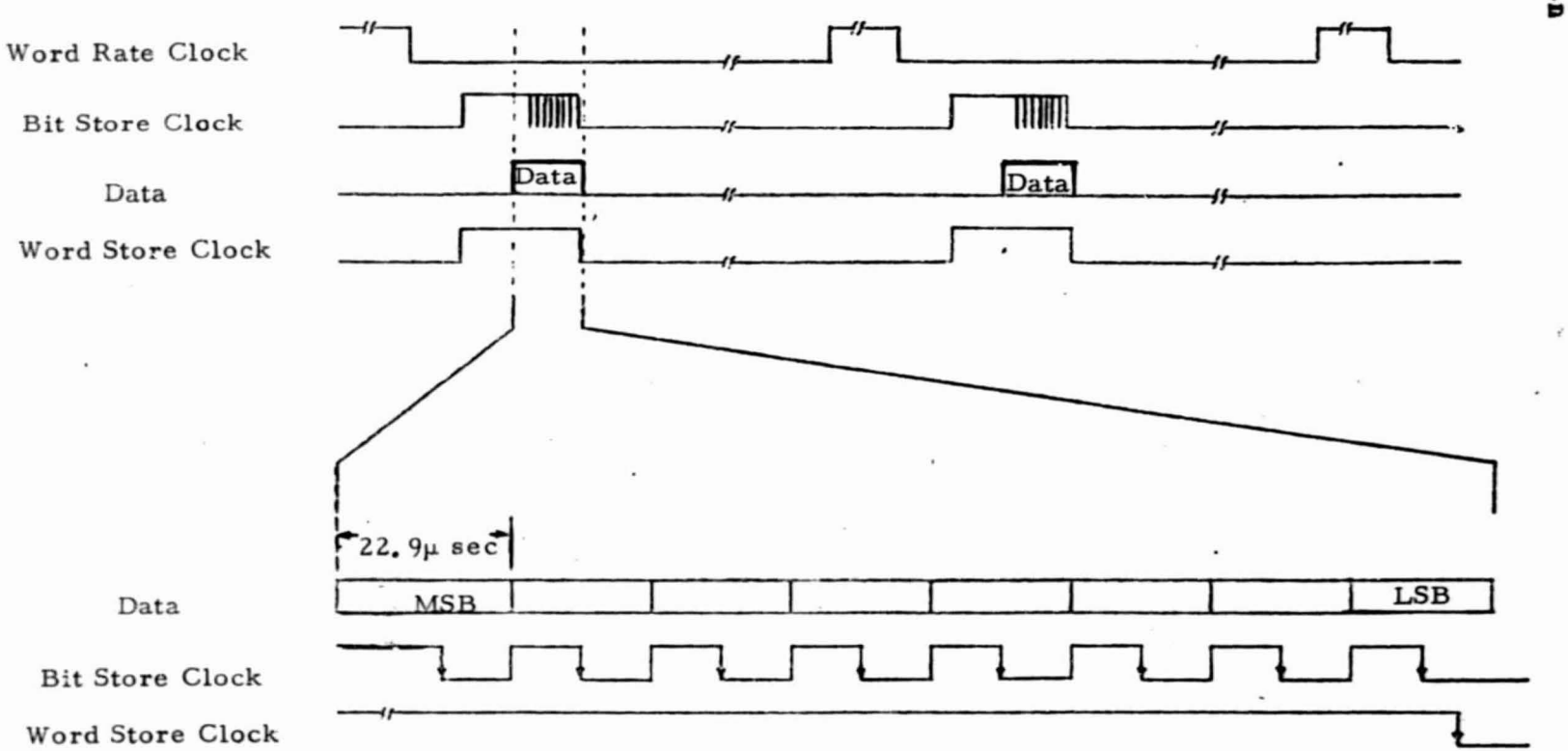


Figure 3.5.2.3-4. DSU Write Timing

NOTE: PW = PROGRAMMABLE WORD

PROGRAMMABLE FORMAT

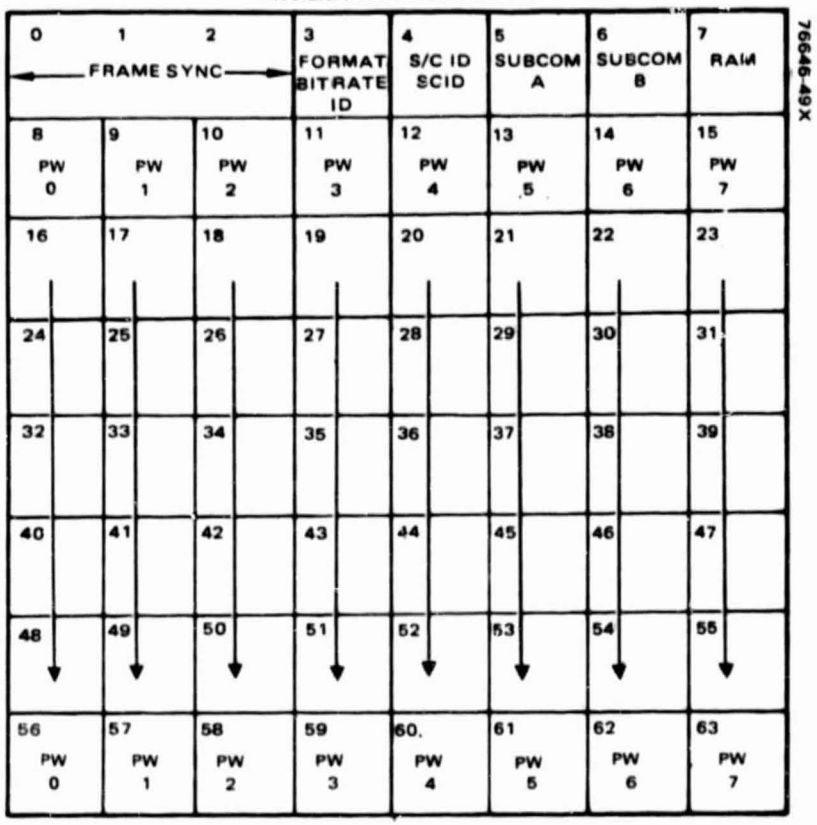


FIGURE 3.5.3.1.1-1. PROGRAMMABLE FORMAT FOR ONE MINOR FRAME

PROGRAMMABLE FORMAT

0 1 2 ← FRAME SYNC →			3 FORMAT BITRATE ID	4 S/C ID SCID	5 SUBCOM A	6 SUBCOM B	7 RAM
8 PLIMTI	9 ATTM1Z	10 PLIMTI	ADP STATUS	12 PLIMTI	13 A*1BRM	14 PLIMTI	15 ADP STATUS
16	17	18	19	20	21	22	23
24	25	26	27	28	29	30	31
32	33	34	35	36	37	38	39
40	41	42	43	44	45	46	47
48	49	50	51	52	53	54	55
56	57	58	59	60	61	62	63

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FIGURE 3.5.3.1.1-2. EXAMPLE OF PROGRAMMABLE FORMAT FOR ONE MINOR FRAME

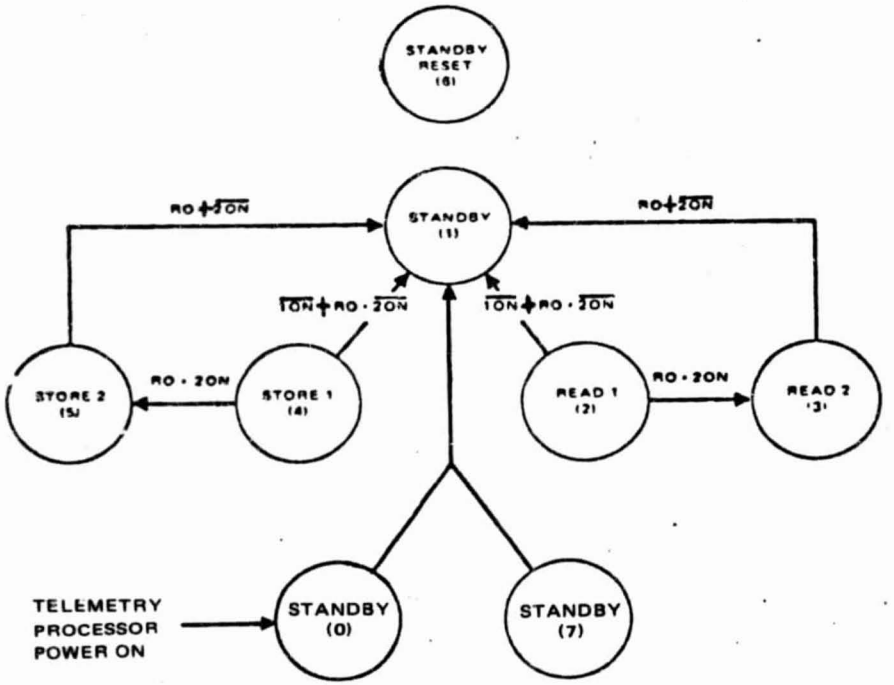
MINOR FRAME \ BIT	MSB 0	1	2	3	4	5	6	LSB 7	COMMENTS
0	ID	X	2 ON	1 ON	0	DSU STATE			ID = 0 FOR BUS AND 1 FOR ORBITER. 2 ON AND 1 ON = 1 WHEN SPECIFIED DSU IS ON. 3 BITS SHOW DSU IN 1 OF 8 POSSIBLE STATES.
1	TEMPORARY STORAGE USED TO PROCESS USER RATE DATA								MINOR FRAME, MAJOR FRAME AND 4096 SEC RATES ARE PROCESSED AND TRANSFERRED TO THE INTERNAL USER RATE REGISTER (INVALID DATA).
2	FRAME FORMAT			BIT RATE					MOST SIGNIFICANT 8 BITS OF QUANTITATIVE COMMAND
3	DSU COMMAND			DMA	X	S	C	D	LEAST SIGNIFICANT 8 BITS OF QUANTITATIVE COMMAND
4	0	0	WORD COUNTER						6-BIT COUNTER
5	0	0	MINOR FRAME COUNTER						6-BIT COUNTER
6	DSU STATE			0	0	0	0	0	3 BITS TO SHOW DSU IN 1 OF 8 POSSIBLE STATES
7	DYNAMIC TIMECODE WORD 0								LEAST SIGNIFICANT WORD OF 4 DYNAMIC TIMECODE WORDS (8 BITS)
8	DYNAMIC TIMECODE WORD 1								THIRD WORD OF 4 DYNAMIC TIMECODE WORDS (8 BITS)
9	DYNAMIC TIMECODE WORD 2								SECOND WORD OF 4 DYNAMIC TIMECODE WORDS (8 BITS)
10	STATIC TIMECODE WORD 1								LEAST SIGNIFICANT WORD OF 3 STATIC TIMECODE WORDS
11	STATIC TIMECODE WORD 2								SECOND WORD OF 3 STATIC TIMECODE WORDS
12	STATIC TIMECODE WORD 3								MOST SIGNIFICANT WORD OF 3 STATIC TIMECODE WORDS
13	FORMAT INFORMATION								LEAST SIGNIFICANT 8 BITS OF 12-BIT FORMAT WORD (INVALID DATA).
14	S/C ID		MINOR FRAME COUNT						SPACECRAFT ID = 00 FOR BUS AND 10 FOR ORBITER. 6 BITS FOR MINOR FRAME COUNT
15	0	0	0	0	FORMAT INFORMATION				MOST SIGNIFICANT 4 BITS OF 12-BIT FORMAT WORD
16	DYNAMIC TIMECODE WORD 3								MOST SIGNIFICANT WORD OF 4 DYNAMIC TIMECODE WORDS (8 BITS)
17	SPARE								WILL INITIALIZE WITH ALL BITS = 0
18	SPARE								WILL INITIALIZE WITH ALL BITS = 0
19	SPARE								WILL INITIALIZE WITH ALL BITS = 0
20	SPARE								WILL INITIALIZE WITH ALL BITS = 0
21	SPARE								WILL INITIALIZE WITH ALL BITS = 0
22	SPARE								WILL INITIALIZE WITH ALL BITS = 0
23	PW3 TYPE	PW4 TYPE	PW5 TYPE	PW6 TYPE	PW7 TYPE	PW0 TYPE	PW1 TYPE	PW2 TYPE	SPECIFIES TYPE OF PROGRAMMABLE WORDS 0-7. ANALOG = 1 AND DIGITAL = 0.
24	DIM NO			CHANNEL NO					PROGRAMMABLE WORD 0
25	DIM NO			CHANNEL NO					PROGRAMMABLE WORD 1
26	DIM NO			CHANNEL NO					PROGRAMMABLE WORD 2
27	DIM NO			CHANNEL NO					PROGRAMMABLE WORD 3
28	DIM NO			CHANNEL NO					PROGRAMMABLE WORD 4
29	DIM NO			CHANNEL NO					PROGRAMMABLE WORD 5
30	DIM NO			CHANNEL NO					PROGRAMMABLE WORD 6
31	DIM NO			CHANNEL NO					PROGRAMMABLE WORD 7

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NOTE X SIGNIFIES UNUSED BIT WHICH MAY BE IN LOGIC 0 OR 1 STATE

IN WORD 3 S = SUBCARRIER ON/$\overline{\text{OFF}}$
C = CONVOLUTIONAL ENCODER BYPASS/$\overline{\text{ENCODE}}$
D = DATA ON/$\overline{\text{OFF}}$

FIGURE 3.5.3.1.3-1. RAM CONTENTS



State Name	State No.	Command
STANDBY	0	000
STANDBY	7	111
Standby	1	001
Standby-Reset	6	110
Read 1	2	010
Read 2	3	011
Store 1	4	100
Store 2	5	101

NOTE: 1. ALL STATE CHANGES REQUIRE A COMMAND EXCEPT FOR THE AUTOMATIC STATE CHANGES SHOWN.
2. IT IS POSSIBLE TO GO FROM ANY STATE TO ANY OTHER STATE BY COMMAND.
3. RO – ADDRESS ROLLOVER
1ON – DSU 1 POWERED ON
2ON – DSU 2 POWERED ON

Figure 3.5.3.6-1. DSU State Diagram (For DSU Control Logic Contained in Each TP)

3.6 COMMAND SUBSYSTEM

3.6.1 Subsystem Description. The command subsystem contains all the command processing capability on the spacecraft. The subsystem provides the capability to decode, process and distribute ground commands to spacecraft subsystems and scientific instruments as well as store commands for later execution. Firing pulses are also provided for pyrotechnic devices. Commands may be either discrete or quantitative. A functional block diagram of the subsystem is presented in Figure 3.6.1-1 (Appendix C).

The subsystem consists of two (2) redundant Command Processors (CPs), six (6) Command Output Modules (COMs) and one (1) Pyrotechnic Control Unit (PCU). On the spacecraft equipment shelf, these units are arranged as shown in Figure 3.6.1-2. The two (2) CPs are connected to the essential bus and are not capable of being commanded off. All six (6) COMs receive secondary power from each CP. The associated COM is power strobed on prior to the issuance of a command output. After the command is issued, the COM turns itself off. The PCU receives control power from the essential bus and squib driver power from a lower voltage tap from the spacecraft batteries to minimize essential bus voltage droops during pyrotechnic device firings. Figure 3.6.1-3 shows the major cross-strapping between the elements of the subsystem.

Real-time ground-generated commands are transmitted via the RF uplink to the spacecraft receivers. Each receiver is connected to a single CP, i.e., there is no cross-strapping. A specific receiver/CP combination is selected by transmitting the RF carrier frequency associated with the receiver in the desired pair. On either the Multiprobe spacecraft or Orbiter spacecraft, there is a unique RF carrier frequency assigned to each receiver. The uplink format is PCM/PSK/PM and the command subcarrier output to the CP is PCM/PSK. The tone frequency for a logic "1" is 250 Hz and for a logic "0" is 100

Hz, and the command bit rate is 4 bits per second.

The uplink command format is 48 bits long and is detailed in the lower portion of Figure 3.6.1-1. This format provides the capability to address either the Multiprobe spacecraft or Orbiter spacecraft, to address either CP, to process real-time commands or store the commands in memory for later execution, to store time delays in memory and to distribute two types of commands to spacecraft users, namely discrete and quantitative commands. A 48-bit preamble of alternating logic "1" and logic "0" tones (250 Hz and 100 Hz respectively) is required prior to sending a single command or multiple contiguous commands to the spacecraft.

The two CPs are cross-strapped such that the output of the FSK demodulator portion of each CP can be processed by either CP. By using the address bit in the uplink command format, either CP can be selected to process the transmitted uplink command. The FSK demodulator verifies the 13 bit sync code prior to sending the transmitted command to the central command processing portion of the real-time processor of either CP. In the central portion, the spacecraft address, the CP address and the poly code are verified before the command is routed to the addressed COM or to the command memory. The polycode is used to detect errors in the 28 information bits (i.e., bits 13 through 40) of the command format.

The subsystem contains two command memories (one in each CP), each of which can store 128 entries. Each entry, which can be a discrete or quantitative command or a time delay, is 24 bits long and is contained in bit locations 17 through 40 of the command format. Once the command sequence is loaded into memory, the sequence can be executed by sending a real-time start command, closure of the spacecraft separation switch or by use of the 4096 second clock from the Telemetry Processor. The 4096 second clock is used as an accurate time reference to start the memory at a later time. Time delays from 125 millisecond to

291 hours can be loaded into each slot of the command memory. The command resolution and time delay resolution for the memory is 125 milliseconds. Commands issued from the memory are executed within 125 microseconds, not including delays such as mechanical relay response time. Timing in the CP permits nearly concurrent processing of commands from the memory and RF uplink. RF uplink commands to one CP and memory commands from the other CP must not address the same COM within 62.5 milliseconds to avoid loss of both commands.

The CP routes all commands to the individual COMs for distribution to the users. Each COM can supply 64 discrete commands and 4 quantitative commands. Different subsystem and scientific instrument users are assigned to the six (6) COMs. Almost all commands are redundant, so the primary command is sent to one COM and the redundant (or back-up) command to a different COM.

The subsystem also provides the firing pulses for all pyrotechnic devices on the spacecraft through the PCU. The PCU receives the standard discrete command output from the COMs and current amplifies these pulses to the level required to actuate the pyro device. The PCU contains six (6) squib drivers each of which can provide a minimum of 5 amps firing current to three (3) squibs. The PCU is divided into two equal redundant halves so that pyrotechnic devices with redundant squibs can be fired simultaneously.

3.6.2 Units Descriptions

3.6.2.1 Command Processor. The CP is the main element in the command subsystem. The CP receives a PCM/PSK subcarrier from the transponder in the communications subsystem and relays control signals to the COMs to provide both discrete and quantitative commands to subsystem and scientific instrument users. A shift register memory is provided in the CP which holds a maximum of 128 entries so that commands and time delays can be stored and then executed at a later time.

When essential bus power is applied to the CP initially, the unit will turn-on without generating any spurious commands. It takes from 0 to 60 seconds for the CP to become operational after the essential bus power is applied, due to the cyclic fuse in the CP (Refer to Section 3.6.2.1.7).

When power is applied to the CP, the SCL remains OFF until commanded ON; Receiver reverse logic configuration will be random.

The major functional components of the CP, as shown in Figure 3.6.2.1-1, are the FSK demodulator, stored command logic, receiver reverse logic, real time processor, telemetry logic, output control logic and power supply. In the following sections, these components are described.

3.6.2.1.1 FSK Demodulator. The FSK demodulator provides the interface with the communication subsystem for uplink real-time command processing. Each CP contains a single FSK demodulator that is connected by a three wire interface to a single transponder; there is no cross-strapping between transponders and demodulators. The FSK demodulator is selected by frequency-selection of the spacecraft transponder connected to it. Either of the two redundant CPs can be selected for processing the output of either of the redundant FSK demodulators by use of the address bit in the uplink command format; i.e., there is cross-strapping between the FSK demodulators and the central command processing portions of the real-time processor of each CP. The central portion of the CP provides power from the essential bus and a 1024 Hz timing signal to the FSK demodulator.

The input FSK command subcarrier is converted to a return-to-zero (RZ) data output, a bit timing clock and a sync code verification pulse. These three signals are used internally in the CP and are also cross-strapped to the redundant CP. The FSK command subcarrier consists of tone frequencies of 100 Hz and 250 Hz, representing

logic zero and logic one data respectively. The bit rate of the FSK input is 4 bits per second. The demodulator also receives a transponder in-lock signal from the transponder that indicates when the transponder is phase-locked to an uplink carrier. If this signal is a logic zero (i.e., no carrier), the demodulator is inhibited from detecting the uplink sync pattern, and consequently, the sync code verification pulse output to the CPs will be a logic zero. The sync code verification output provides a pulse to both CPs after a valid 13-bit sync code has been detected.

As shown in the functional block diagram in Figure 3.6.1-1, the FSK subcarrier signal is first applied to an input amplifier stage. The amplifier drives 3 bandpass filters, one tuned to the logic zero subcarrier tone, a second tuned to the logic one subcarrier tone, and the third tuned to the 1400 Hz noise reference. The output of each filter is rectified through a precision half-wave rectifier that gives both a negative and positive output. The positive output associated with the "1" filter is summed with the negative output associated with the "0" filter in a conventional amplifier summing stage. From here, the output of the amplifier is fed to a low pass filter that drives a hard limiter which converts the positive and negative voltages into TTL levels. The resulting limiter output is sampled and temporarily held in a flip-flop before being clocked out to the real-time processor by the bit synchronizer.

The negative outputs of both the "1" and "0" filter/rectifiers are sent to the FSK demodulator squelch circuit along with the output of the noise filter. The squelch is a comparison circuit that compares the energy in the noise filter bandwidth to the total energy in the two tone filer bandwidths in order to determine whether a subcarrier is present or not. The noise filter is an active two-pole filter with a center frequency of 1400 Hz. Since the noise input to the tone filters is a function of the input signal strength to the transponder, the

output of the noise filter is used as the reference to determine if a signal is present or not. This mechanization is called a variable threshold squelch. The output of the noise filter is sent to rectifier and integrator circuits before being compared with the integrated output of the tone filters. The comparator circuit provides an enable signal to the bit synchronizer when a signal is present and inhibits the bit synchronizer when there is no subcarrier input. The status of the squelch state is provided in telemetry by CDHD1S for CP1, and CDHD2S for CP2 (A logic "1" indicates the demod is unsquelched, and a logic "0" indicates that the demod is squelched). The performance of the squelch circuit is a function of uplink received signal strength. For an input level of -110 dBm or greater, the PSK demodulator will remain "unsquelched" indefinitely in the absence of a subcarrier. However, for lower signal strengths the PSK demodulator will "squelch" when the subcarrier is removed (i.e., the uplink command is completed) as a function of signal level. The characteristic of the squelch circuit is given in Figure 3.6.2.1.1-1.

The clock that is used to output the data from the PSK demodulator is generated by the bit synchronizer. The bit synchronizer receives a 1024 Hz square wave from the output control logic and divides it to a 4-bit per second variable phase clock. The phase of the clock is set during the acquisition time. At the beginning of a command sequence, a preamble of 48 bits of alternating logic ones and zeros must be transmitted. After squelch is released, the first zero-to-one data transition from the hard limiter resets the bit synchronizer to a phase which will cause the data flip-flop to be clocked at the optimum sample time. The phase of the clock is automatically adjusted in either direction to an accuracy of one-sixteenth of a bit time as required by the incoming data. After initial resetting of the bit synchronizer, the positions of the transitions in incoming data with respect to the phase of the clock are constantly monitored. As long as the transitions

fall within a given time "window", the timing will remain unchanged. If a data transition occurs before or after the window, the clock phase will be shifted by one-sixteenth of a bit time in the appropriate direction to maintain optimum bit timing. This process will continue for the duration of a command sequence.

3.6.2.1.2 Real Time Processor. The PSK demodulator transmits RZ data, a 4 Hz clock and the sync code verification pulse to the Real Time Processor (RTP). These signals are received by the RTPs of both CPs, but only the CP that detects the appropriate address bits processes the data from the active PSK demodulator.

Before the RTP will begin processing the incoming uplink command, (either a real time command or a command intended to be stored for later execution) the sync code verification pulse must be received from an active PSK demodulator. After the sync code verification pulse is detected, the incoming command is simultaneously loaded into a shift register, fed to the address check logic and fed to the polynominal code check logic. From the uplink command format given in Figure 3.6.1-1, it can be observed that bits 13, 14 and 15 contain the spacecraft and CP address information. The RTP performs two address checks on these three bits. If the first address check detects an error, the input control logic is reset until the next sync code verification pulse is received. At this check, there is no telemetry indication that the command was not accepted. The second address check is performed when the last bit (i.e., bit 47) of the command is loaded into the shift register. At this time, the polynomial code check is also completed. Once the shift register is loaded, the polynomial code check and two address check states are examined. If no errors are found, and the command is to be routed to a COM, a processing pulse is sent to the output control logic. If the command is to be routed for storage, the command is routed directly to the SCL. If either the polynomial code or second address check is false, a reject bit (logic "1") is sent to

telemetry (i.e., CREJXS, where X = 1 for CP1 and X = 2 for CP2). The reject bit does not prevent the processing of subsequent commands, and it can be reset by the CP CONFIGURE quantitative command (CPCQ1 for CP1, or CPCQ2 for CP2). When all checks are satisfactory, a counter in the telemetry logic is incremented by one and the current count is provided as telemetry output CCMDXC, where X was defined previously.

The timing in the CP is designed to accommodate real time and stored command executions nearly simultaneously. The main clock generates 8 Hz (i.e., 62.5 msec half period) time slots that are assigned alternately to real time and stored command processing. Bit 16 in the uplink command format is used to determine whether the output is to go to the output control logic or stored command logic. If the data are transferred to the output control logic section, the RTP adds a "redundant code check" bit to the outgoing word to indicate whether the information is valid or invalid, i.e., to indicate that the data bits have not picked up an error in transmission. The output control logic responds at the proper time to the processing pulse from the RTP by sending back to the RTP a high speed output clock pulse (generated in the output control logic for the purpose of transferring in data) to start the output process at the start of the next real-time time slot.

3.6.2.1.3 Stored Command Logic. The Stored Command Logic (SCL) consists of the command memory, memory control logic and configuration control logic. The SCL provides the capability for spacecraft control when real-time commands through the RF uplink are not available. Any command that can be executed from the ground in real-time can be stored in the command memory and executed under control of the SCL. All stored command mnemonics carry, as a sixth alphanumeric, the letter "A" for CP1, or the letter "B" for CP2. Sequences of commands with predetermined time delays can be executed by programming the command memory with both command data and time delay information.

The command memory is a 3072 bit shift register organized as 128 words (each word a command or a time delay) of 24 bits each (i.e., bits 17 through 40 of the uplink command format). Bit 16 of the uplink command format is used to select the command memory and bit 17 specifies whether a command or time delay (CMTQ1 for CP1; CMTQ2 for CP2 for a time delay command) is being transmitted to the command memory. The time code contained in the time delay word is 23 bit longs and is decremented at an 8 Hz rate (i.e., in 125 millisecond increments) until zero is reached, at which time the next 24 bits are read from the memory. Each time delay word can be coded to provide time delays from 125 milliseconds to 291 hours with a resolution of 125 milliseconds. Time delay words can be programmed consecutively in memory. Since the alternating real-time and stored command processing slots are 62.5 milliseconds in length, the maximum rate at which words (commands or time delays) can be executed from the command memory is one per 125 milliseconds.

The SCL provides several other functions in addition to the execution of stored commands and time delays. These are load, verification and command memory maintenance functions. Section 3.6.3.2 describes the operation of the SCL and the SCL state diagram.

The SCL in each CP is designed so that a failure of one SCL will not interfere with the operation of the other SCL. An SCL OFF discrete command (MEM1Ø for SCL 1 and MEM2Ø for SCL 2) is provided to turn off the SCL in case of a failure of the configuration control logic.

3.6.2.1.4 Output Control Logic. The Output Control Logic receives inputs from the real-time processor and SCL. To start the output process, the SCL or real-time processor transmits an enable flag to the output control logic to indicate that it desires to be serviced. At the beginning of the proper time slot (i.e., real-time for the real-time processor or stored for SCL), a signal is sent from the input select logic in the output

control to the output shift register of the real-time processor or SCL. The Output Shift Register (OSR) in the output control logic sends a clock to the real-time processor or SCL to allow the information to be loaded in the output shift register. The output control logic generates the real-time and stored command processing time slots for the CP.

When the COM address portion of the word is loaded into the OSR, it is sent to the output decoder to determine which COM is to receive the information. Once the COM is selected, a 2 millisecond introduction pulse is sent to the COM. During the introduction, the input word is loaded into the OSR. The check bit at the end of the input word is then sampled to determine if the command data bits are valid. If the check bit is false, the output process is termined.

The control bit contained in the uplink format that specifies whether the command is discrete or quantitative is then interrogated. When this is complete, the logic is set within the OSR to shift out reformatted data to the Manchester encoder. These data then go to the output decoder stage where it is routed to the selected COM output buffer.

3.6.2.1.5 Receiver Reverse Logic. The Receiver Reverse Logic (RRL) is used to automatically switch the antenna connections to the spacecraft receivers and select the omni antenna nominally every 36.4 hours ±14 sec. This is necessary to maintain control of the spacecraft in the case of a receiver failure when communication to the spacecraft is through an antenna connected to the failed receiver. The status of the time code stored in the RRL timer is given in 9.1 hour (±4 sec.) increments by two bits, bits 6 and 7, of CP status word 1, which is telemetry output RREVXC, where X was previously defined.

The RRL accepts an internal 4 Hz clock and counts 2^{19} periods (i.e., 36.4 hours) before executing the receiver reverse functions. The timer is reset when power is applied to the CP and

whenever a valid command is received by either CP. Under normal conditions, there is no way to defeat the operation of the timer. When the timer runs out, two pulses will always be generated, one to select the omni and one to reverse the position of the RF switch that controls the antenna connections to the receivers.

The pulse generating circuits are activated by a transition on the timer output or from the CP CONFIGURE quantitative command used to control the RRL. When a normal or reverse pulse is received from the configuration control as a result of the quantitative command, the selected state is set and a normal or reverse pulse is generated. In this case no select omni pulse is generated. When the timer goes to zero, a select omni pulse is generated; and, if the RRL is in the normal state, a reverse pulse is generated and the RRL enters the reverse state, and vice versa. The state of the RRL is provided to telemetry as bilevel signal CREVXS, where X was defined earlier.

3.6.2.1.6 Telemetry Logic. The telemetry logic provides the CP status and command memory data to the Data Input Modules (DIMs) for insertion in the telemetry data stream. The status consists of four serial digital data words shifted out on four different lines in response to DIM read envelopes and a clock and two bilevel telemetry outputs. The serial digital data words are updated continuously until they are read into the DIMs. The contents of the command memory can be verified by one of these serial digital data outputs by using the command memory readout format and placing the SCL in the read state (refer to Section 3.6.3.2). Table 3.6.2.1.6-1 shows the word formats for the four serial digital words. See Appendix A for further details.

3.6.2.1.7 Power Supply. The power supply for the CP is a non-commandable supply. The functions performed by the power supply are generation of secondary voltages for the CP and COMs, current limiting,

TABLE 3.6.2.1.6-1

COMMAND PROCESSOR TELEMETRY OUTPUTS

Telemetry Word	Format	Active State: Logic 1 Inactive State: Logic 0	Mnemonic (See Note)
Word 1	Bit 0 ⟶ Bit 3 (MSB) (LSB)	SCL state status; see SCL state diagram, Figure No. 3.6.3.2-1.	CLOGXS
	Bit 4	SCL ON/OFF Status.	CSCLXS
	Bit 5	Demodulator status that indicates squelch state.	CDMDXS
	Bit 6 ⟶ Bit 7 (MSB) (LSB)	MSB bits of receiver reverse timer, 18.2 hour and 9.1 hour, respectively. (The two bits produce four time combinations)	RREVXC
Word 2	Bit 0	Reject flat indicating that real time processor has detected an error in an RT command.	CREJXS
	Bit 1 ⟶ Bit 7 (MSB) (LSB)	7 bit count indicating number of verified real time commands processed.	CCMDXC
Word 3	Bit 0	Separation switch status.	SSEPXS
	Bit 1 ⟶ Bit 7 (MSB) (LSB)	SCL address pointer indicating present location in SCL memory.	CMEMXC
Word 4	Bit 0 ⟶ Bit 7 (MSB) (LSB)	Represents 8 bits of 24 bit SCL command word stored in memory. During SCL memory dump, 3 telemetry words are needed to completely read the 24 bit SCL command word.	CMEMX

Note: X = 1 for CP 1
X = 2 for CP 2

undervoltage and overvoltage detection and shutdown, power strobing and cyclic power control for overcurrent conditions. The undervoltage and overvoltage set points are beyond the UV/OL limits for the main bus lines; the latter will trip before the former will.

When an overcurrent condition (i.e., current > 660 milliamperes ±20%) is detected, The cyclic power control circuit turns the power supply off within approximately 1.5 seconds. The power supply remains off for approximately 60 seconds and then it is turned back on. If the overcurrent conditions remain, the power supply is again turned off in approximately 1.5 seconds. This cyclic action remains as long as the fault remains; once the fault is cleared the power supply remains on.

3.6.2.1.8 Internal Command Response. The CP receives both quantitative and discrete commands from the COMs to control the configuration and operation of the unit. The following sections describe the internal response of the CP to the different commands received.

3.6.2.1.8.1 CP CONFIGURE Quantitative Command (CPC01 for CP1; CPC02 for CP2). The 16-bit quantitative command (i.e., bits 25 through 40 of the uplink command format) used to configure the CP is shifted into a register in the configuration control logic in the SCL. Bits 33 through 40 of the quantitative command word are decoded and processed by the configuration control logic; the other 8 bits are not used. The verification pulse at the end of the quantitative command is used to initiate this processing. Two of the bits (bits 39 and 40) are used to select the CP. It is desirable to code bits 39 and 40 so as to select "either" command processor (code 11). By selecting the "either" code it will not be necessary to match the CP select code (code 01 for CP1 and code 10 for CP2) in bits 39 and 40 to the CP selected by bit 15. If the CP is not selected (code 00), no further processing is done. However, if the CP is selected, the other

bits are decoded and used to control their specified functions.

Two bits of the quantitative command (bits 33 and 34) are decoded to provide a pulse to clear the command counter and reject status flip-flop in the telemetry logic (i.e., code 11) or, to control the receiver reverse functions in the receiver reverse logic. Either the normal omni antenna/receiver configuration (code 01) or the reverse omni antenna/receiver configuration (code 10) can be selected. When a normal or reverse pulse is received from the configuration control logic, the receiver reverse logic will generate a normal or reverse command, depending upon which is commanded. These commands are routed to switch drivers in the communications subsystem to select the desired omni antenna/receiver configuration.

The other four bits of the quantitative command (bits 35 through 38) are decoded by the configuration control logic to produce SCL control commands. Three of these commands (SCL CLEAR/ON (STORCMD = CLER1 or CLER2), INDEX (STORCMD = INDX1 or INDX2) and SCL OFF (STORCMD = OFF1 or OFF2)) are generated by the configuration control logic. The other commands are used to configure the SCL to various operation states (see Figure 3.6.3.2-1, SCL State Diagram). These commands are only partially processed by the configuration control logic. Signals are sent from the configuration control logic to the SCL control logic where processing of the SCL state command is completed.

As shown in the SCL State Diagram, state changes in the SCL are not only dependent on incoming commands, but are also affected by the present SCL state and other signals such as the separation switch and the 4096 second clock. Allowable state changes are stored in a Read Only Memory (ROM) in the SCL control logic. Each ROM location contains four bits representing an SCL state. The ROM address is determined by the command from the configuration control logic, the present SCL state, and other control signals.

The state can only change at the beginning of an SCL time slot. After the time slot ends the new state is selected by reading the content of the ROM at the specified address. This may or may not be the state specified in the command, depending on the allowable state changes in SCL operation.

The SCL CLEAR/ON, INDEX, and SCL OFF commands are decoded by the configuration control logic and processed during the verification pulse time of the quantitative command. The CLEAR/ON command sets a latch which controls the SCL power and triggers the initial condition circuit. These circuits are also in the configuration control logic. The CLEAR/ON command turns on power to the SCL and clears the memory. If power is already on, the memory is just cleared. The OFF command resets the latch which controls the SCL power and thus turns off the SCL power. The INDEX command is routed to the memory control logic and advances the memory until the first memory slot is at the shift register output.

3.6.2.1.8.2 SCL OFF Discrete Command. The SCL OFF command (i.e., MEMX/) is a discrete command provided as a backup to the SCL OFF command provided in the quantitative command for configuring the SCL. This command is routed through an input buffer in the configuration control logic to the latch which controls the SCL power. This command will reset the latch and thus command SCL power OFF.

3.6.2.2 Command Output Module. The Command Output Module (COM) provides a standardized command interface to all subsystem and scientific instrument users. There are six (6) COMs on the Orbiter spacecraft. Power, grounding and control signals are provided to each COM from both CPs. Each COM distributes four (4) quantitative commands and 64 discrete commands to spacecraft users. Each quantitative command consists of a clock, envelope and data line. A backup discrete command input is provided to each COM to command it off in the event that a failure occurs that causes the power to remain on after the completion of processing a

command; the unit is normally power strobed through the control input.

The COM has only two operating modes: standby and processing. The power dissipated by the COM is 1.6 milliwatts (maximum) during standby and 1.1 watts (maximum) during processing of a command. A true off state without power consumption does not exist in the COM. In addition, there is 1.6 milliwatts during standby and 1.0 watts during processing that is dissipated within the CP due to the inefficiency of the power supply. The COM OFF command is implemented to insure that an internal logic failure would not maintain a continuous 2.1 watt (maximum) load on the battery. The addition of this command permits the COM to be returned to the standby mode. In the event the COM has to be commanded off, that OFF command is sent through a different COM. (A COM that is hung up in the "ON" state may process commands normally, depending on the nature of the failure, but will be an undesirable steady state load. Certain failures could preclude the further use of the COM). There is no direct telemetry indication that a COM has failed in the processing mode. This type of failure can be investigated by sending a few operationally safe commands to the suspect COM and monitoring the associated telemetry for the expected responses. A failure to respond to all the commands sent is a positive indication that at least a segment of the COM is no longer usable.

The input control word to the COM consists of an introduction, data and a turn-off sequence. The introduction consists of a logic "1" level for approximately two milliseconds. This permits the power strobe circuit to turn power on for initial conditions to be established and for the oscillator to reach steady state. After the introduction ends, the data input begins and is sent to the Manchester decoder. The output of the Manchester decoder is shifted into a shift register in NRZ form.

The control logic then determines from the control bits in the shift register whether the

command is discrete or quantitative. If the command is discrete, the output line address is decoded and the output is transmitted when the hyperpulse is received. For a quantitative command, the sixteen data bits are transferred to the correct quantitative command output. The control logic generates the envelope and clock signals from the Manchester decoder and gates them to the output. The COM shuts down after the last bit of the quantitative command or removal of hyperpulse for a discrete command. During processing, checks are made in the control logic and Manchester decoder circuits. If a data transition does not occur at the proper time, the parity check fails, or the hyperpulse arrives early, the output will be inhibited.

At the end of a quantitative command, a verification pulse is sent as part of the output. This pulse indicates that the correct quantitative command length was transmitted from the COM to the user. For a quantitative command, if the length of the command is different than 16 bits, the incorrect number of bits will be sent to the user without the verification pulse.

3.6.2.2.1 Internal Command Response. Each COM receives one discrete command, called the COM OFF command, via a different COM, to turn the COM off in case it does not reset from the process command mode (i.e., continuously ON state). The command is given by COMXØ, where X = 1, 3 and 5. When X = 1, the command turns off COMs 1 and 2, when X = 3, the command turn off COMs 3 and 4, and when X = 5, the command turns off COMs 5 and 6.

The COM OFF command is routed to an input buffer configured to operate in the memory mode. In normal operation, the input signal from the CP will turn unit power ON and latch the input buffer ON. At the completion of command processing the latch is turned off by a signal from the control logic. The COM OFF command is provided as a backup to turn the latch (and thus power) off in case of a failure in the control logic.

3.6.2.3 Pyrotechnic Control Unit. The Pyrotechnic Control Unit (PCU) is used to provide firing pulses for spacecraft pyrotechnic devices. The PCU receives a standard 35 ±4 millisecond pulse from the COMs and current amplifies the pulse to a minimum of five amperes into a 1 ±5% ohm load. The output current pulse is of the same duration as the input pulse. The Orbiter spacecraft contains one PCU.

The PCU consists of two fully redundant halves. In each half there are three squib drivers each with three outputs. A squib driver is activated by first sending an arm command and then following it with a fire command. One arm circuit enables one squib driver and a second arm circuit enables the remaining two squib drivers. The PCU contains an automatic disarm which disarms the PCU 18.5 ±4.5 seconds after it is armed. An arm delay circuit is also included that delays the arming of the PCU for 4 ±2 seconds after receipt of the arm command. These timing relationships are shown in Figure 3.6.2.3-1.

The fully redundant structure of the PCU enables two independent signals to fire one redundant squib. A single command from a COM is cross-strapped to a squib driver in each half to fire both halves of redundant squibs simultaneously. The firing simultaneity for this configuration is less than 250 microseconds.

3.6.2.3.1 Internal Command Response. The PCU receives three different types of disctete commands, i.e., ARM, FIRE and DISARM. The internal response to these commands is described in the following sections.

3.6.2.3.1.1 PCU ARM Discrete Command. The PCU ARM (i.e., ORD11 or ORDA1; ORD13 or ORDA3) command is used to provide power to the squib drivers so that a subsequent FIRE command will fire the associated squibs. From the receipt of the ARM command, a timer in the PCU delays arming of the squib drivers by 4.0 ±2.0 seconds and disarms the drivers after 16.5 ±4.5 seconds. The FIRE

command must be received between these events if the squibs are to be fired. (100% assurance of firing is obtained if the FIRE command is received between 6 and 14 seconds after the ARM command is received). The ARM command is routed to both redundant sections of the PCU.

The ARM command is amplified to the level required in the PCU by a discrete input buffer. The output of this buffer will set a switch which supplies 28 volts to the associated arm/disarm switch and resets the switch which supplies 28 volts to the other arm/disarm switch. The output of the discrete buffer also starts the timer in the PCU. After a delay of nominally 4.0 seconds a signal from the timer will activate (arm) the selected arm/disarm switch which will apply battery voltage to the associated squib drivers. After a total delay of nominally 18.5 seconds a signal from the timer will disarm the arm/disarm switch and thus remove battery voltage from the squib drivers.

The two PCU arm commands are interlocked. If one "arm" circuit is energized (i.e., relay closed), sending a command to the second "arm" relay will arm it, but will "disarm" the other arm/disarm relay. In other words, both "arm" relays cannot be closed at the same time (Reference: 1.5.17).

3.6.2.3.1.2 PCU FIRE Discrete Commands. The PCU FIRE (i.e., ORD12 or ORDA2; ORD14 or ORDA4; ORD15 or ORDA5); command is routed to the squib drivers through an input buffer. Power will only be supplied to the input buffer and squib driver when the associated arm/disarm switch is armed. The squib driver amplifies the squib fire command and is capable of supplying current to fire three squibs. If the command is to fire redundant squibs, it is connected to two input buffers by wiring external to the PCU.

3.6.2.3.1.3 PCU DISARM Discrete Command. The PCU DISARM (i.e., ORD10 or ORDA0) command is provided as a backup for the DISARM command generated by the 18.5 second timer in the PCU. This command is routed through an input buffer to the arm/disarm

switches. If the timer fails to disarm the arm/disarm switch, this command will disarm the switch and thus remove battery voltage from the squib drivers. The DISARM command is routed to both redundant sections of the PCU. The DISARM command cannot truncate the 18.5 second timer count (i.e., disarm prior to completion of the nominal 18.5 second delay) when the timer is functioning properly.

3.6.3 Operational Description. The following sections describe the operational usage of the command subsystem, including real-time command, stored command and pyrotechnic device operations.

3.6.3.1 Real-Time Command Operations. All real-time command operations with the spacecraft are through the RF uplink. A PCM/PSK/PM uplink carrier is transmitted to the spacecraft receivers. The spacecraft contains two transponders; each contains a frequency addressable receiver. The receivers demodulate the uplink carrier and provide an PCM/PSK subcarrier signal to the PSK demodulator in the CP. Each receiver is connected to one PSK demodulator without any cross-strapping. The output of each PSK demodulator is cross-strapped to the central command processing circuitry in each CP. Consequently, either CP can be used to process real-time commands.

Commands are sent to the spacecraft at 4 bps using the 48 bit command format given in Figure 3.6.1-1. Prior to sending the 48 bit command word, a 48 bit preamble must be sent to synchronize the PSK demodulator with the incoming command. The preamble should be an alternating pattern of logic "1" and logic "0" tones in either order, i.e., 1010 . . . or 0101 . . . The logic "1" tone is 250 Hz and the logic "0" tone is 100 Hz. After the initial preamble is sent to the spacecraft, the PSK demodulator will remain synchronized if the tone frequency is changed a minimum of once in every 48 bit times. Therefore, if 48 bits of logic "1" tones are sent followed by 48 bits of logic "0" tones, and this pattern is repeated continuously, the command

word may be sent without preceding it with the preamble. However, when a command is transmitted it is necessary that the first bit of the command be started within 1/16 of a bit time of the idle bit transition time or the PSK demodulator will go out of synchronization. Therefore, the PSK demodulator will remain locked-up for 48 bit times without a transition in the tone frequencies, and once lock is lost the preamble must be sent prior to transmitting another command word.

The 48 bit command format consists of a 13 bit fixed sync pattern, 28 bits of address information and command data and seven bits of coding for the polynomial code check performed by the CP. The 13 bit sync pattern is given in Figure 3.6.1-1. The seven polynomial code bits are generated from the 28 data bits and are a function of the contents of these bits. The AHC command software automatically creates this seven bit code and attaches it to the end of the command word. The CP decodes these seven bits and uses this information to determine if an error in transmission has occurred in the 28 data bits received by the CP. This procedure detects random errors of any of the following types: single bit, double bit, odd number of bits, all bursts of length seven bits or less, 98.4% of bursts of length eight bits and 99.2% of longer than bursts of length eight bit random errors. If an error is detected by the CP, a command reject flag is set in CP telemetry output CREJXS, where X equals 1 for CP 1 and X equals 2 for CP 2.

The spacecraft and CP address information is contained in bits 13, 14 and 15 of the command format. To select the Orbiter spacecraft, a "10" code must be placed in bit locations 13 and 14. The selection of the CP used to process the uplink command word is accomplished by bit 15. A "0" in bit position selects CP 1 and a "1" selects CP 2. In the shelf layout given in Figure 3.6.1-2, CP 1 is the bottom unit. As a further reminder, the PSK demodulator in either CP is connected to a transponder that is

frequency-selected, whereas the remainder of each CP is selected by bit 15 of the command word.

Bit 16 is used to specify whether the command word is to be executed in real-time or stored in the command memory to be executed at a later time. If this bit is a logic "1", the command will be executed immediately and if it is a logic "0", the following 24 bits will be stored in the command memory. Before sending a command to the memory, the SCL must be commanded to state 14 (Loading) or the command will be ignored.

For real-time commands, bit 17 can be either a logic "1" or a logic "0" (i.e., "don't care") because this bit is used to specify whether the following data bits contain a time delay or command information in the stored command format. If commands are sent to the SCL and a logic "0" is selected, the data will be processed as a time delay, but if a logic "1" is selected, the data will be taken as coding for a command. If time delay information is entered into the command word, the following 23 bits (i.e., bits 18 through 40) are used for coding the time delay. The 23 bit code for the time delay is the binary representation for the desired time in increments of 125 milliseconds plus a bias of 125 milliseconds. Therefore, binary 0 is equal to 125 milliseconds, binary 1 equals 250 milliseconds, etcetera. The binary number is loaded LSB first, so that bit 18 in the command format is the LSB and bit 40 is the MSB. The telemetry verification of the memory contents (i.e., CMEMX) will be read out LSB first for the time delays as loaded.

The required time delay between two events can be calculated by using the following general equation:

$$T = 8t - n$$

where:

t = desired time delay between events in seconds

$n = n_2 - n_1$

n_1 = memory location (slot number) of the command entry from which time delay begins

n_2 = memory location (slot number) of the command entry being timed.

Example 1

Command ABC in slot 5.
Command XYZ in slot 7.
Desire 10 seconds delay between commands:

$$T = 8t - n$$
$$= (8 \times 10) - (7-5)$$
$$= (80) - (2)$$
$$T = 78$$

$$78_{10} = 1001110_2$$

Therefore the 23 bits of command data in the command word are

011100 10000000000000000
bit 18 bit 40

When bits 18 to 40 are used for stored command information (bit 17 = "1") in the stored command format, these bits are structured to provide two types of commands - discrete and quantitative. Bit 22 is used to select either a discrete command (i.e., logic "0") or a quantitative command (i.e., logic "1"). For discrete commands, bits 18 through 21 select the COM to which the command is to be routed and bits 35 through 40 designate the COM output allocated to

the user connected to the COM. These two sets of codes are conventional binary representations for the decimal number placed on these locations; i.e., COM 0 to COM 7 is specified as 0000 to 0111 and COM outputs 0 to 63 are coded as 000000 to 111111. COM 0 is a fictitious unit for sending an all zeroes test command to the spacecraft. COM 7 is located only on the Large Probe spacecraft, but for the Orbiter spacecraft (as well as the Multiprobe spacecraft), it is used for sending an all ones test command. Output 63 to COM 7 is not assigned to any user on the Multiprobe spacecraft. Bits 23 to 34 are all zeroes for discrete commands.

When a quantitative command is desired, bits 18 to 21 also select the COM, but bits 23 and 24 are coded to specify the COM quantitative command output, i.e., COM output 0 equals 00 and COM output 3 equals 11. Sixteen bits ar allocated to code the quantitative data in the command. These data bits are in locations 25 through 40. The allocation of these bits in the quantitative commands to each of the spacecraft subsystem users is specified in PC-455 (Reference: Paragraph 1.5.1).

The time to process a command in the CP from the time it enters the FSK demodulator to the time it is sent to a COM is a minimum of -3.5 milliseconds and a maximum of +137 milliseconds. The actual time is random (uniform distribution).

3.6.3.2 Stored Command Operations. The use of the stored command capability on the spacecraft begins with the loading of the command memory through the use of real-time commands via the RF link. The command word format required to place commands (i.e., discrete or quantitative) and time delays in the memory was discussed in the previous section.

The operational use of the command memory is centered around the state diagram for the SCL. The SCL consists of the configuration control logic which processes command inputs to the SCL, the control logic and the command memory. Figure

3.6.3.2-1 presents the complete SCL state diagram. In order to describe the use of stored commands on the spacecraft, a detailed explanation of the SCL is presented in the following sections.

3.6.3.2.1 Initial Application of Bus Power and SCL ON/OFF Control. When the essential bus is connected to the CP for the first time, the SCL is initialized to be in the OFF state. To turn the SCL ON, the CP CONFIGURE quantitative command must be sent to the spacecraft. The coding for the CP CONFIGURE quantitative command is given in PC-455 (Reference: Paragraph 1.5.1). Subsequent ON/OFF control of the SCL is by the quantitative command since the CP cannot be commanded off. A backup discrete command, SCL OFF, is also available in case there is an SCL failure which prevents the SCL from being commanded off by the quantitative command. However, in the case of an overcurrent condition that causes the cyclic fuse to turn off the CP, the SCL will be turned off and remain off after the fault is cleared and the CP power supply operation resumes. For this case, the SCL will have to be commanded back on if SCL operations are required. The ON/OFF status of the CP and the SCL are provided in telemetry outputs CPWRXS and CSCLXS, respectively, where X equals 1 or 2 for CP1 or CP2, respectively.

3.6.3.2.2 Preparation for SCL Operations. The format for the uplink command word for loading commands and time delays into the memory is given in Figure 3.6.1-1 and was discussed in Section 3.6.3.1. The steps that need to be taken to prepare, load and verify the memory contents before executing the sequence are given as follows.

3.6.3.2.2.1 Turning On the SCL. To begin with, the SCL must be commanded on with the CP CONFIGURE quantitative command using one of the two CLEAR/ON command codes in the quantitative command structure as defined in PC-455, Reference: Paragraph 1.5.1. When this command is executed by the CP, the SCL is initialized to be in the Standby State (SCL state 15), all memory slots read zero, and the memory address pointer

will initialize at address 0. The SCL ON/OFF status, state, and memory address are provided to telemetry as CSCLXS, CLOGXS and CMEMXC, respectively, where X equals 1 or 2 for SCL 1 or SCL 2, respectively. The structure of the serial digital telemetry outputs associated with the SCL are given in PC-454, Reference: Paragraph 1.5.2, and in Figure 3.6.1-1. Prior to turning on the SCL, CSCLXS should be "0", CLOGXS should be "1111", CMEMXC should be "1111111" and CMEMX should be "11111111". The telemetry output after commanding on the SCL should be "1" for CSCLXS, "1111" for CLOGXS, "0000000" for CMEMXC and "00000000" for CMEMX. CMEMX is the command memory readout telemetry channel.

3.6.3.2.2.2 Loading. Prior to sending commands to the memory, the SCL must be transferred to the Loading State (SCL state 14). This is accomplished by sending the LOAD command code (0100) in the quantitative command. Telemetry verification of CLOGXS should be "1110". Each SCL can be loaded only from its associated Command Processor (i.e., there is no cross-strapping between SCLs and CPs regarding loading of SCLs).

When loading the command memory, the address pointer, CMEMXC, will display the next memory location to be loaded. When the memory is full, the address pointer will recycle to address 0, and any additional inputs will be written over existing commands in the memory.

After loading the desired command sequence in the memory, the STANDBY command must be sent to take the SCL out of the Loading State; no other command will transfer the SCL out of the Loading State. The SCL does not automatically transfer to the Standby state when the memory is completely loaded.

3.6.3.2.2.3 Verification. The contents of the command memory can be verified prior to executing the loaded sequence by commanding the SCL into the Read State (SCL state 10). In this state CLOGXS is "1010". The Telemetry Processor (TP) must be

then commanded into the Command Memory Readout format for the telemetry system to sample the contents of the memory. The commands to the spacecraft have to be sent in this order so as to prevent sampling the memory on the first sample in less than an eight bit byte. The TP and CP are unsynchronized so that the Read state could be started in the middle of a DIM read envelope and could cause all subsequent eight bit bytes to be skewed off by the number of bits contained in the first sampling of the memory.

In order to read the full contents of the command memory, the INDEX command should be sent to the SCL to place the address pointer at address zero. The INDEX command has the dominant purpose of setting the memory address pointer to zero. If the address pointer is not reset to zero, the telemetry output will begin at the address present when the first telemetry sample is taken and terminate when the address pointer is at zero. If the memory is sampled and not in the Read State, the telemetry output will be all zeroes if the SCL is on. At the completion of reading out the command memory into telemetry, the SCL automatically transfers back to the Standby state. When reading the memory, the memory address pointer, CMEMXC, indicates the next memory location to be sampled. If the TP is commanded out of the Command Memory Readout format prior to the completion of the memory readout, the SCL should be set back to address 0 by sending the INDEX command.

The Command Memory Readout format is designed to read out memory entries from SCL 1 and memory entries from SCL 2 in each minor frame. A telemetry word is eight bits long so it takes three telemetry words to read out one memory entry into the minor frame format. The minor frame format is divided in half with words 9 through 35 devoted to SCL 1 and words 37 through 63 dedicated to SCL 2. Therefore, verification of the memory contents is accomplished by making a bit by bit comparison of the 24 bit words sent to the SCL with the 8 bit telemetry word received in the Command Memory Readout format.

3.6.3.2.2.4 Correcting the Contents of the Memory. If the wrong sequence is loaded or a new sequence is to be inserted into the memory, the CLEAR command can be sent to the CP to load zeroes in all the memory locations. This function can only be accomplished from the Standby state. Alternatively, the SCL can be commanded off from any state, and then commanded on with the CLEAR/ON command to load zeroes into the memory slots.

A single entry in the command memory can be corrected or changed, if necessary. This can be accomplished by first advancing the memory to the address location where the new entry is needed. To do this, the SCL must be commanded into the Standby state (SCL State 15) and then the ADVANCE and STANDBY commands must be sent in pairs to advance the address pointer by one step. The ADVANCE command will move the counter, but the STANDBY command is also needed to complete the sequence. When the SCL is advanced to the location desired, the SCL must be commanded to the Loading state (SCL state 14) and the new entry to be transmitted can then be entered into the memory by the uplink command format.

This process can be very time consuming if a correction or change has to be made more than halfway into the sequence because it takes more time to make a single correction than it does to reload the whole sequence again. Even for a short sequence, it is less time consuming to reload the whole sequence up to the place where the new entry has to be made, than to send twice the number of commands to advance the SCL to the required location. However, for the case of a marginal RF link, it is probably more efficient to make single or multiple corrections in this manner than to risk additional errors in the retransmission of the sequence.

3.6.3.2.2.5 Loading Multiple Sequences. A number of short command sequences can be loaded at the same time and executed individually by using the ADVANCE command. For example, if the second sequence is needed to be performed first, the

ADVANCE command could be used to set the memory address pointer at the address location of the first command word in this sequence. The ADVANCE command must be followed by the STANDBY command to move the pointer one address location. Again, this capability only exists from the Standby state. In addition, if the second sequence has been completed and the first sequence needs to be performed next, the Index command can be sent to place the address pointer at the address location of the first command word of the first sequence.

3.6.3.2.3 Starting the SCL. The SCL can be started in three different ways; namely, by a real-time command, from the 4096 second clock generated in the CP or by the closure of the spacecraft separation switches. The 4096 second clock gives an accurate reference point for delaying the start of a command sequence, and the separation switch input is used for spacecraft separation to start a special command sequence to spin-up the spacecraft. The following sections describe the different ways the SCL can be started.

3.6.3.2.3.1 Immediate Start. To start the SCL in real-time, the IMMEDIATE START command must be sent to the CP. The command memory will begin operating within 400 milliseconds after the uplink command is received by the CP; real-time command processing slots occur every 125 milliseconds. The actual delay time depends on the type of command, the CP in use, and the SCL in use, as seen in Table 3.6.3.2.3-1. After the SCL is started it immediately transfers from the Standby State to the Run State (SCL state 12) and determines whether the entry is a command or a time delay. If the entry is a command, the SCL will transfer into the Process Command state (SCL state 8); if the entry is a time delay, the SCL goes into the Load Timer state (SCL state 13), and extracts that entry in the sequence out of the memory. The memory address pointer, CMEMXC, will then indicate the next location in memory. The two SCLs cannot be started simultaneously with real-time commands. Since it takes 12 seconds to transmit a command to the spacecraft, the closest the two SCLs can be started together

is 12 seconds. However, if the first entry in one command sequence is used to command the start of the other SCL, both SCLs can be started within 250 milliseconds by sending one real-time command.

TABLE 3.6.3.2.3-1

TIME DELAY RANGE FOR COMMAND MEMORY OPERATION, FOLLOWING RECEIPT OF UPLINK COMMAND BY CP

	TIME DELAY (milliseconds)	
TYPE OF COMMAND:	CP1 to SCL1 (or CP2 to SCL2)	CP1 to SCL2 (or CP2 to SCL1)
Discrete	187 to 327	129 to 394
Quantitative	188 to 388	130 to 395

3.6.3.2.3.2 Timed Start. To start the command memory from the 4096 second clock, the TIMED START command must be sent to the CP. The 4096 second clock is a continuous square wave clock that is generated from the 2048 second bit of the spacecraft time code in each TP. The output of each TP is cross-strapped to both CPs as shown in Figure 3.6.1-1. Consequently, both SCLs can be started nearly simultaneously by this method. Since the two CPs operate asynchronously, the start of the two SCLs could be different by a maximum of 125 milliseconds. When the TIMED START command is received by the SCL, it transitions from Standby to the Timed Start 0 state (SCL state 1). The telemetry indication for this state for CLOGXS is "0001". If the 4096 second clock is a logic "0" when this state is entered, the SCL will remain in this state until the clock changes to logic "1" (i.e., from 0 to 2048 seconds in SCL state 1). When the 4096 second clock switches to logic "1", the SCL transfers to the Timed Start 1 state (SCL state 5), and remains there for 2048 seconds until the clock again switches to logic "0". CLOGXS is "0101" in the Timed Start 1 state. If

the 4096 second clock is a logic "1" when the TIMED START command is sent, it will transition immediately from SCL state 1 to SCL state 5, and remain there until the 4096 second clock switches to logic "0" (i.e., duration from 0 to 2048 seconds). The 4096 second clock is a square wave so that there is ample time to receive a telemetry status on the logic state of the clock (i.e., the second MSB of DCLOK2). The SCL transfers immediately into the Run state when the clock switches from logic "1" to logic "0", and the memory begins to process the loaded sequence.

The first command in the sequence will be executed within 129 to 254 milliseconds (for a discrete command) or 130 to 255 milliseconds (for a quantitative command) of the 4096 second clock transition because the CP can only process stored commands every 125 milliseconds. This is one of three error sources in starting the SCL.

The second error source is due to the way the clock is generated in the TP. The TP outputs the 4096 second clock transitions synchronously with the word rate clock. Therefore, the accuracy of this output is a function of operating bit rate at the time of the occurrence of the clock transition and the bit rates used prior to selecting the bit rate used during the transition. The previous bit rates affect the accuracy because some bit rates have word and minor frame periods that are not integer submultiples of 4096 seconds. Table 3.6.3.2.3-2 presents the results of an analysis of the maximum inaccuracy of the 4096 second clock due to the TP. The vertical scale in the matrix is self-explanatory, but the horizontal scale requires some discussion. This scale shows the highest bit rate that the TP was commanded to operate at, from the time it was first turned on, until the time it was commanded to the bit rate at which the 4096 clock transition took place. This does not mean that all bit rates had to be used up to and including the highest bit rate selected, but that at least the highest bit rate was selected at some time prior to going into the bit rate at the transition. This scale starts at

16 bps, because the TP is initialized to start operating at this rate.

A subtlety in these results is that the length of time in the highest bit rate has an effect on the accuracy. If the length of time is just right the error will be zero, but if the length of time is some special value the maximum error given in the table will result. This is best illustrated by an example. Take, for instance, the case

TABLE 3.6.3.2.3-2

ACCURACY OF 4096 SECOND CLOCK

HIGHEST BIT RATE SELECTED SINCE TP TURNON

OPERATING BIT RATE AT 4096 SECOND TRANSITION	16	64	128	$170\frac{2}{3}$	256	$341\frac{1}{3}$	512	$682\frac{2}{3}$	1024	2048
8	0	0	0	0	0	500	500	750	750	750
16	0	0	0	0	0	0	0	250	250	250
64		0	0	0	0	0	0	0	0	0
128			0	0	0	0	0	0	0	0
$170\frac{2}{3}$				$31\frac{1}{2}$	$32\frac{1}{2}$	$31\frac{1}{2}$	$31\frac{1}{2}$	$31\frac{1}{2}$	$31\frac{1}{2}$	$31\frac{1}{2}$
256					0	0	0	0	0	0
$341\frac{1}{3}$						$15\frac{5}{8}$	$15\frac{5}{8}$	$15\frac{5}{8}$	$15\frac{5}{8}$	$15\frac{5}{8}$
512							0	0	0	0
$682\frac{2}{3}$								$7\frac{13}{16}$	$7\frac{13}{16}$	$7\frac{13}{16}$
1024									0	0
2048										0

CONTENT OF EACH CELL IS THE MAXIMUM ERROR IN MILLISECONDS BETWEEN THE DESIRED START TIME AND THE OBSERVANCE BY THE CP OF THE 4096 SEC CLOCK TRANSITION.

where the highest bit rate selected is 341 1/3 bps and the bit rate at the transition is 8 bps. In the matrix, the maximum error is given as 500 milliseconds. One way to arrive at this maximum error (and there are a number of combinations) is to command on the TP and have it operate at 16 bps for 135 minor frames which is 4064 seconds into the first full 4096 second time period because the TP is initialized to start the first 4096 clock period 256 seconds after turn on. Then command on 341 1/3 bps for 21 minor frames, which is an additional 31.5 seconds. This now gives a total time period of 4095.5 seconds. If the TP is now transferred to 8 bps, the 4096 second clock will transition to logic "0" at 4096.5, for an error of 0.5 seconds. The reason for this is the word period at 8 bps is 1 second and the transition must wait until the end of the telemetry word to occur. If the starting bit rate of 16 bps was maintained for 134 minor frames instead of 135 (i.e., 4032 seconds into the period) and the 341 1/3 bps was used for a 42 minor frames rather than 21 (i.e., an additional 63 seconds into the period), the error would be zero after the 8 bps bit rate was commanded into operation. This is because only one second remained in the clock period which is coincident with the word period at 8 bps. Therefore, it is obvious that the error in starting the command memory is highly dependent on the bit rates and time periods over which they are used. Consequently, this effect must be accounted for in programming the start of the memory. In audition, an IMMEDIATE START command can be sent to the SCL when it is either in the Timed Start 0 or Timed Start 1 states to override the use of the 4096 second clock.

The third error source in starting a command sequence or executing a defined function is due to the stability of the oscillator in the CP. If time delays are used in the command sequence, either as the first entry in the sequence or interspersed throughout the sequence, the variation in the oscillator frequency will effect the programmed delays. This will cause the execution of the final event to be in error. The

longer the accumulated time delay, the larger will be the uncertainty. The oscillator stability is 50 ppm worst case.

Additional discussion and examples are contained in Reference 1.5.31.

3.6.3.2.3.3 Separation Switch Start. To start the SCL from the closure of the spacecraft separation switch, the ARM SEPARATION SWITCH command, which is part of the CP CONFIGURE quantitative command, must be sent to the CP. This command must also be sent to the redundant CP to arm the redundant separation switch that, upon closure, activates the sequence loaded in the redundant CP command memory. The purpose of this command is to prevent an inadvertent closure or premature closure of the separation switch from starting the SCL and causing the spacecraft spin-up sequence to begin. The ARM SEPARATION SWITCH command transfers the SCL into Separation State 0 (SCL state 0). The telemetry status of CLOGXS for this state is "0000". If the separation switch is closed (i.e., SSEPXS equal to logic "1") prior to sending this command, the SCL will remain in state 0 and never start the spin-up sequence. Referring to Figure 3.6.1-1, it can be seen that each separation switch is connected to only one CP. Therefore, the premature closure of only one switch prior to the arm command will not prevent the spin-up sequence from occurring because the normal closure of the other switch will initiate a similar sequence loaded in the other SCL memory. In addition, Separation State 0 can be overridden by an IMMEDIATE START command which transfers the SCL to the Run State (SCL state 12). If the separation switch is open prior to sending the ARM command the SCL transfers immediately from Separation State 0 to Separation State 1 state (SCL state 4). The telemetry status as given by CLOGXS for this state is "0100". The SCL remains in this state until separation switch closure, at which time the SCL jumps to the Run State and executes the spin-up sequence. Closing of the separation switch causes a latch in the SCL to be set which prevents switch bounce from affecting the

operation of this function. If needed, the Separation State 1 state can be overridden by the IMMEDIATE START command.

3.6.3.2.4 Stopping the SCL. After the SCL transfers to the Run State, it extracts a command memory word from the memory and begins the processing cycle. If the command memory word is a command (either discrete or quantitative), the SCL transitions to the Process Command State (SCL state 8), and remains there, instead of returning to the Run State, as long as the following command memory entries are commands. The telemetry status will remain constant at "1000" for CLOGXS for all the time that contiguous commands are processed out of the memory. To stop the SCL from this state, the STOP command must be sent to the CP. When this command is received by the SCL, a transfer will be made to the STOP 1 state (SCL state 2), and the SCL will remain in this state until another command is received. The SCL will complete processing of the current command before it switches over to STOP 1. An IMMEDIATE START command, an ADVANCE command or a STANDBY command followed by a RUN command is needed to get out of the STOP 1 state and continue processing. In this state they are effectively the same command. The ADVANCE command causes the SCL processing route to go through the Run State while the IMMEDIATE START command terminates directly in the Process Command or Load Timer status. The sampling of the telemetry system is not fast enough to distinguish the different paths; the whole process takes place in 125 milliseconds including the execution of the command if the next entry is a command. In summary, if either command is sent, the SCL will end up in the Process Command State if the next entry in memory is a command and in the Load Timer State (SCL State 13) if the next entry is a time delay.

When the SCL is in the Load Timer state, independent of how it got there (i.e., from states 2, 8 or 12), the SCL can again be stopped by sending the STOP command. From this state, the SCL will go to the Stop 2 state (SCL state 3) and will give a telemetry readout on CLOGXS of

"0011". In the process of counting down a time delay (T = "1" in Figure 3.6.3.2-1 when the time delay command has been executed, and the time count has not yet decremented to zero), the SCL goes to and remains in the SUB 1 state (SCL state 9) until the time count is decremented to zero (i.e., T becomes "0"). If the time delay has not been completely counted down when a STOP command is issued, the SCL will go to the Stop 2 state and hold the last time count in the Subtract 1 counter (T remains "1"). The time countdown can be resumed by sending the IMMEDIATE START command and the SCL will return to the Sub 1 state. However, if the remaining time count must be skipped, the ADVANCE command will extract the next entry out of the memory by transferring the SCL to the Run State. If, by chance, the STOP command arrives just after the countdown of the time delay has been completed, the SCL will advance to the Stop 1 state. Telemetry status will be recognizable in the Stop 1, Stop 2, Process Command (for many contiguous commands) and Sub 1 (for long time delays) states, but will not be observable for the Run or Load Timer states because the SCL transfers through these states rapidly. A sampling of the status of these states will occur on a very infrequent basis. Likewise for very short time delays or alternating commands and time delays in the memory, the Process Command and Sub 1 states will not be observed on a regular basis.

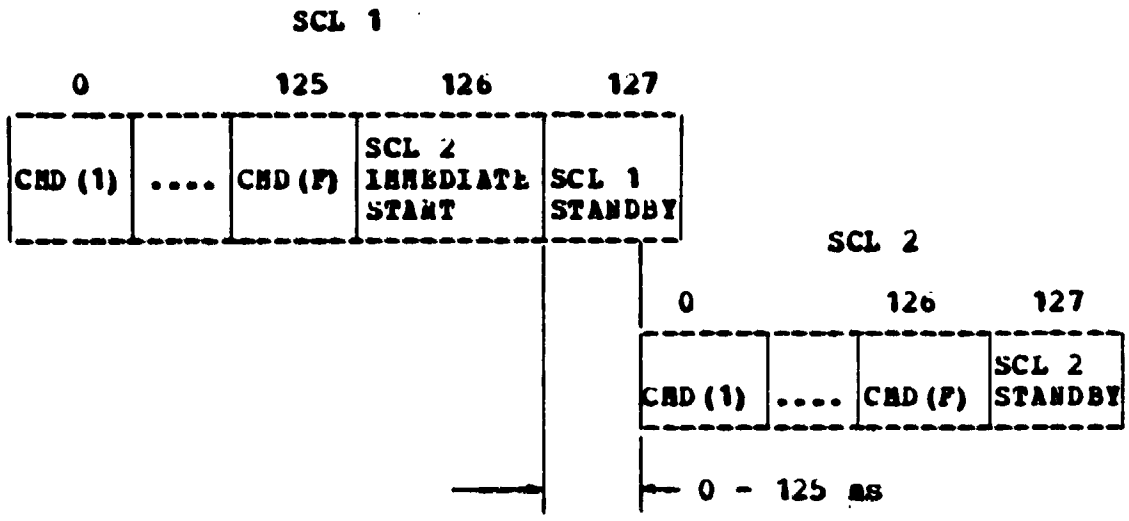
3.6.3.2.5 Considerations for Designing Command Sequences. The method for loading command words into the memory through the use of real-time commands has been discussed in a previous section. However, there are certain facts that should be considered prior to designing a sequence to be placed in the memory. The first fact is that the SCL will continue to process entries from the memory until commanded to stop. The SCL will cycle through all 128 memory slots and then repeat these entries over and over until terminated by a STANDBY, STOP or OFF command. This feature is especially useful for orbit operations when a 24 hour orbital sequence has to be repeated over many orbits. However, if for example, only the

first 20 slots are used in the memory for a particular command sequence, the SCL will process these 20 entries and then step through each of the remaining 108 slots in 125 millisecond steps (assuming each of these slots contains all zeroes). The SCL will repeat this process continuously, unless the 20th entry is one of three stop-type commands mentioned above. An extension of this is when multiple command sequences are loaded into the memory at one time. If each sequence is not terminated by a STANDBY or STOP command, all the sequences will be executed in series and continuously repeated.

A second consideration is that the SCLs in both CPs can be operated simultaneously, as with the spacecraft spin-up operation, but that the commands in each sequence must not address the same COM at the same time i.e., within the same 62.5 msec. period. If a control word is received at a COM from both CPs, the command will not get processed through the COM. The two CPs operate asynchronously; each oscillator in each CP is independent with a 50 ppm stability requirement. As a result, command words occupying the same slot in memory could be executed at the same time, or many seconds apart with a sequence containing many commands and long time delays. This fact must be accounted for in designing parallel sequences.

Another consideration is that both SCLs cannot start precisely at the same time because the stored command processing time slots are not synchronized between the CPs. Therefore, the SCLs could start as much as 125 milliseconds apart with nominal clock frequencies in each CP.

Another use of the SCLs is for series operation to effectively increase the command memory to 256 locations. This can be accomplished by designing the sequence in one command memory to transfer to the other command memory. An example of this operation is given as follows:



This example shows how the two SCLs are used together to execute one 256 command sequence once without repeating it automatically. It is assumed that both SCLs were commanded on through real-time commanding prior to starting the sequence in SCL 1. For SCL 1, memory slots 0 through 125 are used for the first part of the desired sequence, but the last two slots must be reserved to transfer over to SCL 2 and terminate SCL 1. In slot 126, the SCL 2 IMMEDIATE START command is inserted to start SCL 2. Following this command, the SCL 1 STANDBY command is inserted in slot 127 to place SCL 1 in standby and prevent it from repeating its sequence. SCL 2 will start between 0 and 125 milliseconds after the SCL 2 IMMEDIATE START command is executed because of the asynchronous operation of the two CPs. Therefore, the first command word in memory 2 should be a 125 millisecond time delay or a command that is not routed to the same COM through which the SCL 1 STANDBY command is processed. Interference between the SCL 1 STANDBY command and the first command in memory 2 will cause neither command to be executed and result in the SCL 1 sequence being repeated simultaneously with the SCL 2 sequence. The final command in the SCL 2 sequence must be the SCL 2 STANDBY command to prevent repeating the sequence in SCL 2.

If there is no intention to repeat the 256 word sequence, the STANDBY commands should be replaced with OFF commands to conserve power. Should it be desired to repeat the sequence continuously, slot 126 in SCL 2 must contain the START command for SCL 1. There are a number of variations to this procedure, but basically this is the method that must be used to operate both SCLs in series.

3.6.3.2.6 SCL Operating Constraints. The configuration control logic in the CP can receive both real-time commands and stored commands to control the SCL operation. However, there must be a minimum of 125 milliseconds between all commands received by the SCL. If a stored command sequence is being processed, a real-time command should not be sent to the SCL within 125 milliseconds of the time the SCL is outputting a command to itself. Since the real-time time and stored command processing time slots are 62.5 milliseconds apart, it is possible for the SCL to receive two commands as close together as 62.5 milliseconds.

Similarly, if both SCLs are processing commands, one SCL should not send a command to the other SCL within 62.5 milliseconds of the time the other SCL is sending a command to itself or receiving a real-time command.

3.6.3.3 Pyrotechnic Device Operations. The operational aspects related to pyrotechnic devices on the Orbiter spacecraft involve the firing of the Orbit Insertion Motor (OIM) and deployment of scientific instrument devices, such as the OMAG and OETP booms, ONMS breakoff hat and the ORAD antenna mechanism. The following two sections describe these operations.

3.6.3.3.1 OIM Firing Sequence. The firing of the OIM will be accomplished by using the two command memories simultaneously. Only four command words are nominally required in each memory to perform this operation. The four entries are: (1) Time Delay 1 (TD 1), (2) ARM OIM discrete command (ORD11 and ORDA1), (3) Time Delay 2 (TD 2), and (4) FIRE OIM discrete command (ORD12 and ORDA2). This

sequence will be initiated by using the 4096 second clock (i.e., the TIMED START feature of the SCL) to achieve the accuracy needed to fire the OIM for proper insertion into Venus orbit.

The first time, TD 1, should be a delay greater than 36 hours to insure that the sequence can be started before losing communication with the spacecraft for the receiver reverse timeout period of 36 ±4 hours in the event the receiver that is connected to the forward omni fails. If TD1 was less than 36 hours, and there was a failure in the forward omni receiver, then communication to the spacecraft would be blocked off for 36 hours. Because commands would not be received by the spacecraft, the time window for OIM firing would be missed if it occurred within that 36-hour time frame (see Section 4.2.3 for more details). Communication to the spacecraft is only through the forward omni at this time in the mission due to the orientation of the spacecraft. The exact time selected for TD 1 is a function of when the 4096 second occurs in relation to the desired time of OIM firing, the value of TD 2 and an allowance of 250 milliseconds to execute the ARM and FIRE commands. The second time delay, TD 2, provides the necessary time delay between the ARM COMMAND and the FIRE COMMAND. As described in Section 3.6.2.3, there is a time delay of 4 ±2 seconds built into the PCU design between the actual arming of the PCU and the receipt of the ARM command. Consequently, a FIRE command following as closely as two seconds after the ARM command will not be executed. TD 2 is used in conjunction with the timing relationship in the PCU to prevent a premature catastrophic firing of the OIM. With TD 2 programmed to be between 6 and 14.5 seconds, the OIM will be fired normally in the absence of a SCL failure mode.

If one SCL should fail prior to the beginning of the countdown of TD 1 or during TD 1, causing it to be truncated, TD 2 will also be ignored and the FIRE command will follow within 125 milliseconds of the ARM command. For this case, the OIM firing will be inhibited in this SCL and

the parallel sequence contained in the other SCL command memory will perform a nominal firing.

The other case is when the failure of the SCL occurs during the processing of TD 2. If it occurs greater than two seconds into the countdown (worst case), the GIM will be fired, but this accelerated event will not have a significant impact on achieving Venus orbit. Naturally, should the SCL failure happen within the first two seconds of TD 2, no firing will take place.

Since two identical sequences will be loaded into each command memory for this operation, the precaution should again be taken to address the ARM and FIRE commands to different COMs, or stagger these commands in time in the two sequences.

3.6.3.3.2 Scientific Instrument Devices. The firing of the scientific instrument pyrotechnic devices is performed by one arm command and two fire commands. These functions can be performed by real-time commands sent to the spacecraft or under the control of the SCL.

The OMAG boom is deployed shortly after launch. The ARM MAG BOOM, OETP BOOM DEPLOYMENT, ONMS B/O HAT, and OMAD RELEASE MECHANISM command (ORD13 or OMDA3) is first sent to the spacecraft, followed by the RELEASE MAG BOOM fire command (ORD14 or ORDA4). The ARM command arms the two squib drivers associated with these events. There is no telemetry of the "ARM" status for the PCU. The RELEASE MAG BOOM fire command must be contiguous with the arm command for real-time commandings and must be delayed as a minimum of six seconds and a maximum of 14.5 seconds for commanding from the memory to account for the timing in the PCU. The bilevel status of the Mag Boom Deployment is inserted into telemetry as SBOOMS.

After orbit insertion, the ONMS break-off hat, OETP boom and ORAD antenna mechanism are released. This is accomplished by the RELEASE

OMNS B/O HAT, DEPLOY OETP BOOM, UNLOCK ORAD MECHANISM fire command (ORD15 or ORDA5). The same arm command as for the OMAG boom (ORD13 or ORDA3) must be sent prior to sending the above fire command. The same constraints exist for timing delays between the arming and firing for these devices as for the OMAG boom. For an example of these sequences, see Table 4.2.3.

3.6.3.4 Receiver Reverse Operations. The receiver reverse capability in the command subsystem is designed to prevent loss of uplink control of the spacecraft for more than 36 ±4 hours. The CP sends out two discrete commands automatically every 36 ±4 hours when no valid uplink commands are received in that time period. One command is used to select the forward omni and the other is sent to change the receiver/antenna RF switch to the normal or reverse position. When the switch is in the normal position, the reverse pulse is sent out, and vice versa. For redundancy, each CP performs the same function.

When power is initially applied to both CPs the receiver reverse counter in each CP is reset to zero and each begins counting. If a valid command is received by either CP, the counters in both CPs are reset to zero through the cross-strapping between CPs. Under nominal conditions, there is no way to defeat the operation of this function once prime power is applied to the CPs. The application of power may cause the two CPs to initialize in different logic states, i.e., one CP may be in the normal state ready to send out the reverse pulse and the other in the reverse state ready to output the normal pulse. In order to place the CPs in the same state, both CP configure commands, CPCQ1 and CPCQ2, must be sent in the Receiver Normal mode, or both commands must be sent in the Receiver Reverse mode. If this condition is not corrected, the two outputs from the CPs will negate each other and the receiver switch will not change position. The logic state of CP1 and CP2 are given in telemetry as CREV1S and CREV2S, respectively, where logic "1" is normal and logic "0" is reverse.

To insure that the receiver reverse logic in both CPs are in the same state, the CP quantitative command must be sent to each CP. Bits 33 and 34 of the quantitative commands must be set to "01" to obtain the normal state and "10" to get the reverse state. If the "Either CP" address for Bits 39 and 40 (i.e., "11") is not used, then those two bits must be carefully selected to address the CP to which the command is to be sent, i.e., "01" for CP 1 and "10" for CP 2. Even though there is only a single quantitative command to each CP from a different COM, there is a possibility that the receiver reverse function can be defeated through selection of the wrong address bits in the quantitative command sent to a CP. If the wrong CP address is sent, the normal and reverse outputs will be inhibited, but the omni select output will always occur at receiver reverse timeout. The two flip-flops that control the power strobe to the output buffer and the gating of the output pulse are placed in opposite logic states. Consequently, the output pulse will be gated to an "off" output buffer and will be inhibited from going to the "on" output buffer. This occurs basically because one flip-flop is connected to a decoder that senses the address bits and the other is connected to a decoder that does not. The status of the normal/reverse logic is only sensed on one flip-flop so that it is impossible to determine from telemetry if the wrong address bits were sent.

It takes a combination of three human errors and hardware failures for this problem to have a catastrophic effect on the mission. The combination is: (1) a receiver failure to necessitate the use of the receiver reverse capability, (2) a CP failure (or a human error that causes the CP to be addressed incorrectly), and (3) a human error that causes the remaining operational CP (or second operational CP) to be addressed incorrectly. If there is every any uncertainty as to whether the CPs are correctly configured prior to entering a long period without command transmissions, the correct quantitative commands could be retransmitted to

the spacecraft. This is done by sending both CP configure commands, CPCQ1 and CPCQ2, in the Receiver Normal mode, or sending both commands in the Receiver Reverse mode.

To preclude the occurrence or this problem, it is best to always code bits 39 and 40 as "11". This code selects either CP, and bit 15 of the uplink command should be used only to select the desired CP.

Furthermore, if the CP should experience an overcurrent condition that causes the power supply to be turned off by the cyclic fuse, and if the fault was later removed and power re-applied, the receiver reverse counter will be reset to zero and the logic will initialize in some random state. This means the two CPs may not be synchronized and, consequently, the CP CONFIGURE quantitative command should be sent to each CP so that the two CPs will output the same commands simultaneously.

3.6.4 Command Response. Table 3.6.4-1 presents the command responses for the command subsystem. The table lists every command that directly effects the subsystem and the telemetry indication that verifies the proper execution of the command. The mnemonics for each of the command and telemetry parameters are also included.

TABLE 3.6.4-1

COMMAND RESPONSES FOR COMMAND SUBSYSTEM

COMMAND			TELEMETRY VERIFICATION	
MNEMONIC	TITLE	RESPONSE	MNEMONIC	INDICATION AND/OR REMARKS
The sixth alphanumeric is "A" for CP1, or "B" for CP2.	Stored Command	The SCL of the appropriate CP, if it is in the Load state, will accept the stored command, and its address pointer will advance by one count for each stored command.	CLOG1S CLOG2S	SCL State: State 14. (Load)
			CMEM1C	Command Memory Address Pointer: Command memory address will advance one count for each memory load command.
CMTQ1 for CP1; or CMTQ2 for C P2	Stored Time Delay Command.	No apparent difference from response to any stored command (above).		• Same as for any stored command (above). • Memory verification will indicate if correct time delay was loaded.
CP Configure (SCL Command) (CPCQ1 or CPCQ2) STORCMD = XXXXX.	SCL Standby (0001) (STORCMD = STDBY)	If SCL is ON and in any state except 10, SCL will go to State 15.	CLOG1S CLOG2S	SCL State: State 15.
	SCL Immediate Start (0010) (STORCMD = IMMED)	If SCL is ON and in States 0, 1, 4, 5 or 15, SCL will go to State 12 for 62.5 ms and then switch among States 8, 9, 12 and 13, depending on the stored commands being processed. If SCL is ON and in State 2 and the next memory entry is a command (MSB = 1), SCL will go to State 8 and continue processing. If SCL is ON and in State 2 and the next memory entry is a delay (MSB = 0), SCL will go to State 13 and continue processing. If SCL is ON and in State 3, SCL will go to State 9 and continue processing the delay stored in the time delay counter.	CLOG1S CLOG2S	SCL State: States 8, 9, 12 and 13, depending on stored commands being processed and TM sampling of state.
			CMEM1C CMEM2C	Command Memory Address Pointer: Command memory address will be advancing as commands and time delays are being processed.
				Other TM: Other Subsystems will be commanded by the stored commands.

TABLE 3.6.4-1. COMMAND RESPONSES FOR COMMAND SUBSYSTEM (Continued)

COMMAND			TELEMETRY VERIFICATION	
MNEMONIC	TITLE	RESPONSE	MNEMONIC	INDICATION AND/OR REMARKS
CP Configure (SCLCMD) (Continued) (CPCQ1 or CPCQ2) STORCMD = XXXXX.	SCL Timed Start (0011) (STORCMD = TIMED)	If SCL is ON and in any state except 6, 10, 11 or 14, SCL will go to State 1. If and when the 4096 clock is one, SCL will go to State 5. When the 4096 clock (i.e., the second MSB of DCLOK2) goes from one to zero, SCL will go from State 5 to State 12 for 62.5 ms then switch among States 8, 9, 12 and 13, depending on the stored commands being processed.	CLOG1S CLOG2S	SCL State: State 1 until the 4096 clock (i.e., the second MSB of DCLOK2) goes to one and then the SCL goes to 5. If clock is already one, the transition from State 1 to 5 will occur at once. When clock goes from one to zero, switching between States 8, 9, 12 and 13 will occur, depending on stored commands being processed and TM sampling of state.
			CMEM1C CMEM2C	Command Memory Address Pointer: When 4096 clock goes from one to zero, command memory address will start advancing as commands and time delays are being processed.
				Other TM: Other subsystems will be commanded by the stored commands.
	SCL Load (0100) (STORCMD = LOAD)	If SCL is ON and in State 15, SCL will go to State 14. SCL must be in State 14 to load the memory by sending memory load commands.	CLOG1S CLOG2S	SCL State: State 14.
			CMEM1C CMEM2C	Command Memory Address Pointer: Command memory address will advance one count for each memory load command.
	SCL Advance (0101) (STORCMD = ADVAN)	If SCL is ON and in State 15, SCL will go to State 11. The memory is shifted 24 bits which results in a one-word advance in memory. SCL will then go to State 6 and remain there until a standby command is received.	CLOG1S CLOG2S	SCL State: State 6 (advance complete). State 11 will probably not be observed by TM.
			CMEM1C CMEM2C	Command Memory Address Point: Advances count by one for each issuance of advance/standby command pair.

TABLE 3.6.4-1. COMMAND RESPONSES FOR COMMAND SUBSYSTEM (Continued)

COMMAND			TELEMETRY VERIFICATION	
MNEMONIC	TITLE	RESPONSE	MNEMONIC	INDICATION AND/OR REMARKS
CP Configure (SCL CMD) (Continued) (CPCQ1 or CPCQ2) STORCMD = XXXXX.	SCL Advance (0101) (STORCMD = ADVAN) (Continued)	If SCL is ON and in State 2 or 3, the SCL then goes to State 12 for 62.5 ms, and switches among States 8, 9, 12 and 13, depending on the stored commands being processed.	CLOG1S CLOG2S	SCL State: States 8, 9, 12 and 13, depending on stored commands being processed and TM sampling of state.
			CMEM1C CMEM2C	Command Memory Address Pointer: Command memory address will be advancing as commands and time delays are being processed.
				Other TM: Other subsystems will be commanded by the stored commands.
	SCL Stop (0110) (STORCMD = STOP)	If SCL is ON and in State 8, SCL will go to State 2. If SCL is ON and in State 13 or 9, and the time delay counter is zero, SCL will go to State 2. If SCL is ON and in State 13 or 9 and the time delay counter is not zero, SCL will go to State 3. The counter will stop counting and retain the count that existed when the STOP command was received.	CLOG1S CLOG2S	SCL State: State 2 or 3, depending on conditions listed under command response.
	SCL Arm Separation Switch (0111) (STORCMD = ARMSP)	If SCL is ON and in State 15, and separation switch is closed, SCL will go to State 0 and remain indefinitely until commanded out of that state. If SCL is ON and in State 15, and separation switch is open, SCL will go to State 0 and them immediately to State 4. When the separation switch is closed, SCL will go from State 4 to State 12 and start processing stored commands as described under SCL Immediate Start command. (Closing of the separation switch sets a latch in SCL which can only be reset by removing essential bus voltage from the unit which is not possible by command).	CLOG1S CLOG2S	SCL State: State 0; if the separation switch is closed (i.e., SSEPXS=1). State 4 if the separation switch is open (i.e., SSEPXS=0).

TABLE 3.6.4-1. COMMAND RESPONSES FOR COMMAND SUBSYSTEM (Continued)

COMMAND			TELEMETRY VERIFICATION	
MNEMONIC	TITLE	RESPONSE	MNEMONIC	INDICATION AND/OR REMARKS
CP Configure (SCL CMD) (Continued) (CPCQ1 or CPCQ2 STORCMD = XXXXX.	SCL Clear/ON (1000 or 1001) (STORCMD = CLER1 or CLER2)	If SCL is OFF, SCL power will be turned ON with SCL in State 15. Command memory will be cleared and the command memory address pointer will be zero. If SCL is ON and in State 15, command memory will be cleared and the command memory address pointer will be zero.	CLOG1S CLOG2S	SCL State: State 15.
			CSCL1S CSCL2S	SCL State: ON (Logic 1)
			CMEM1C CMEM2C	Command Memory Address Pointer: Zero.
			CMEM1 CMEM2	Command Memory Readout: Entire memory reads zero.
			PBUSLI	Spacecraft Loads Current: An increase of about 72 milliamperes (1 LSB) when SCL is turned ON.
			PLIMTI	Bus Voltage Limiter Current: Decreases as much as PBUSLI has increased, if sufficient surplus solar panel current is present.
	SCL Index (1010 or 1011) (STORCMD = INDX1 or INDX2)	If SCL is ON and in States 15 or 10, command memory will be end around shifted until the first stored word occupies address 0. SCL will go to State 15.	CLOG1S CLOG2S	SCL State: State 15.
			CMEM1C CMEM2C	Command Memory Address Pointer: Zero.
	SCL Read (1106 or 1101) (STORCMD = READ1 or READ2)	If SCL is ON and in State 15, SCL will go to State 10. SCL must be in State 10 for memory verification. SCL will go from State 10 to State 15 when the address pointer goes to zero during memory verification or by an index command.	CLOG1S CLOG2S	SCL State: State 10.
			CMEM1C CMEM2C	Command Memory Address Pointer: Command memory address will be advancing during memory verification.

TABLE 3.6.4-1. COMMAND RESPONSES FOR COMMAND SUBSYSTEM (Continued)

COMMAND			TELEMETRY VERIFICATION	
MNEMONIC	TITLE	RESPONSE	MNEMONIC	INDICATION AND/OR REMARKS
CP Configure (SCL CMD) (Continued) (CPCQ1 or CPCQ2) STORCMD = XXXXX.	SCL OFF (1110 or 1111). (STORCMD = OFF1 or OFF2)	SCL Power will be turned OFF.	PBUSLI	Spacecraft Loads Current: A decrease of about 72 milliamperes (1 LSB) when SCL is turned OFF.
			PLIMTI	Bus Voltage Limiter Current: Increases as much as PBUSLI has decreased, if sufficient surplus solar panel current is present.
			CSCL1S CSCL2S	SCL Status: OFF (Logic 0)
			CLOG1S CLOG2S	SCL Status: All Logic 1 bits.
CP Configure (Receiver and Telemetry Commands) (CPCQ1 or (CPCQ2) RCV= XXXX	Receiver Normal (RCV = NORML)	The Receiver Normal command will set the Command Processor Receiver Reverse State to NORMAL and generate an envelope to select the normal omni antenna/receiver configuration.	CREV1S CREV2S	CP Receiver Reverse State: Normal (Logic 1).
			RRCVRS	RF Switch Position: Receiver 2 to despin antennas and Receiver 1 to aft antenna (Logic 1).
	Receiver Reverse (RCV = RVRSE)	The Receiver Reverse command will set the Command Processor Receiver Reverse State to REVERSE and generate an envelope to select the reverse omni antenna/receiver configuration.	CREV1S CREV2S	CP Receiver Reverse State: Reverse (Logic 0).
			RRCVRS	RF Switch Position: Receiver 1 to despin antennas and Receiver 2 to aft antenna (Logic 0).
	Clear Command Counter and Reject Flag (RCV = CLEAR)	The command counter and reject flag in the CP will be set to zero.	CREJ1S CREJ2S	Command Reject: Zero
			CCMD1C CCMD2C	Command Counter: Zero

TABLE 3.6.4-1. COMMAND RESPONSES FOR COMMAND SUBSYSTEM (Continued)

COMMAND			TELEMETRY VERIFICATION	
MNEMONIC	TITLE	RESPONSE	MNEMONIC	INDICATION AND/OR REMARKS
CP Configure (CP Select) (CPCQ1 or (CPCQ2) CMD PROC SEL = XXXX	Command Processor Select (CMD PROCSEL = None/No. 1/No. 2/ Either 0/1/2/3)	Sets up logic to configure receiver reverse for normal or reverse outputs.	N/A	No telemetry verification.
COM1Ø COM3Ø COM5Ø	COM 1 and 2 OFF COM 3 and 4 OFF COM 5 and 6 OFF	For some failure modes in the COM which leave the COM powered ON, the COM can still be used by sending a COM OFF command after each command using that COM. In normal operation, the COM is automatically turned OFF at completion of command processing.	N/A	If a COM fails to respond to a command, send a COM OFF command and try to send the command again. If the COM responds to the command, the COM can still be used, by sending a COM OFF Command after each command using the COM.
MEM1Ø	SCL 1 OFF	Discrete command which will turn SCL 1 Power OFF.	CSCL1S	SCL 1 Status: Zero
			PBUSLI	Spacecraft Loads Current: A decrease of about 72 milliamperes (1 LSB) when SCL is turned OFF.
			PLIMTI	Bus Voltage Limiter Current: Increases as much as PBUSLI has decreased, if sufficient surplus solar panel current is present.
MFM2Ø	SCL 2 OFF	Discrete command which will turn SCL 2 Power OFF.	CSCL2S	SCL 2 Status: Zero
			PBUSLI	Spacecraft Loads Current: A decrease of about 72 milliamperes (1 LSB) when SCL is turned OFF.
			PLIMTI	Bus Voltage Limiter Current: Increases as much as PBUSLI has decreased, if sufficient surplus solar panel current is present.

TABLE 3.6.4-1. COMMAND RESPONSES FOR COMMAND SUBSYSTEM (Continued)

COMMAND			TELEMETRY VERIFICATION	
MNEMONIC	TITLE	RESPONSE	MNEMONIC	INDICATION AND/OR REMARKS
ORD11 ORDA1	Arm Orbit Insertion Motor	Provides power to the squib drivers for firing Orbit Insertion Motor, starting nominally between 4.0 seconds and 18.5 seconds after the command.	N/A	Verified if associated fire command is accepted. There is no telemetry indication of PCU arm/disarm functions.
ORD12 ORDA2	Fire Orbit Insertion Motor	If associated squib drivers are armed, this command fires Orbit Insertion Motor squibs.	N/A	Verified by proper S/C orbit response. Some control subsystem telemetry may show transients indicating OIM fired.
ORD13 ORDA3	Arm Mag Boom, OETP Boom, Deployment, ONMS B/O Hat and ORAD Release Mechanism	Provide power to the squib drivers for specified functions starting 4.0 seconds after the command and ending 18.5 seconds after the command.	N/A	Verified if associated fire command is accepted. There is no telemetry indication of PCU arm/disarm functions.
ORD14 ORDA4	Release Mag Boom	If associated squib drivers are armed, this command fires squibs to release Mag Boom.	SBOOMS	Magnetometer Boom Deployed Status: Deployed (Logic 1)
ORD15 ORDA5	Release ONMS B/O Hat, Deploy OETP Boom, Unlock ORAD Mechanism	If associated squib drivers are armed, this command fires squibs for specified functions.	N/A	Indirectly verified by proper instrument operation.
ORD1Ø ORDAØ	Disarm PCU	For some failure modes in which the timer in the PCU fails to disarm the squib drivers, this command will disarm them if sent after 18.5 seconds from arming.	N/A	There is no telemetry indication of PCU arm/disarm functions.

This Figure is a Foldout.

See APPENDIX C

Figure 3.6.1-1.

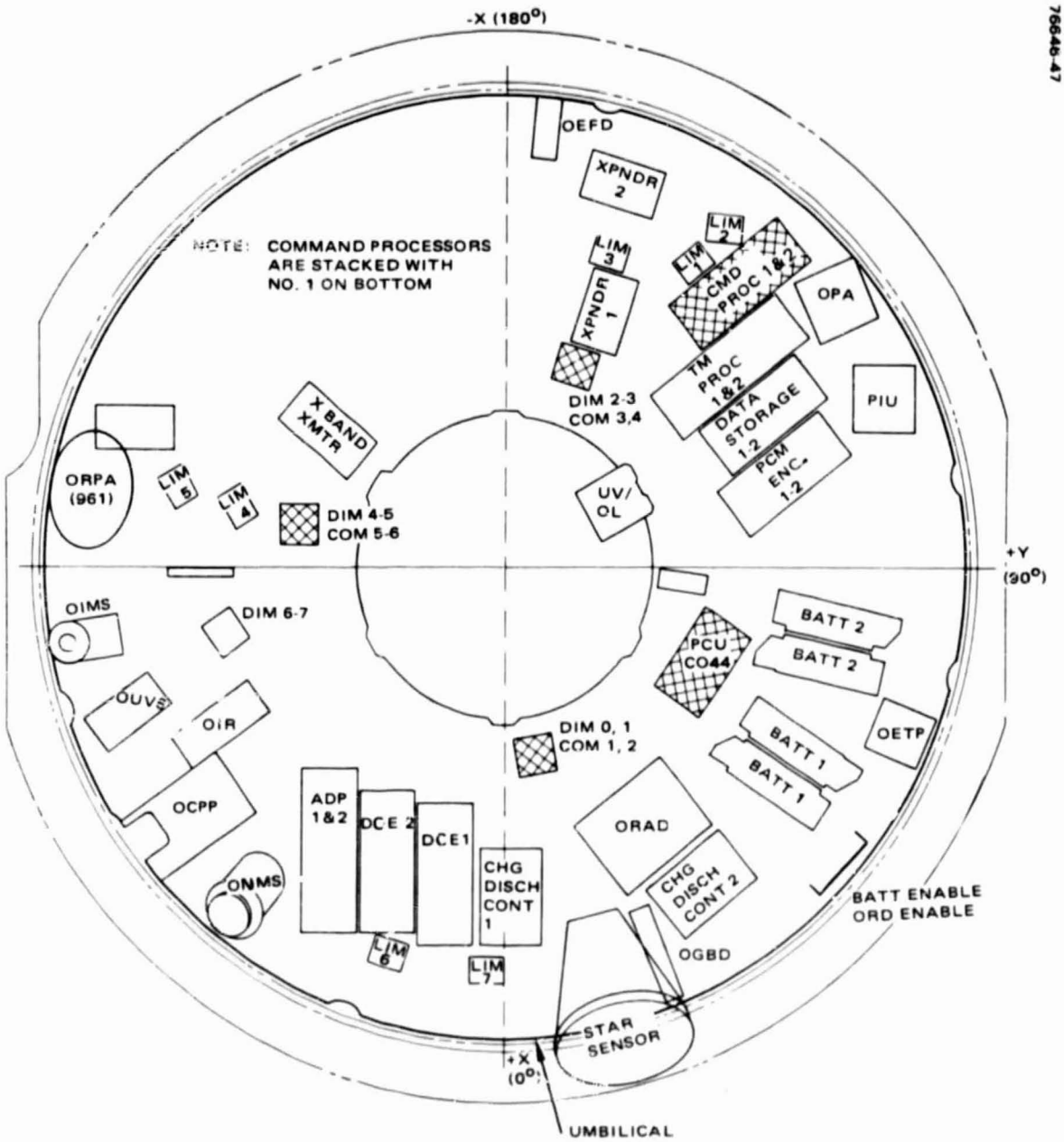


FIGURE 3.6.1-2. SHELF LAYOUT OF COMMAND SUBSYSTEM

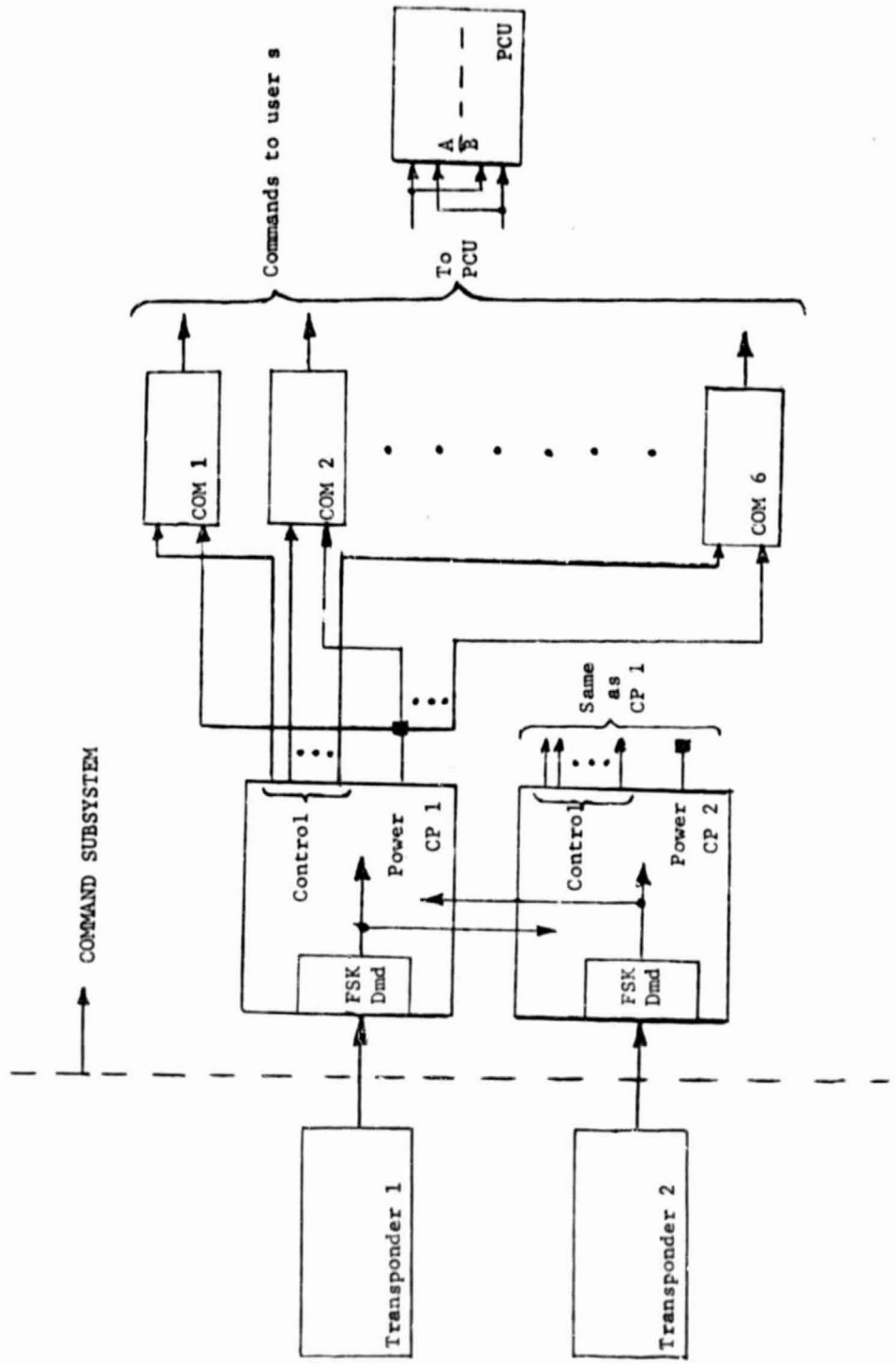


Figure 3.6.1-3. Orbiter Command Subsystem Redundancy Block Diagram

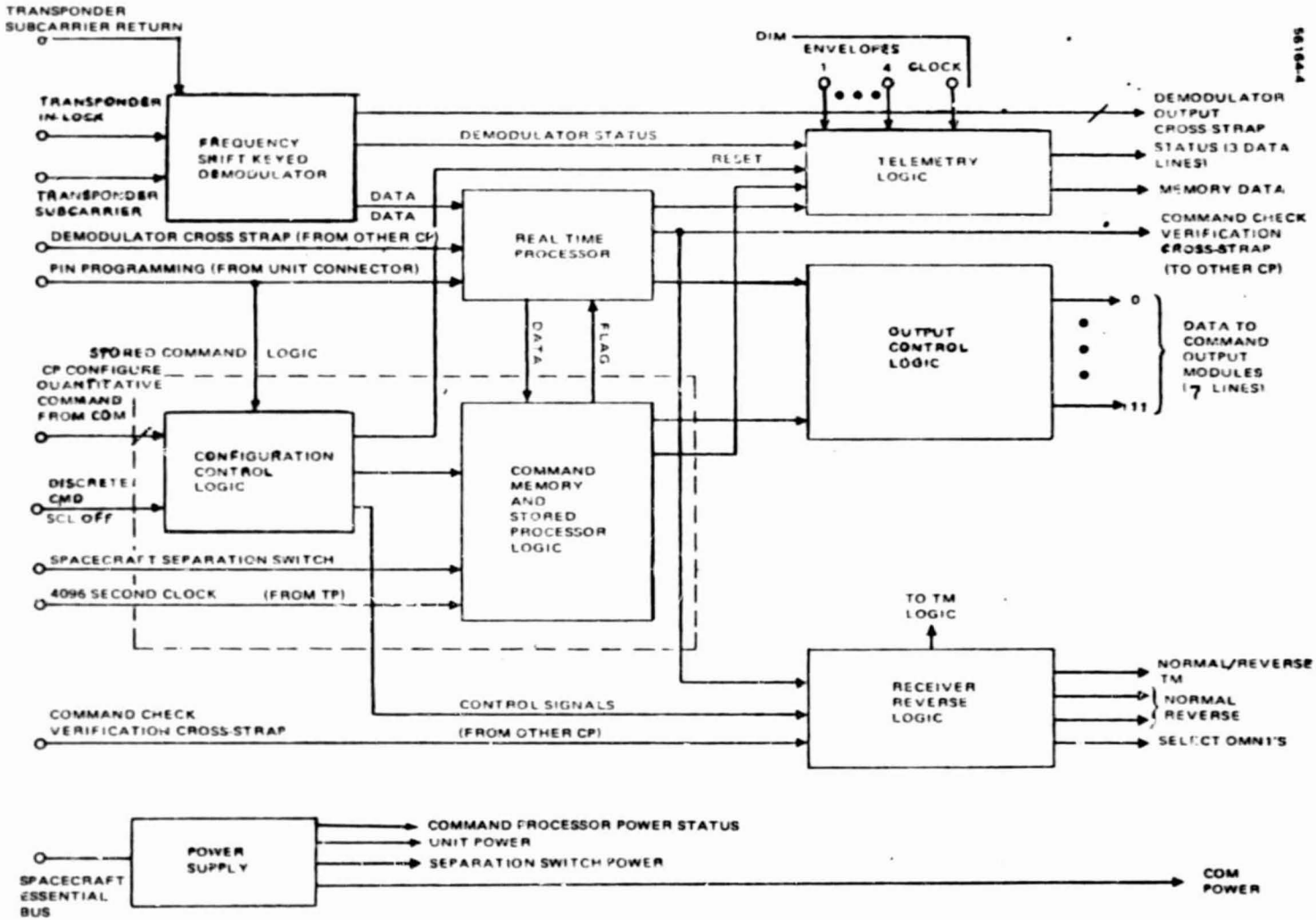


Figure 3.6.2.1-1. Functional Block Diagram of Single Command Processor

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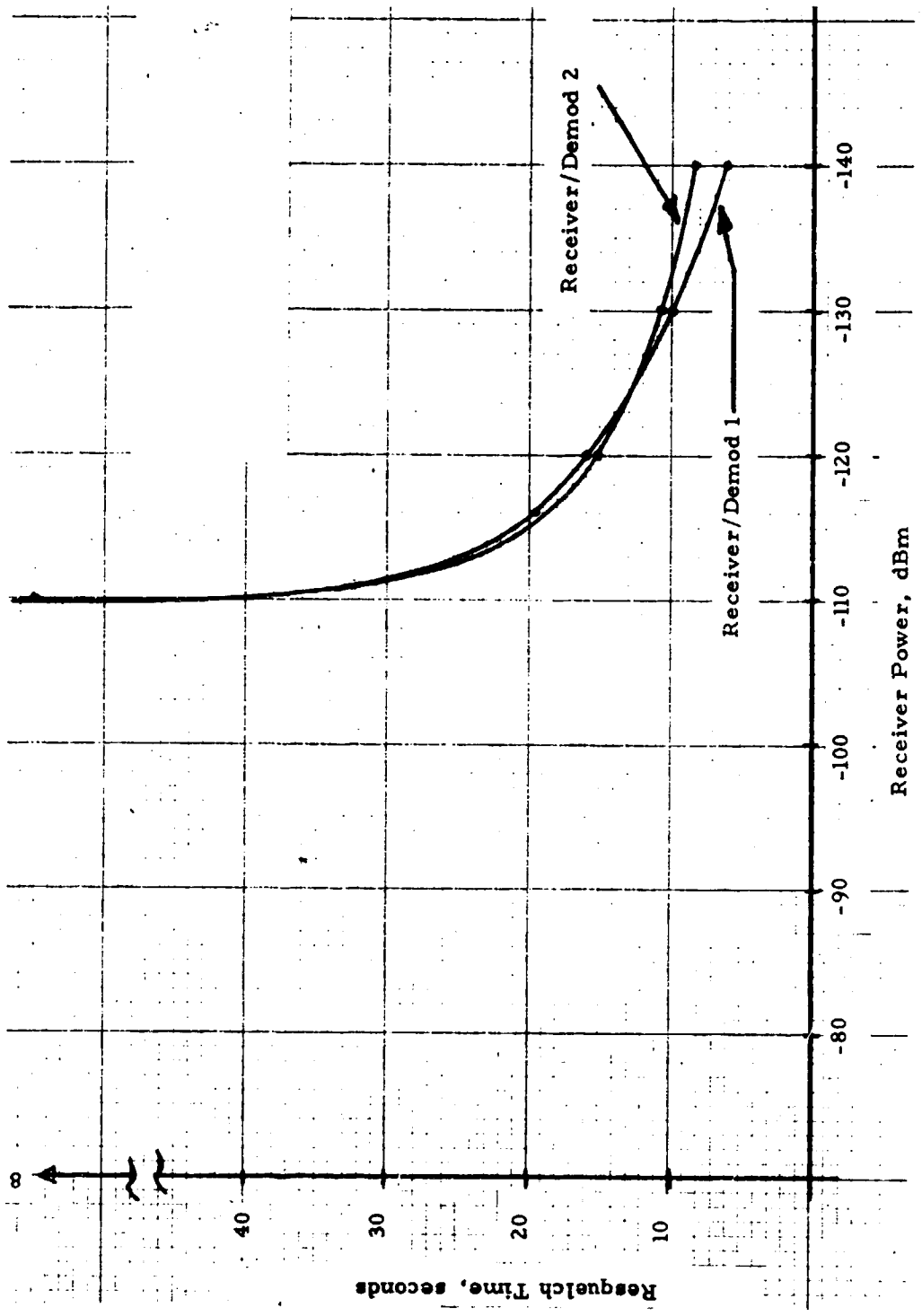


Figure 3.6.2.1.1-1. Resquelch Time Versus Receiver Input Power (Worst Case)

C-6

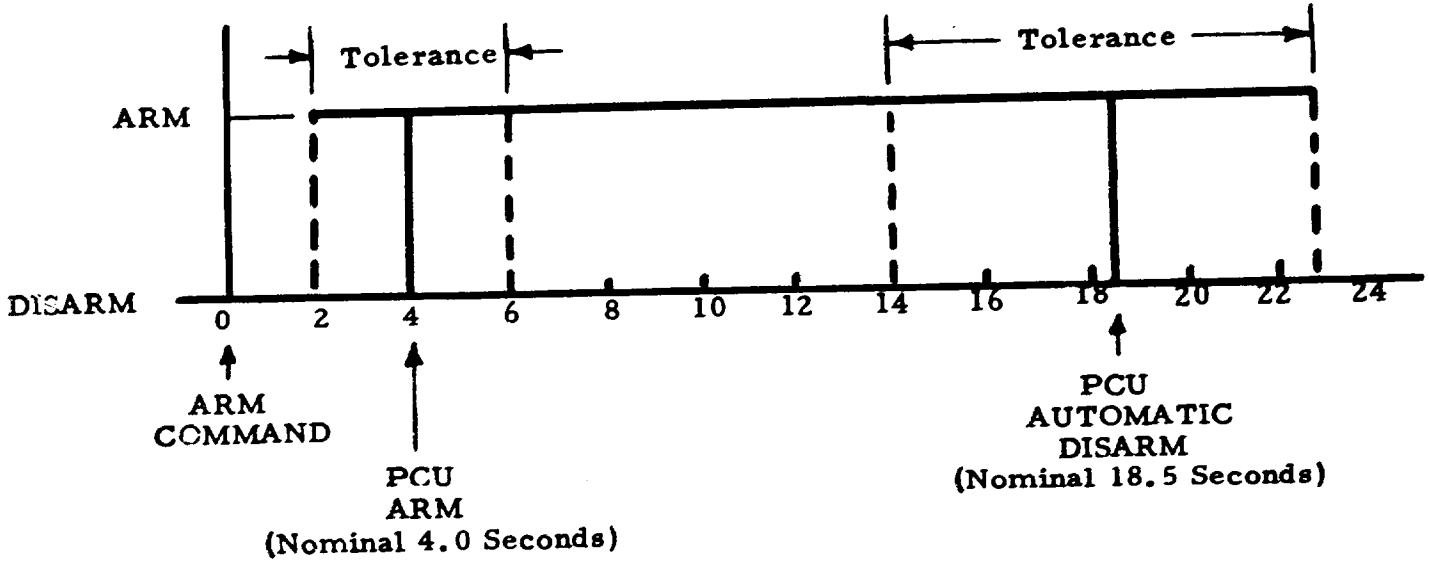
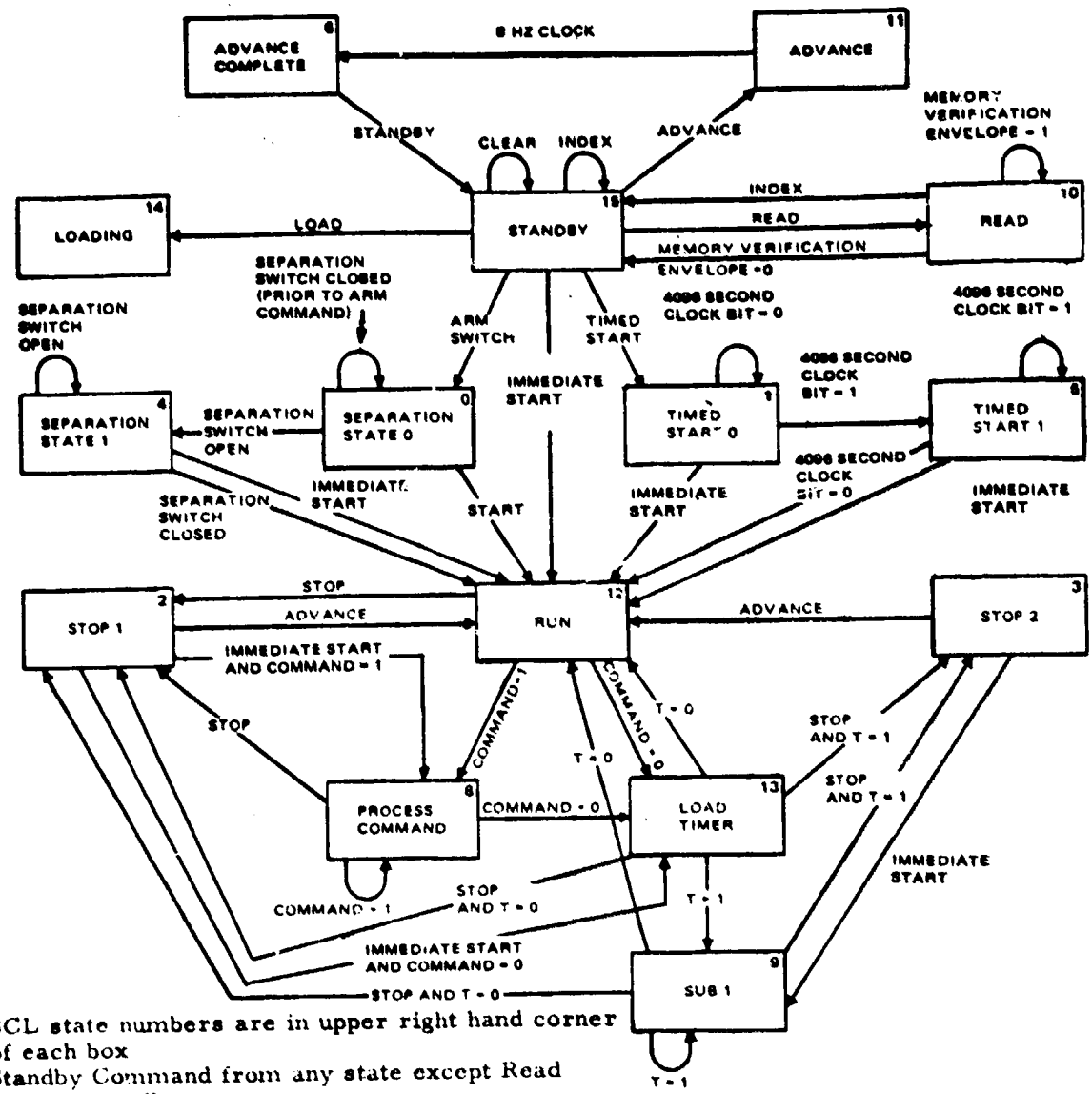


Figure 3.6.2.3-1. PCU Timing Relationships



- SCL state numbers are in upper right hand corner of each box
- Standby Command from any state except Read goes to standby
- Timed start command from any state except 6, 10, 11 and 14 goes to state 1
- T=1 when a time delay command has been executed, and the time delay has not yet decremented to zero
- T=0 when the time delay has decremented to zero
- COMMAND=0 for a time delay entry
- COMMAND=1 for a command entry.

Figure 3.6.3.2-1. SCL State Diagram

3.7 COMMUNICATIONS SUBSYSTEM

3.7.1 Subsystem Description. The Orbiter communication subsystem generates an S-band downlink signal, phase modulates it with the PSK modulated telemetry subcarrier from the telemetry processor, and transmits it, with nominally either 10 or 20 watts of RF power fed into one of several antennas. The subsystem also receives and demodulates the uplink phase modulated S-band signal and provides the PSK tones to the command processor. In addition, the subsystem provides an unmodulated X-band downlink carrier from a one watt (into the antenna) transmitter for the occultation experiment. The frequency of this X-band carrier is coherent with the S-band downlink frequency. Hemispherical transmit and receive coverage is provided for the S-band signals by the multiple antennas shown in Figure 3.7.1-1. The forward omni covers approximately zero to 95° from the +Z axis, and 95° to 180° is covered by the aft omni. The X-band signal is radiated by the high gain antenna, which also provides narrow beam S-band coverage by virtue of the dual frequency, S/X-band feed. A doughnut shaped pattern (symmetric with the spacecraft Z axis) is provided by the high gain backup antenna, that may be utilized if the HGA is spun up. Selection of the desired antenna and transmitter/receiver combination is accomplished by means of the RF switches, as shown in the functional block diagram of Figure 3.7.1-2 and the appropriate "ON/OFF" command of the power amplifiers and transponder exciters. Transponder 1 receiver or transponder 2 receiver selection is done by tuning the uplink frequency to the desired receiver's unique operating frequency (2112.289352 MHz and 2112.630401 MHz, respectively) since both receivers are always on. The "in-lock" status signal from each receiver is sent to the command processor to guard against the noise output of the unused receiver from being processed by the command processor. The transmitter consists of four 10-watt solid-state power amplifiers which can be operated one at a time or in pairs. When operated in pairs, their outputs are summed by a 3 dB hybrid to provide

the 20 watt S-band RF capability. Each of the four power amplifiers is driven by a dedicated solid-state driver amplifier that receives its dc power from the power amplifier power converter that it is driving. All four driver amplifiers are driven simultaneously by means of the four-way power divider (6 dB hybrid, whose input is selected by the SPDT switch (S-switch) to be either one of the transponder exciters). A simultaneous transmit and receive capability through any of the antennas is provided by the two diplexers.

The communication subsystem units are arranged on the spinning equipment shelf as shown in Figure 3.7.1-3. Placement of the units is intended to minimize coax line length and RF insertion loss. The power amplifiers, due to their high internal heat dissipation, have been located over the shelf thermal louvers. Since the forward omni antenna, high gain antenna, and high gain backup antenna are all located on the despun portion of the spacecraft, RF power is coupled to these antennas via the rotary joint. The rotary joint contains one channel for S-band (destined for any one of the three antennas) and one channel for X-band (destined for the HGA only). Thus, selection of the three despun antennas is accomplished by the despun switches mounted on the antenna mast. Commands for these switches are routed via slip rings in the BAPTA.

A detailed Logic Flow diagram for the subsystem is shown in Figure 3.7.1-4 (Appendix C). It identifies all command inputs to their terminal points, and all telemetry outputs from their generic sources.

3.7.2 Unit Descriptions

3.7.2.1 Transponder. The transponder consists of a receiver section and an exciter section. A functional block diagram of the transponder is shown in Figure 3.7.2.1-1. The frequency plan shown in the block diagram is given in terms of the S-band downlink frequency, nominally 2294 MHz, as indicated by 240 F_1. The frequency

indicated by F_2 is the second IF frequency and is nominally 12.25 MHz.

3.7.2.1.1 Transponder Receiver. The transponder receiver, hereafter referred to as the receiver, is an S-band phase lock receiver that demodulates the uplink command tones from the carrier and provides a frequency source for the exciter that is phase coherent with the uplink carrier. The uplink RF is routed first to the RF converter module. This module limits the receiver predetection bandwidth to 60 MHz by means of the preselector filter.

After conversion to the first IF frequency of 13 F_1, the signal is routed to the first IF amplifier where it is amplified and converted to the second IF frequency of F_2. The second IF amplifier further amplifies the signal, performs narrow-band filtering and provides a hard limited output and a linear output to the phase detector assembly.

The detector demodulates the linear input to provide a coherent AGC and the in-lock signals. The hard limited input is demodulated to provide the error signals for the phase lock loop and the command subcarrier. The phase lock error signal is conditioned by the second order loop filter in the VCO module. The output of the loop filter controls the VCO phase and frequency and is also telemetered as the static phase error (SPE). The VCO output frequency is multiplied and translated (via a fixed oscillator at 2 F_2) in the local oscillator assembly to provide the mixer frequencies of 26 F_1 to the RF converter, and (13F_1) - F_2 to the first IF amplifier modules. It also provides the reference frequency (2 F_2) to the detector module for the coherent phase detection process.

Power for the receiver modules is provided by a power converter that converts the nominal 28 volt input voltage (essential bus) to +9V (receiver), +9V (relay voltage for exciter control functions) and -6V (receiver). Overcurrent protection for the receiver is provided by a current limiter set

at 200 ±30 ma. Receiver fusing is included externally in the spacecraft harness (3/8 amp fuse). The receiver cannot be commanded OFF, and is ON whenever dc is applied to the dc input. Since both receivers are connected to the spacecraft essential bus, they are always ON and the receiver addressing is via appropriate uplink frequency selection. The important operational characteristics of the receiver are as follows:

• Noise Figure	7.4 dB
• Carrier Threshold (at receiver input terminal)	-153 dBm
• Threshold Loop Noise Bandwidth ($2B_{LO}$)	18 Hz
• Tracking Capability	±200 kHz at 55 Hz/sec at -140 dBm input level
• AGC Range	100 dB minimum
• Command Capability	None
• Command Threshold (Calculated, for BER = 3.5×10^{-4})	-118.6 dBm/m² (worst case flux density) (-136.61 dBm at the receive antenna output port).

Beat Lock Frequency Stability (Also See Figure 3.7.2.1-2):

Condition	Transponder No. 1 (SN 3)	Transponder No. 2 (SN 4)
-4°F to +126°F	22.0 kHz	15.4 kHz
+32°F to +90°F	6.6 kHz	4.1 kHz
at +81°F for 10 hours	3.8 kHz	2.3 kHz

Telemetry Information (More details are found in Appendix A):

- Automatic Gain Control (AGC)
- Static Phase Error (SPE)
- In-lock Status
- VCO Temperature

3.7.2.1.2 Transponder Exciter. The exciter provides an S-band signal that is phase modulated by the telemetry subcarrier. The frequency source for the exciter is either the receiver VCO output, or the redundant transponder VCO, or an auxiliary oscillator in the exciter. The frequency source is selected according to the logic shown in Table 3.7.2-1. The auxiliary oscillator is ON whenever the exciter is ON, regardless of whether the receiver is locked to the uplink frequency or not. The inhibit coherent mode command causes the exciter to use the auxiliary oscillator frequency source when the receiver is locked and the transfer to VCO command (test only, shorted out by flight test access connector) causes the exciter to use the unlocked VCO as the frequency source. Selection of the frequency source is done by a solid-state switch, with both sources always present at the input to the switch.

A 30 F_1 signal, generated in the X11/X15 multiplier, is applied to the phase modulator where it is phase modulated by the telemetry subcarrier. The telemetry subcarrier from the telemetry processor is regenerated in the exciter in order to control the amplitude of the modulating signal that is applied to the phase modulator. This makes the modulation index stability independent of the interface between the telemetry processor and the communication subsystem, thus resulting in a stable modulation index. The commandable selection of modulation index is accomplished by attenuating the regenerated subcarrier with a variable voltage divider. The X8 multiplier/S-band power amplifier generates the 240 F_1 output, amplifies it and keeps the output level constant by means of an automatic leveling loop.

A dc to dc power converter provides +11, +9 and -6 Vdc for the exciter modules. Overcurrent protection for the exciter is provided by a current limiter set for 360 ±50 ma. Exciter fusing is included externally in the spacecraft harness (3/4 amp fuse). The exciter is commanded ON/OFF by the command that turns the power converter ON/OFF. The ON/OFF command is such

TABLE 3.7.2-1

TRANSPONDER EXCITER FREQUENCY SOURCE SELECTION LOGIC

Telemetered Command Status		Telemetered In Lock Status		Selected Frequency Source for Downlink Only
Inhibit Coherent Mode (RCOHXS)3*	Transfer* to VCO (RVCOXS) 3*	Own Receiver In Lock (RLOKXS)3*	Redundant Receiver In Lock (RLOKYS)4*	
0	0	No	No	Auxiliary Oscillator 2*
0	0	No	Yes	Redundant Receiver VCO
0	0	Yes	Either Yes or No	Own Receiver VCO
0	1	No	No	Own Receiver VCO
0	1	No	Yes	Redundant Receiver VCO
0	1	Yes	Either Yes or No	Own Receiver VCO
1	Either 0 or 1	Either Yes or No	Either Yes or No	Auxiliary Oscillator 2*

* Test only. This command is disabled (set to a permanent "0" state) by ground connection made by test access flight plug.

2* Auxiliary oscillator is always the one on in the exciter of interest, since only one exciter can be commanded on at a time.

3* X = 1 or 2.

4* Y = Designation for Redundant Receiver.

that only one exciter can be on at a time, but both can be off simultaneously.

The important operational characteristics of the exciter are as follows (further details of commands and telemetry are in Section 3.7.4 and Appendix A, respectively):

- Modulation Index — Commandable to 1.18 radians or 0.65 radians (67.6° or 37.2° respectively)
- Auxiliary Oscillator Frequency Stability vs Temperature:

Temperature (°F)	Transponder 1 (SN 3)	Transponder 2 (SN 4)
-4°	2293.872960	2294.246640
+14°	2293.883520	2294.256720
+32°	2293.888920	2294.261520
+41°	2293.890360	2294.262480
+50°	2293.890720	2294.262480
+61°	2293.890720	2294.261880
+68°	2293.890120	2294.260560
+77°	2293.888920	2294.259240
+90°	2293.887720	2294.256600
+126°	2293.885920	2294.252280

- Command Capability (Refer to Section 3.7.4) for details)
 - Exciter 1 ON, 2 OFF
 - Exciter 2 ON, 1 OFF
 - Exciters 1 and 2 OFF
 - High Mod Index
 - Low Mod Index
 - Inhibit Coherent Mode
 - Restore Coherent Mode
 - Transfer to VCO ON (test only)
 - Transfer to VCO OFF (test only)
- Telemetry Information (Refer to Appendix A for details)
 - Auxiliary Oscillator Temperature
 - Exciter ON/OFF Status
 - Inhibit/Restore Coherent Mode Status
 - Transfer to VCO ON/OFF Status

3.7.2.2 Driver Amplifier. The driver amplifier consists of three transistor stages that amplify the +14 dBm input from the exciter to approximately +34 dBm. The first stage operates as Class B and the second and third stages operate as Class C. The dc power for the driver amplifier is supplied by the power amplifier dc regulator. There are no fuses or current limiting in the driver amplifier. There are no commands or telemetry for the driver amplifier. The driver amplifiers are turned ON/OFF whenever the power amplifiers ar turned ON/OFF respectively.

3.7.2.3 Power Amplifier. The power amplifier amplifies the S-band output signal from the driver amplifier and provides a minimum of 9.5 watts output. This is accomplished by two high power, MSC3005, transistors in parallel. The RF input power is split into two paths by the same type of 3 dB hybrid that combines the output from the two MSC3005s, as shown by the block diagram in Figure 3.7.2.3-1. The two transistors are protected by the output isolator, so that it is possible to operate the amplifier into an open circuit or short circuit (due to spacecraft failure), although operationally this should always be avoided due to the additional stress encountered.

DC power for the two RF transistors is provided by a temperature compensated series regulator. Temperature compensation is necessary to limit the current consumption to the specified 1.2 amperes at the ambient and lower temperatures where the transistors put out more RF power but draw more current. This regulator also supplies the dc power to the driver amplifier. The control circuits for the amplifier are in the dc section or "dc frame". On/Off control is accomplished by means of a latching circuit that latches the regulator on with the "ON" pulse command and turns it off with the "OFF" pulse command or if the bus voltage drops too low, approximately seven volts. There is also current limiting, which is set at two amperes, and if the fault is such so as to cause more current to be drawn, the regulated voltage will decrease ("fold back") in order to maintain the two amperes

current limit. Furthermore, for a severe fault, the regulated voltage can fall below the seven volts threshold and the regulator will unlatch, that is turn off. The amplifier can be turned on again by the "ON" command but, if the fault still exists, it will only shut off again.

The power amplifier is fused internally with two (2) four ampere fuses and a 0.2 ohm resistor is in series externally in the harness with one fuse so that one fuse will always blow first.

- Power Amplifier Operational Characteristics

Output Power	9.5 watts minimum
Current Consumption	1.2 amperes (0.4 amperes additional for driver amplifier)
Current Limit	2 amperes
Minimum Bus Voltage to Remain "Latched" On	7 volts
Telemetry Information	Power Output Amplifier Temperature (Refer to Appendix A for details)

3.7.2.4 Hybrids, Diplexer, Isolator and Rotary Joint. Two types of hybrids are used in the Orbiter Communication Subsystem. The 6 dB hybrid is used to split the output power of the exciter four ways to drive the four driver amplifiers. The 3 dB hybrid is used as a power summer to combine the output of two power amplifiers. Both hybrids are passive devices and have no command or telemetry capability.

Diplexer. The diplexer permits simultaneous transmission and reception through the same antenna. The diplexer utilizes two coaxial (TEM) mode filters connected as shown in the schematic in Figures 3.7.2.4-1A and 3.7.2.4-1B; operation is explained by the block diagram in the same figures. Energy from the transmitter is prevented from entering the receiver by the bandpass filter at the receiver port that is

tuned to the receive center frequency. This filter has high attenuation at the transmit frequency and low attenuation at the receive frequency. Transmitter energy due to spurs that might fall into the receive band is attenuated by the band reject filter that is tuned to the receive frequency also. This filter has high attenuation at the receive frequency at the transmitter input port but looks like an open circuit at the antenna port at the receive frequency.

The response of the SN 5 diplexer, for a combination of each of the ports, is shown in Figure 3.7.2.4-2. Since the diplexer is a passive device, it has neither a command nor a telemetry capability.

Isolator. The isolator is a three-port ferrite circulator with a matched termination on the third port. The termination is capable of dissipating 12 watts. The signal is passed in the forward direction with very low insertion loss (less than 0.3 dB) and with very high insertion loss (greater than 25 dB) in the reverse direction. It is used between the driver amplifier and the power amplifier. The power amplifier is protected by its own bult-in isolator (same basic design, but 40 watt termination).

Rotary Joint. The rotary joint provides the S and X-band RF links between the stationary part (despun antenna) and the rotating part of the Orbiter. The dual-channel rotary joint consists of two concentric coaxial lines, the central line of which carries the X-band signal, and the outer line the S-band at minimal loss. Maximum life and minimum RF noise is ensured by the use of noncontacting choke joints to couple the RF energy across the interface between the stationary and moving parts. The only contacting parts are the bearing and power transfer assembly (BAPTA) ball bearings used to maintain a constant separation between the stationary and rotating portions of the rotary joint.

The rotary joint is contained within the BAPTA, which provides the mechanical rotating structure and mounting supports.

Waveguide-to-coaxial transitions are used to couple in and out of the rotary joint. The X-band transition uses a simple probe coupling technique. The S-band transition uses a modified shorted center conductor in a single ridge waveguide for both the input and output sections. The spin section transitions for both S and X-bands contain the noncontacting chokes.

3.7.2.5 Switches and Switch Driver. Two types of RF latching relay switches are used in the Bus/Orbiter Communications Subsystem, an S switch and a C switch. The S switch is a SPDT switch and the C switch is a cross-over switch in which two inputs are alternated between two outputs, as shown in Figure 3.7.2.5-1.

Two coils are used on each switch, one for each position, thus requiring two circuits to provide the switching current pulses. In the absence of switching pulses, the switches remain in the state corresponding to the last pulse. A minimum specified pulse voltage of 24 volts is required to cause switching.

Switch Driver. The switch driver consists of seven identical modules packaged in a single unit. Each module is basically two command pulse amplifiers which increase any one of the input command signals to the voltage and current levels required to operate the RF switch. The standard 35 msec, 9 to 17 Vdc command pulse at any input, controls a transistor switch that applies the unregulated Bus (essential bus) to the designated RF switch coil; pulsing a "2" command input provides a pulse to the "2" side of the RF switch, and a pulse at a "1" input provides a pulse to the "1" side of the switch. A bilevel telemetry output is provided to indicate commanded switch status. A "0" indicates that the last executed signal was position "1" or "normal" and a "1" indicates that the last executed signal was position "2" or "reverse".

When power is first applied, or removed and reapplied, the telemetry comes up in the "0" state, regardless of the actual switch position. The switch drivers are connected to the spacecraft essential bus.

3.7.2.6 X-Band Transmitter. The X-band transmitter outputs a carrier (without modulation) that is coherent with the S-band downlink frequency with the frequency ratio of 11/3. It receives its input drive signal from the exciter portion of either transponder. The input frequency ranges from 210.045 MHz to 211.035 MHz at a level of 0 dBm ±2 dB. The input frequency is multipled X40 by four separate diode multipliers and amplified such that output signal is 1100 mw minimum over the frequency range of 8401.8 MHz to 8441.4 MHz. A block diagram is shown in Figure 3.7.1-4. The low frequency assembly accepts either input drive signal without the use of a switch by using a 3 dB hybrid summer. A times two multiplier and a times four multiplier provide an output frequency of approximately 1689 MHz to the power amplifier assembly. This assembly provides the drive for the two parallel X5 multiplier assemblies. The two X5 outputs ar combined by a power summer that is followed by a coax to waveguide adapter that contains the power monitor telemetry pickup probe. The signal from the pickup probe is amplified by a dc amplifier contained within the dc regulator assembly. The dc regulator assembly provides the necessary voltages for the low frequency assembly and the power amplifier assembly and provides the ON/OFF command function. An overcurrent limiter within the dc regulator limits the input current to one ampere and turns off the unit when this limit is reached. Two 4-ampere fuses in parallel with a one resistor in series with one fuse protect the spacecraft power bus.

The flight X-band transmitter (S/N 002) power output for three test temperatures is as follows:

Temperature (°F)	Power Output (watts) 8401.8 MHz	8441.4 MHz
39	1.36	1.39
70	1.25	1.28
120	1.14	1.21

The X-band transmitter temperature (RXSTRT) is telemetered in addition to the power output (RXSTRW) telemetry.

3.7.2.7 Antennas. The Orbiter utilizes a forward omni and an aft omni antenna to obtain 4π steradian antenna coverage for command reception (either simultaneously), and telemetry transmission (one at a time). A despun high gain antenna (HGA) provides the primary telemetry link and a sleeve dipole antenna provides backup to the HGA should the despin mechanism fail. A brief description and a performance summary for these antennas is given below:

3.7.2.7.1 Atennas Description

High Gain Antenna (HGA) Description. The High Gain Antenna consists of a 43-inch diameter aluminum honeycomb parabolic reflector and a S/X-band dual frequency feed. Design frequencies of the feed are 2115 ±5, 2295 ±5 and 8415 ±5 MHz. The HGA assembly is mounted on the despun section of the Pioneer Venus Orbiter spacecraft.

The reflector and the focal point feed are mounted independently on the mast assembly to permit tilting of the reflector of approximately 10° in elevation for scanning of the 2295 and 8415 MHz beams electrically greater than ±15° for the scientific occultation experiment.

The dual frequency feed is a coaxial horn operating at S-band (2115 to 2300 MHz) where the hollow center conductor is used as an X-band conical feed horn. The S-band feed is excited by four symmetric probes which are fed at equal amplitude and with a 90° phase progression to obtain circularly polarized radiation. The feed network for the four probes is a one to four

power dividing network made up from three quadrature branchline hybrids.

The X-band feed consists of the hollow center conductor of the coaxial horn excited as a circular horn, a two pin polarizer and a right angle rectangular-to-circular waveguide transition.

High Gain Backup Antenna (HGBUA) Description. The High Gain Backup Antenna is a six element center fed sleeve dipole array, that mounts on top of the high gain antenna mast on the Orbiter spacecraft. The antenna operates at both transmit and receive frequencies of 2295 ±5 and 2115 ±5 MHz, respectively. It provides a medium gain azimuthal omnidirectional pattern to be used as backup to the high gain antenna in case the despun mechanism malfunctions.

The sleeve dipole array is a center fed standing wave array consisting of several concentric coaxial transmission lines, such that the outer conductor of one line serves as the center conductor of the next level of concentric coaxial line. The central coaxial line is the feed line to the forward omni which is mounted on top of the backup antenna. This cable is electrically independent of the backup antenna. It is a straight through feed line with connectors on both ends. The second concentric coaxial line is the input line to the backup antenna. Its input is a coaxial tee junction and the line couples to the third level of concentric line through an annular ring. The third and final concentric line couples directly to the sleeve dipoles.

This third coaxial line is dielectrically loaded to reduce the phase velocity in the line. This and center feeding together make the antenna radiate a broadside beam with minimal beam squint at the transmit and receive frequencies. The dielectric loading of the outer coax line and a fiberglass sleeve over the whole assembly add strength along the length of the sleeve dipole array.

Forward Omni Antenna Description. The forward omni antenna is a two arm log conical spiral. Identical units are mounted on top of masts located on the forward side of both the Multiprobe Bus and and Orbiter spacecrafts to provide RF communication coverage over angles from 0° to approximately 95° from the plus Z axis.

The antenna consists of an epoxy glass conical shell on which two copper conductors are desposited. The radiating elements are fed at the top of the cone by a microstrip balun which is supported in a fiberglass tube. A coax to microstrip transition is located at the base of the antenna. This design radiates a right hand circularly polarized axial mode pattern.

Aft Omni Antenna Description. The aft omni antenna is a common turnstile element above a conical ground plane. Identical units are mounted on top of masts located on the aft end of both the Multiprobe Bus and Orbiter spacecrafts to provide RF communication coverage over angles from 95° to 180° from the plus Z-axis.

The turnstile element is used to produce circular polarization and typically produces a relatively narrow 3 dB beamwidth (70°) pattern. The conical ground plane is incorporated to broaden the beamwidth and generate the nearly hemispherical gain coverage.

The antenna is all aluminum welded construction except for a small teflon dielectric bead in the input connector.

The design is capable of surviving the extreme temperature (500°F) resulting from its proximity to the motor plume on the Orbiter. The input coaxial connector (type TNC) and semi-rigid feed cable are protected inside the mast on which the antenna is mounted.

3.7.2.7.2 Antennas Patterns

Omni Antennas. Plots of the minimum gain for the forward and aft omni antennas, in terms of radiated power density (EIRP), are given in Figure 3.7.2.7-1. The difference between these plots and actual antenna gain is that the transmitter power and circuit loss to each antenna has been included; however, the shape of the curves is identical to the actual antenna gain. Such a presentation has been judged to be more useful due to the desirability of indicating the spherical antenna coverage of the spacecraft on one set of coordinate axes and since each antenna has different feed circuit loss. It is important to note that Figure 3.7.2.7-1 includes the minimum gain of the antennas at each elevation angle as a function of azimuth. That is, the plots are obtained by making an antenna measurement while holding elevation angle constant and varying the azimuth angle. The azimuth angle, for which the gain is minimum, is then used to derive Figure 3.7.2.7-1. The variation between minimum and maximum gain is indicated by the conic plots, or gain versus azimuth angle, given in Figure 3.7.2.7-2A and 3.7.2.7-2B.

The minimum receive gain for the omni antennas is shown in terms of flux density in Figure 3.7.2.7-3. The command link threshold, is based on the specified -103.2 dBm per square meter maximum uplink power density, and a bit error rate (BER) of 3.5×10^{-6}. This BER has been shown to provide the specified probability of false command execution and missed command, and corresponds to a signal to noise ratio of 17.2 dB at the transponder output. It can be seen from Figure 3.7.2.7-3 that for communication angles greater than approximately 100°, the aft omni is more effective for reception.

High Gain Antenna. A summary of the high gain antenna measured gain and beamwidth is given in Table 3.7.2-2. The azimuth and elevation patterns for the HGA for 0° dish tilt angle for 2295 GHz are given in Figures 3.7.2.7-4 and

3.7.2.7-5, respectively. The azimuth and elevation patterns for 8.415 GHz, 0° scan, are given in Figures 3.7.2.7-6 and 3.7.2.7-7, respectively. The on axis axial ratios for the HGA are 1.8 dB at 2.15 GHz, 1.9 dB at 2.95 GHz, and 2.1 dB at 8.415 GHz. These axial ratios were included in the gain calculation with respect to a perfect right hand circularly polarized source. It should be noted that since the feed is fixed and the dish is rotated for beam scanning, the X and S beam centers diverge from each other as the dish is tilted. The beam divergence and the beam steering as a function of dish tilt angle is given in Table 3.7.2-3, with definition of the HGA Pattern Coordinates shown in Figure 3.7.2.7-8. The effect of the magnetometer boom on the HGA Gain pattern is seen in Figure 3.7.2.7-9.

High Gain Backup Antenna. A summary of the high gain backup antenna performance is given in Table 3.7.2-4. The elevation pattern for the HGBUA is given in Figures 3.7.2.7-10 and 3.7.2.7-11 and the conic or azimuth pattern for 90° (+Z axis) is given in Figures 3.7.2.7-12 and 3.7.2.7-13. The HGBUA peak gain is found by taking the minimum value of the conic pattern.

3.7.2.8 Engineering Instrumentation - (See Appendix A)

3.7.3 Operations Description

3.7.3.1 Normal Operating Modes

3.7.3.1.1 Transmission Modes. The Orbiter communication subsystem has a number of operating modes which are defined according to the transmitter configuration, the antenna used, and the receiver used. The two basic transmitter modes are high power and low power, but since there are redundant power amplifiers, there are several combinations of amplifiers available for each mode. Furthermore, there are two basic routes that the RF power can take, either through the rotary joint to a despun antenna or to the aft omni. Table 3.7.3-1 lists the modes which result from these various combinations. The switch positions, and associated commands and telemetry

TABLE 3.7.2-2

HGA PERFORMANCE SUMMARY

Frequency, gHz	Peak Antenna Gain (Note 2) In dB Vs. Tilt Angle (Note 1)			S-Band Antenna Gain (Note 4) @ ±4.15° Beamwidth Vs. Tilt Angle (Note 1)		
	Maximum Tilt Angle (107.1°)	Tilt Angle = 90°	Minimum Tilt Angle (69.5°)	Maximum Tilt Angle (107.1°)	Tilt Angle = 90°	Minimum Tilt Angle (69.5°)
2.295	24.8	25.2	25.5	20.6	21.2	21.1
2.115	23.5	24.4	23.4	19.9	21.2	21.2
8.415 (Note 3)	33.8	35.0	34.5	-	-	-

Note 1: Tilt Angle is that angle between the RF Beam Boresight and the S/C +Z Axis.
Note 2: Peak Antenna Gain is measured on the Beam Boresight.
Note 3: Measured X-Band 3 dB Beamwidth is ±1.1 degrees.
Note 4: 4.15° Beamwidth is required by PC 410.

TABLE 3.7.2-3
HGA BEAM STEERING SUMMARY & BEAM COINCIDENCE

Dish Tilt*	S-Band MEASURED Beam Direction		X-Band MEASURED Beam Direction		Difference In S- and X-Bands Beam Directions	
Angle (Deg)	Azimuth	Elev.	Azimuth	Elev.	Azimuth	Elev.
0	0.40	-0.45	0.40	-0.49	0	+0.04
+2.0	0.38	+3.65	0.40	+3.74	-0.02	-0.09
+4.0	0.35	+7.57	0.40	+7.65	-0.05	-0.08
+6.0	0.36	+11.46	0.40	+11.59	-0.04	-0.13
+8.0	0.20	+15.36	0.40	+15.51	-0.20	-0.15
-2.0	0.50	-4.18	0.40	-4.23	+0.10	+0.05
-4.0	0.50	-8.15	0.42	-8.18	+0.08	+0.03
-6.0	0.50	-11.24	0.45	-11.26	+0.05	+0.02
-8.0	0.50	-15.74	0.45	-15.86	+0.05	+0.12
+max	0.20	+16.85	0.38	+17.11	-0.18	-0.26
-max	0.70	-20.27	0.40	-20.48	+0.30	+0.21

* $\pm$0.2 degrees accuracy.
**See Figure 3.7.2.7-8 for coordinates diagram.

TABLE 3.7.2-4
HIGH GAIN BACKUP ANTENNA PERFORMANCE SUMMARY

Frequency (gHz)	Angle (Degrees) W.R.T. S/C +Z Axis	Minimum Gain (dB)
2295 Downlink	90°	7.8
	85° to 95°	4.2
2115 Uplink	90°	1.65
	85° to 95°	-2.15

plus the power amplifier commands and telemetry for the twelve transmitter modes, are given in Table 3.7.3-1. Twenty-four spacecraft RF transmission modes result from the 12 transmitter modes being used in conjunction with the four spacecraft antenna, as given in Table 3.7.3-2 (only odd number transmitter modes are used with the despun antennas and even number transmitter modes are used with the aft omni). Another twenty-four spacecraft transmission modes (total of 48) exist as a result of using transponder number two exciter (number one exciter is used for the first 24 modes). Switch posiitons, commands, and associated telemetry for the forty-eight spacecraft transmission modes are given in Table 3.7.3-2. The telemetry indications given in Tables 3.7.3-1 and 3.7.3-2 provide a convenient means for identifying the spacecraft transmission modes. The spacecraft EIRP for the first 24 modes (identical for second 24) is given in Table 3.7.3-2.

The communication subsystem has the capability of being commanded to either a high modulation index (1.18 radians) or a low modulation index (0.65 radians) for telemetry subcarrier modulation of the downlink carrier. In general, the high modulation index is utilized for the higher data rates with the high gain antenna and the low modulation index is used for the low data rates when the spacecraft omni antennas are being used for transmission, such as during orbit insertion. Downlink S-band Carrier-only operation is possible by commanding the subcarrier OFF via the Telemetry Processor Control command (TPCQ1 for TP1; TPCQA for TP2).

The X-band transmitter operation is independent of the spacecraft transmission mode except for some operating time limitations imposed by transmitter modes 5, 6, 9 and 10. These limitations are discussed further in Section 3.7.3.4. The X-band transmitter receives its RF drive at 210.45 MHz (from Exciter 1) or 210.35 MHz (from Exciter 2), according to whichever exciter is turned on. To transmit an X-band signal, it is only necessary to turn on an

TABLE 3.7.3-1 ORBITER S-BAND TRANSMITTER MODES

Mode No.	Power Mode	Antenna Used	Power Amp(s) Used	Power Amp Commands	Switch Positions			Switch Commands			Telemetry Indications				
					3SW019 Position	3SW022 Position	2SW019 Position	2SW019	3SW022	2SW019	Power Amp		Switch Drivers (Bit State)		
											Power Out.	Temp	3SW019 RAMP1S	3SW022 RAMP3S	2SW019 RANT1S
1 (1)	Low Pwr 1	Despun Antenna	#1 ON, Rest OFF	AMP19/AMPA9 AMP39/AMPC9	2	NA	2	AMP11/ AMPA1	NA	ANT11/ ANTA1	RAMP1W	RAMP1T	1	NA	1 (1)
2	Low Pwr 1	Aft Omni	#1 ON, Rest OFF	AMP19/AMPA9 AMP39/AMPC9	2	NA	1	AMP11/ AMPA1	NA	ANT12/ ANTA2	RAMP1W	RAMP1T	1	NA	0
3	Low Pwr 2	Despun Antenna	#2 ON, Rest OFF	AMP29/AMPB9 AMP39/AMPC9	1	NA	2	AMP12/ AMPA2	NA	ANT11/ ANTA1	RAMP2W	RAMP2T	0	NA	1
4	Low Pwr 2	Aft Omni	#2 ON, Rest OFF	AMP29/AMPB9 AMP39/AMPC9	1	NA	1	AMP12/ AMPA2	NA	ANT12/ ANTA2	RAMP2W	RAMP2T	0	NA	0
5	High Pwr 1&3	Despun Antenna	1 & 3 ON, Rest OFF	AMP19/AMPA9 AMP39/AMPC9	1	2	1	AMP12/ AMPA2	AMP31/ AMPC1	ANT12/ ANTA2	RAMP1W RAMP3W	RAMP1T RAMP3T	0	1	0
6	High Pwr 1&3	Aft Omni	1&3 ON, Rest OFF	AMP19/AMPA9 AMP39/AMPC9	1	2	2	AMP12/ AMPA2	AMP31/ AMPC1	ANT11/ ANTA1	RAMP1W RAMP3W	RAMP1T RAMP3T	0	1	1
7 (2)	High Pwr 1&4	Despun Antenna	1&4 ON, Rest OFF	AMP19/AMPA9 AMP49/AMPD9	1	1	1	AMP12/ AMPA2	AMP41/ AMPD1	ANT12/ ANTA2	RAMP1W RAMP4W	RAMP1T RAMP4T	0	0	0
8 (2)	High Pwr 1&4	Aft Omni	1&4 ON, Rest OFF	AMP19/AMPA9 AMP49/AMPD9	1	1	2	AMP12/ AMPA2	AMP41/ AMPD1	ANT11/ ANTA1	RAMP1W RAMP4W	RAMP1T RAMP4T	0	0	1
9	High Pwr 2&3	Despun Antenna	2&3 ON, Rest OFF	AMP29/AMPB9 AMP39/AMPC9	2	2	1	AMP11/ AMPA1	AMP31/ AMPC1	ANT12/ ANTA2	RAMP2W RAMP3W	RAMP2T RAMP3T	1	1	0
10	High Pwr 2&3	Aft Omni	2&3 ON, Rest OFF	AMP29/AMPB9 AMP39/AMPC9	2	2	2	AMP11/ AMPA1	AMP31/ AMPC1	ANT11/ ANTA2	RAMP2W RAMP3W	RAMP2T RAMP3T	1	1	1
11 (2)	High Pwr 2&4	Despun Antenna	2 & 4 ON, Rest OFF	AMP29/AMPB9 AMP49/AMPD9	2	1	1	AMP11/ AMPA1	AMP41/ AMPD1	ANT12/ ANTA2	RAMP2W RAMP4W	RAMP2T RAMP4T	1	0	0
12 (2)	High Pwr 2&4	Aft Omni	2 & 4 ON, Rest OFF	AMP29/AMPB9 AMP49/AMPD9	2	1	2	AMP11/ AMPA1	AMP41/ AMPD1	ANT11/ ANTA1	RAMP2W RAMP4W	RAMP2T RAMP4T	1	0	1

NOTES: (1) Launch Mode: Despun Section is spinning and forward omni is in use.
(2) Limits X-Band Operation Time due to thermal restriction.

TABLE 3.7.3-2 ORBITER SPACECRAFT RF TRANSMISSION MODES

Transmission Mode No.	Antenna Used:	Transmitter Mode:	Exciter Used:	Switch Positions: 1SW022	2SW022	4SW022	Switch Commands: 1SW022	2SW022	4SW022	Switch Drivers Telemetry: (Bit State) 1SW022 RANT2S	2SW022 RXCTR5	4SW022 RANT3S	S-Band Worst Case EIRP: (dBm)	Comments
1	Aft Omni	2		NA		NA	NA		NA	NA		NA	38.9	@ 110° from +Z Axis.
2		4											38.9	
3		6											41.4	
4		8												
5		10												
6		12												
7	Fwd Omni	1		1		1	ANT22/ ANTB2		ANT32/ ANTC2	0		0	35.5	@ 90° from +Z Axis.
8		3											35.3	
9		5											37.9	
10		7												
11		9												
12		11												
13	HGA	1	1	2	2	NA	ANT21/ ANTB1	EXC11/ EXCA1	NA	1	1	NA	63.5	Peak of Beam, @ 90° Scan Angle from +Z axis.
14		3											63.5	
15		5											66.1	
16		7												
17		9												
18		11												
19	HGBUA	1		1		2	ANT22/ ANTB2		ANT31/ ANTC1	0		1	42.7	Peak of Beam, @ 90° from +Z Axis.
20		3											42.7	
21		5											45.2	
22		7												
23		9												
24		11												
25 to 48	Same combinations as above.	Same combinations as above.	2	Same combinations as above.	1	Same combinations as above.	Same combinations as above.	EXC21/ EXCB1	Same combinations as above.	Same combination as above.	0	Same combinations as above.	Same combinations as above.	

exciter, if one is not already on (commands EXC19/EXCA9 or EXC29/EXCB9) and to command the X-band transmitter on (Command XBT19/XBTA9). No antenna switching is necessary, since the X-band transmitter output permanently feeds the high gain antenna X-band feed (the spacecraft must be despun, with the HGA pointed at Earth, if a signal is to be received on Earth). The X-band signal will be radiated regardless of reception of an uplink S-band signal, since the exciter auxiliary oscillator is automatically the source of X-band drive as well as S-band drive in the absence of uplink lock. However, in order to achieve an X-band signal with the least phase jitter, the transponder should be operated coherently, i.e., locked to an uplink signal, through the high gain antenna.

Telemetry will indicate a non-zero RF output power for the X-band transmitter only when an exciter is turned ON to provide RF input to the transmitter. Approximately 0.5 amp will be drawn once the X-band transmitter has RF drive. A similar situation occurs for the S-band Power Amplifiers (Reference: Paragraph 1.5.17).

Figures 3.7.3-1 through 3.7.3-3 show the nominal spacecraft Data Transmission capability with the three antennas for all or part of the Mission. These were extracted from Reference: Paragraph 1.5.32. Figures 3.7.3-4 through 3.7.3-9 show the expected received signal strength for the indicated conditions.

3.7.3.1.2 Reception Modes. Since the two transponder receivers are always powered "ON", their use for reception of the uplink signal is controlled by selecting the uplink frequency to correspond to the nominal rest frequency of the desired transponder (transponder 1:2112.289352 MHz and transponder 2:2112.630401 MHz). It is also necessary to take into consideration which antenna is connected to the desired receiver and the communication angle as determined by the spacecraft attitude. In general, whenever the orbiter is oriented perpendicular to the ecliptic plane with the HGA despun and pointed at the

Earth, the HGA will be selected for both transmission and reception. If the spacecraft is perpendicular to the ecliptic plane and spun up, the HGBUA may be used if the orbiter's displacement from the ecliptic plane (as given in Section 4.3.4) is less than approximately 8 degrees (the exact communication angle supportable by the HGBUA is dependent on communication range and DSN station). The aft omni antenna is always connected to a receiver, as shown in Table 3.7.3-3, but the choice between a despun antenna and the aft omni for reception is dependent on communication angle (as dictated by spacecraft orientation) and the desired transmission mode, as given in Table 3.7.3-2.

The receiver reverse command eliminates the possibility of losing reception capability due to several possible spacecraft failures. These possible failures are receiver failure, unwanted spin up of despun section, and other failures which cause uplink communications to be lost. In the event of one of these failures occurring, the on-board command processor automatically issues a receiver reverse command after a period of 36 hours has occurred without processing any spacecraft command. The command is sent to coax switch 1SW022 and causes the HGBUA and forward omni feed line to be connected to the rotary joint signal path. Simultaneously, the command is also sent to coax switch 4SW022 so that the forward omni is connected to the feed line selected by 1SW022 and thus, through the rotary joint signal path. At the same time, the antenna input connections to both transponder receivers are reversed by actuation of coax switch 1SW019. When the receiver reverse command is executed by the spacecraft, the uplink frequency must be shifted to that of the transponder which will be connected to the forward omni. Also, the downlink will be connected automatically to the forward omni, and the new downlink frequency will become 240/221 times the new uplink frequency after onboard phase-lock is established.

The receiver reverse command may also be actuated by ground command, as discussed in Section 3.6.

TABLE 3.7.3-3

RECEIVER/ANTENNA COMBINATIONS

RCVR Mode	Switch Position For 1SW019:	Telemetry Indicator (RRCVRS Bit State)	Antenna Used By RCVR 1 / S/C Xmission Mode (From Table 3.7.3-2)	Antenna Used By RCVR 2 / S/C Xmission Mode (From Table 3.7.3-2)
Reverse	Position 1 (Switch "Normal" State)	0	HGA/13 Through 18 OR: HGBUA/19 Through 24 OR: FWD Omni/7 Through 12	Aft Omni/Any
Normal	Position 2 (Switch "Reverse" State)	1	Aft Omni/Any	HGA/13 Through 18 OR: HGBUA/19 Through 24 OR: FWD Omni/7 Through 12

NOTE: See Table 3.6.4-1, Section 3.6 for explanation of "CP Configure" command that is used to change the receiver modes.

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However, when actuated by ground command, only switch 1SW019 is actuated.

3.7.3.1.3 Exciter and Coherent Mode Usage. Either Exciter 1 or Exciter 2 can be used with either receiver. Automatic cross strapping of the coherent frequency source from the receiver that is in lock is provided, as described in 3.7.2.1.2. Thus, although Exciter 1 (also Receiver 1) is designated as the primary exciter, Exciter 2 can be used as well. However, after some in-flight operational history of the spacecraft has been accumulated, one of the exciters may appear more desirable from the standpoint of frequency stability of the auxiliary oscillator when in the non-coherent mode.

Coherent Mode Versus Non-Coherent Mode. The normal operation of the spacecraft will be with simultaneous uplink and downlink, thus providing for coherent operation of the communication links. As long as the last coherent mode command executed by the spacecraft was "Restore Coherent Mode" (COH19 or COHA9), then selection of the auxiliary oscillator automatically occurs with the absence of an uplink, and the coherent mode is automatically selected when uplink lock reoccurs. The "Inhibit Coherent Mode" command (COH1Ø or COHAØ) prevents this automatic transference in either direction, and causes the auxiliary oscillator to be the frequency source whether or not an uplink is present.

3.7.3.1.4 Communication Subsystem Initialization. Upon application of DC power to the communication subsystem, the units will be in the following states:

Unit	"Initial DC Power ON" State:	Comments
Transponder Receiver	NA	Receiver is on the spacecraft essential bus and is therefore always On
Exciter	ON/OFF TM - "OFF"	
	Transfer to VCO TM - "OFF"	Test only - wired out for flight
	Inhibit/Restore Coherent - Mode TM - "Restore"	
	Mod Index (NO TM) - High	
Power Amplifier	ON/OFF TM - "OFF"	
Driver Amplifier	Same as Power Amplifier	
X-Band Transmitter	ON/OFF TM - "OFF"	
RF Coax Switch	NA	Magnetic latching; will be in state of last command that was sent.

Unit	"Initial DC Power ON" State:	Comments
Switch Driver	NA	Each switch driver is on the essential bus and is therefore always on. However: •For initial spacecraft power up, the switch driver TLM will initialize regardless of switch position, in the logic "0" state (= position 1). •The switches must be commanded to their desired position in order to force the status bits to indicate actual switch position. •The fact that a switch command changed the appropriate status bit to indicate the switch has moved to the proper position does not guarantee that the switch actually moved. It only indicates that the desired command pulse was generated.

3.7.3.1.5 On-Orbit Use of Communication Subsystem Equipment. Prior to a long eclipse, the Orbiter communication subsystem should be configured for use of the forward omni antenna and the 10-watt transmitter that has provided the most satisfactory in-flight performance. Also, the receiver that has provided the most satisfactory in-flight performance should be connected to the forward omni. During the eclipse, the exciter and all transmitters will be turned off. Upon exit from the eclipse, the exciter and power amplifier selected according to the criteria stated above should be commanded on. Once satisfactory spacecraft control has been established, switching to the high gain antenna can be performed.

Prior to going into occultation, the spacecraft will be configured for the 20-watt transmitter mode, according to the constraint discussed in 3.7.3.4, with the X-band transmitter also turned on. Optionally, the spacecraft TM subcarrier may be turned OFF. The receiver which is used for occultation should be the one which has exhibited the quickest acquisiton in flight and the exciter used should be the one which has exhibited the best auxiliary oscillator stability in flight. Prior to beginning occultation, the uplink frequency should be tuned as close as possible to correspond to the receiver best lock frequency at the spacecraft. This will minimize the spacecraft receiver VCO drift when loss of uplink lock occurs and thus aid in faster uplink acquisition when the spacecraft emerges from occultation.

3.7.3.2 Use of Redundancy. The Orbiter Communication Subsystem contains four (4) power amplifiers which offer a redundancy of high and low power modes, as discussed in 3.7.3.1. Two transponders offer uplink reception redundancy, as discussed in 3.7.3.1, also. The HGBUA offers redundancy for the HGA, in the event of HGA spinup, since the HGBUA has a pattern which is symmetrical in azimuth about the Z axis only. However, since the HGBUA has approximately 17 dB less gain than the HGA, this redundancy is at the expense of degraded link performance; i.e., a lower data rate. Since the X-Band transmitter is strictly a radio science experiment and is not used for communicating telemetry, it has no redundancy.

3.7.3.3 Non-Standard Modes. The non-standard modes for the Orbiter Communication Subsystem are as follows:

(a) Only 1 power amplifier "ON" that is connected to the 3 dB hybrid (power summer): It is possible to configure 3SW019 or 3SW022 so that either amplifier 1, 2, 3 or 4 only is "ON" and connected to the input of the 3 dB hybrid. This will result in a power output from the hybrid of approximately

5 watts, with 5 watts being dissipated in the integral hybrid load. In this mode, the hybrid mounting surface temperature will reach the acceptance test level of 136°F, but thermal analysis indicates that the internal temperature may reach 165°F. Although the hybrid will probably survive, this mode is not recommended as a normal operating mode.

(b) Simultaneous transmission through a despun antenna and the aft omni antenna: The communication subsystem may be configured so that the downlink telemetry signal is transmitted by a forward antenna and the aft omni. There is no readily apparent advantage to this mode and it may create an erratic and poorly shaped radiation pattern due to the interferometric effect of two nearby antennas radiating the same signal. Furthermore, this non-standard mode necessitates the use of non-standard mode 1 discussed above, since, by design, one antenna would be fed by a power amplifier (1 or 2) that bypasses the 3 dB hybrid, and the other antenna would be fed by a power amplifier (3 or 4) via the 3 dB hybrid.

3.7.3.4 Operational Restrictions. The following restrictions apply to the operation of the Orbiter Communication Subsystem:

(a) Do not switch "Hot", i.e., do not change the position of an RF switch while an RF transmitter signal is passing through it, except in an emergency situation (Reference: Paragraph 1.5.18). The power amplifiers should always be commanded "OFF" before actuating the RF switches that feed RF into them, or pass RF from them. This is to protect both the switches and the power amplifiers. Inadvertent failure to observe this restriction is not likely to damage

these units, but repeated violations may eventually inflict damage.

The following sequence of operations should be carried out if switching of RF switches is to be performed:

(1) Turn off units which provide input RF power to the switch (if such units are commandable).
(2) Turn off units which obtain RF power from the switch (if such units are commandable).
(3) Initiate switch command.
(4) Turn on units turned off in (2).
(5) Turn on units turned off in (1).
(6) Verify switch is in proper configuration by telemetry and/or spectrum analysis.

In an emergency situation, where an RF link may be lost unless switching is quickly performed, hot switching is allowed. Steps (3) and (6) should then be executed in that order.

(b) Never operate a power amplifier into an open circuit. This restriction applies to amplifiers 3 or 4. Caution must be used to ensure that switch 3SW022 is configured so that the 3 dB hybrid is connected to whichever amplifier is turned "ON" or about to be commanded "ON". Refer to Table 3.7.3-1 for the correct switch position and command.

(c) X-Band Transmitter Operating Time: When the S-band transmitter is operated in the 20-watt mode, the X-band transmitter will reach its acceptance temperature limit of 120°F (mounting surface temperature) in one hour. If amplifier 3 is combined with either amplifier 1 or 2 to produce the 20 watt mode, a steady state temperature of 140°F will be reached after approximately 3 hours of X-band transmitter operation. Furthermore, the power amplifier closest to the X-band transmitter, PA4, must not be on or the transmitter will reach its

acceptance temperature limit in less than one hour.

These time estimates apply for the nominal spacecraft attitude on-orbit, i.e., the spacecraft spin axis is normal to the ecliptic.

(d) The spacecraft Aft omni should not be used at launch, as the geometrical interference from the attached Centaur is significant.

(e) The 3 dB hybrid is used as a power summer to combine the output of two power amplifiers. Should only one of the two power amplifiers, that supply an input signal to the 3 dB hybrid, be "ON", half of the signal power will be dissipated in the integral hybrid load and the other half, approximately five watts, will be output signal from the hybrid. Operation of the hybrid in this manner will cause the hybrid to be hotter than normal. Although the temperature will be within the hybrid design limitations, this mode is not recommended for normal operation.

(f) The best lock frequency of the P-V receivers will "push" if the acquisition sweep rate is low. That is, the spacecraft receiver VCO will move away from the uplink signal. Motorola has stated the maximum rate the VCO will "push" is 5.9 Hz/sec and have recommended that acquisition sweep rates be greater than 10 Hz/sec.

During the weak signal compatibility testing at CTA-21, an acquisition sweep rate of 10 Hz/sec caused no pushing while a 5 Hz/sec rate definitely did. It is recommended a minimum acquisition rate of 15 Hz/sec be used to provide a healthy margin of safety.

3.7.4 Communications Subsystem Command Response. Table 3.7.4-1 presents the command responses for the Communications Subsystem. The table lists every command that directly affects the subsystem, and the telemetry indication that verifies the proper execution of the command. The mnemonics for the command and telemetry parameters are also included.

When a unit is in the unpowered OFF state, all telemetry from that unit will be reading zero (uncalibrated), except temperature telemetry which is powered by DIMs.

TABLE 3.7.4-1

ORBITER COMMUNICATIONS SUBSYSTEM COMMAND RESPONSE

COMMAND	COMMAND TITLE	COMMAND DESCRIPTION AND SUBSYSTEM INTERNAL RESPONSE	TM MNEMONIC	TM TITLE AND/OR REMARKS
POWER AMPLIFIERS				Assumes PA2 was previously ON:
AMP19 AMPA9	Power Amplifier 1 ON/2 OFF	Actuates solid state switches within PA1 and PA2 so that PA2 is turned OFF and PA1 is turned ON.	RAMP1W	RF Power Output 1 = 7 to 10 Watts (@S/C Ambient)
			RAMP2W	RF Power Output 2 = 1 Watt (@ S/C Ambient)
			RAMP1T	PA1 Temperature rises and settles at (80°to 104°F)(@S/C Amb.)
			RAMP2T	PA2 Temperature Decreases & settles at the range of 77°to 95°F)
			PBUSLI	S/C Loads Current increases by approximately 1.5 amp.
AMP29 AMPB9	Power Amplifier 2 ON/1 OFF	Actuates solid state switches within PA1 and PA2 so that PA2 is turned ON and PA1 is turned OFF.		Assumes PA1 was previously ON:
			RAMP1W	RF Power Output 1 = 1 Watt (@ S/C Ambient)
			RAMP2W	RF Power Output 2 = 13 to 16 Watts (@ S/C Ambient)
			RAMP1T	PA1 Temperature Decreases & Settles at the range of (80°to 104°F)
			RAMP2T	PA2 Temperature Rises and Settles at the range of (82°to 108°F)
			PBUSLI	S/C Loads Current increases by approximately 1.5 amp.
AMP1B AMPA8	Power Amplifiers 1 and 2OFF	Actuates solid state switches within PA1 and PA2 so that PA1 and PA2 are turned OFF.		See Note #1:
			RAMP1W	RF Power Output 1 = 1 Watt (@ S/C Ambient)
			RAMP2W	RF Power Output 2 = 1 Watt(@ S/C Ambient)
			RAMP1T/ RAMP2T	According to which of PA1 or PA2 was previously ON, PA1 temp. or PA2 temp. decreases and settles at range of (80°to 104°F)
			PBUSLI	S/C Loads current decreases by approximately 1.5 amp.
			PLIMTI	Bus voltage limiter current likely increases by the same amount as PBUSLI had decreased.
AMP39 AMPC9	Power Amplifier 3 ON/4 OFF	Actuates solid state switches within PA3 and PA4 so that PA4 is turned OFF and PA3 is turned ON.	RAMP3W	Assumes PA3 and PA4 were previously OFF:
				RF Power Output 3 = 11 to 13 Watts (@ S/C Ambient)
			RAMP4W	RF Power Output 4 = 1 Watt
			RAMP3T	PA3 Temperature rises and settles at 78° to 88°F.
			PBUSLI	S/C Loads current increases by approximately 1.5 amp.
			PLIMTI	Bus voltage limiter current likely decreases by the same amount as PBUSLI has increased.
AMP49 AMPD9	Power Amplifier 4 ON/3 OFF	Actuates solid state switches within PA3 and PA4 so that PA4 is turned ON and PA3 is turned OFF.	RAMP3W	ASSUMES PA3 AND PA4 WERE PREVIOUSLY OFF:
				RF Power Output 3 = 1 Watt
			RAMP4W	RF Power Output 4 = 11 to 13 Watts.
			RAMP4T	PA4 Temperature rises & settles at 78° to 94°F (At S/C Ambient)
			PBUSLI	S/C Loads current increases by approximately 1.5 amp.
			PLIMTI	Bus voltage limiter current likely decreases by the same amount as PBUSLI had increased.

NOTE #1: The following telemetry will not be monitorable when the command is executed unless PA3 OR PA4 is in use for the downlink (this would result in non-standard mode #1 as described in Section 3.7.3.3).

TABLE 3.7.4-1 (cont'd)

ORBITER COMMUNICATIONS SUBSYSTEM COMMAND RESPONSE

COMMAND	COMMAND TITLE	COMMAND DESCRIPTION AND SUBSYSTEM INTERNAL RESPONSE	TM MNEMONIC	TM TITLE AND/OR REMARKS
POWER AMPLIFIERS (cont'd)				
AMP3Ø AMPCØ	Power Amplifiers 3 & 4 OFF	Actuates solid state switches within PA3 and PA4 so that PA3 and PA4 are turned OFF.	RAMP3W	RF Power Output 3 = 1 Watt @ S/C Ambient)
			RAMP4W	RF Power Output 4 = 1 Watt @ S/C Ambient)
			RAMP3T/ RAMP4T	According to which of PA3 or PA4 was previously on, PA3 temp. or PA4 temp. decreases & settles at the range of 69°to 88°F.
			PBUSLI	S/C Loads current decreases by approximately 1.5 amps.
			PLIMTI	Bus voltage limiter current likely increases by the same amount as PBUSLI had decreased.
SWITCH DRIVERS				SEE NOTE #2
ANT11 ANTA1	Low Power to Despun Antennas/High Power to Aft Omni Select	Actuates coax switch 2SW019 so that output of the 3 dB hybrid is connected to the aft omni antenna and the output of either PA1 or PA2 is connected to the rotary joint.	RANT1S	Aft Omni/Despun Antennas to High/Low Power Status = Bilevel logic "1" (= Aft to Hi).
ANT12 ANTA2	High Power to Despun Antennas/Low Power to Aft Omni Select	Actuates coax switch 2SW019 so that output of the 3 dB hybrid is connected to the rotary joint and the output of either PA1 or PA2 is connected to the aft omni antenna.	RANT1S	SEE NOTE #2 Aft Omni/Despun Antennas to High/Low Power Status = Bilevel logic "0" (= Aft to Low).
ANT21 ANTB1	High Gain Antenna Select	Actuates mast mounted coax switch 1SW022 so that high gain antenna is connected to the rotary joint. HGBUA and forward omni not usable.	RANT2S	SEE NOTE #3. High Gain Antenna/Low Gain Antennas Switch Position = Bilevel logic "1"(= HGA). HGA should be despun and pointed at Earth. If HGBUA was previously being used for transmission, downlink signal strength increases by approximately 8 dB. If omni was used, signal strength increases by approximately 25 dB.
ANT22 ANTB2	Backup/Forward Omni Antenna Select	Actuates mast mounted coax switch 1SW022 so that high gain antenna is disconnected from the rotary joint and either HGBUA or forward omni is connected.	RANT2S	SEE NOTE #3. Converse of above, i.e., Logic "0" (= HGBUA/Fwd Omni).
ANT31 ANTC1	High Gain Backup Antenna Select	Actuates mast mounted coax switch 4SW022 so that HGBUA is connected to switch 1SW022. If 1SW022 (above) is in ANT22/ANTB2 command state, HGBUA is connected to rotary joint.	RANT3S	High Gain Backup/Forward Omni Switch Position = Bilevel logic "1" (= HGBUA). An increase in downlink signal strength of approximately 7 dB will occur when switching from forward omni to HGBUA.
ANT32 ANTC2	Forward Omni Antenna Select	Actuates mast mounted coax switch 4SW022 so that forward omni is connected to switch 1SW022. If 1SW022 is in ANT22/ANTB2 command state, forward omni is connected to rotary joint.	RANT3S	Converse of above, i.e., Logic "0" (= Fwd Omni).

NOTE #2: If the constraint of never switching "HOT" (Refer to Section 3.7.3.4) is honored, the following TM will not be monitorable when the command is executed (no downlink present).

NOTE #3: If the constraint of never switching "HOT" (refer to Section 3.7.3.4) is honored, the following TM will be monitorable when the command is executed only if the downlink is via the Aft Omni.

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TABLE 3.7.4-1 (cont'd)

ORBITER COMMUNICATIONS SUBSYSTEM COMMAND RESPONSE

COMMAND	COMMAND TITLE	COMMAND DESCRIPTION AND SUBSYSTEM INTERNAL RESPONSE	TM MNEMONIC	TM TITLE AND/OR REMARKS
SWITCH DRIVERS (cont'd)				
AMP31 AMPC1	Amplifier 3 Select	Actuates coax switch 3SW022 so that Amplifier 3 is connected to one of the inputs of the 3 dB hybrid.	RAMP3S	Amplifier 3/4 Switch Position = Bilevel logic "1" (= Amp3). Switch 3SW022 must be in this position before sending command AMP39/AMPC9 (turns PA3 "ON").
AMP41 AMPD1	Amplifier 4 Select	Actuates coax switch 3SW022 so that Amplifier 4 is connected to one of the inputs of the 3 db hybrid.	RAMP3S	Amplifier 3/4 Switch Position = Bilevel logic "0" (=Amp). Switch 3SW022 must be in this position before sending command AMP49/AMPD9 (turns PA4 "ON").
AMP11 AMPA1	Amplifier 1 to Low Power/ Amplifier 2 to High Power Select	Actuates coax switch 3SW019 so that PA2 is connected to one of the inputs of the 3 dB hybrid and PA1 is connected to switch 2SW019.	RAMP1S	SEE NOTE #4. Amplifier 2/1 to High/Low Power Status. Bilevel logic "0" (= PA1 to Hi).
AMP12 AMPA2	Amplifier 1 to High Power/ Amplifier 2 to Low Power Select	Actuates coax switch 3SW019 so that PA1 is connected to one of the inputs of the 3 dB hybrid and PA2 is connected to switch 2SW019.	RAMP1S	SEE NOTE #4. Amplifier 2/1 to High/Low Power Status. Bilevel logic "1" (= PA2 to Hi).
EXC11 EXCA1	Exciter 1 Select	Actuates coax switch 2SW022 so that the exciter output from transponder number 1 is connected to the 6 dB power divider that drives the amplifiers.	RXCTRS	SEE NOTE #5. Exciter 1/2 Switch Position = Bilevel logic "1" (=Exc1). This command should be accompanied by command EXC19/ EXCA9 that turns on Exciter 1 and turns off Exciter 2 for normal operation.
EXC21 EXCB1	Exciter 2 Select	Actuates coax switch 2SW022 so that the exciter output from transponder number 2 is connected to the 6 dB power divider that drives the amplifiers.	RXCTRS	SEE NOTE #5. Exciter 1/2 Switch Position = Bilevel logic "0" (=Exc2). This command should be accompanied by command EXC29/ EXCB9 that turns on Exciter 2 and turns off Exciter 1 for normal operation.
TRANSPONDERS 1 & 2				SEE NOTE #5.
EXC19 EXCA9	Exciter 1 ON/2 OFF	Actuates magnetic latching relay switches inside transponders so that Exciter 1 turns ON and Exciter 2 turns OFF.	RXCT1S RXCT2S	Exciter 1 ON/OFF = Bilevel logic "1" (=Exc1 ON). Exciter 2 ON/OFF = Bilevel logic "0" (=Exc 2 OFF). If transponder 1 is in auxiliary oscillator mode, downlink frequency will shift if Exciter 2 was previously on.
EXC29 EXCB9	Exciter 2 ON/1 OFF	Converse of above.	RXCT1S RXCT2S	SEE NOTE #5. Converse of above. (Interchange "1" & "2) in comment).
EXC1# EXCA#	Exciters 1/2 OFF	Actuates magnetic latching relay switches inside transponders so that both exciters are turned OFF.	RXCT1S RXCT2S	SEE NOTE #5. Exciter 1 ON/OFF = Bilevel logic "0" (= EXC1 OFF). Exciter 2 ON/OFF = Bilevel logic "0" (=EXC2 OFF). Downlink will cease when this command is received.

NOTE #4: If the constraint of never switching "HOT" (refer to Section 3.7.3.4) is honored, the following TM will be monitorable when the command is executed only if non-standard Mode #1 (PA3 or PA4 is ON - described in Section 3.7.3.3) is in use.

NOTE #5: The following TM will not be monitorable when the command is executed, as the downlink has to be re-established.

TABLE 3.7.4-1 (cont'd)

ORBITER COMMUNICATIONS SUBSYSTEM COMMAND RESPONSE

COMMAND	COMMAND TITLE	COMMAND DESCRIPTION AND SUBSYSTEM INTERNAL RESPONSE	TM MNEMONIC	TM TITLE AND/OR REMARKS
TRANSPONDERS 1 & 2 (cont'd)				For whichever exciter is ON, the corresponding following telemetry indicates bilevel logic 1 (= inhibit); the other parameter indicates bilevel logic 0 (unpowered).
COH1Ø COHAØ	Inhibit Coherent Mode	Actuates logic inside transponder that is ON that prevents solid state switch from selecting the VCO as the exciter frequency source when the receiver is locked to uplink signal. Auxiliary oscillator is frequency source.	RCOH1S	Exciter 1 Inhibit/Restore coherent mode.
			RCOH2S	Exciter 2 Inhibit/Restore coherent mode.
				The downlink frequency will correspond to auxiliary oscillator frequency of the exciter that is on. Commands may still be received & processed.
COH19 COHA9	Restore Coherent Mode	Allows solid state switch inside transponder that is on to select VCO as exciters' frequency source, when the receiver is locked to an uplink signal. Downlink frequency will be 240/221 times uplink frequency.	RCOH1S	Exciter 1 Inhibit/Restore coherent mode = 0 (= Restore)
			RCOH2S	Exciter 2 Inhibit/Restore coherent mode = 0 (= Restore)
				Downlink frequency will be 240/221 times the uplink frequency when phase locked.
MIH11 MIHA1	High Mode Index Select	Actuates solid state switch inside transponder that is ON to cause 1.1Ø radians mod index to be selected for exciters.	NONE	No TLM. Downlink carrier suppression will increase by approximately 6.4 dB.
MIL11 MILA1	Low Mod Index Select	Actuates solid state switch inside transponder that is ON to cause 0.65 radians mod index to be selected for exciters.	NONE	No TLM. Downlink carrier suppression will decrease by approximately 6.4 dB.
VCO19 VCOA9	Exciter Transfer to VCO ON (Test only-wired out)	Actuates solid state switch inside transponder that is ON that causes exciter frequency source to be VCO, when the receiver is not locked to uplink signal.		For whichever exciter is on, the corresponding following telemetry indicates bilevel logic 1 (= ON); the other parameter indicates logic 0 (= Unpowered).
			RVCO1S	Exciter 1 Transfer to VCO ON/OFF
			RVCO2S	Exciter 2 Transfer to VCO ON/OFF
				Downlink frequency will shift from auxiliary oscillator frequency to 240/221 times the receiver best lock frequency.
VCO1Ø VCOAØ	Exciter Transfer to VCO OFF (Test only-wired out)	Actuates solid state switch inside transponder that is ON that causes exciter frequency source to revert to auxiliary oscillator whether receiver is or is not locked to uplink signal.	RVCO1S	Exciter 1 transfer to VCO ON/OFF = 0 (= OFF)
			RVCO2S	Exciter 2 Transfer to VCO ON/OFF = 0 (= OFF)
				Downlink frequency will shift back to auxiliary oscillator frequency of the exciter that is ON.
X-BAND TRANSMITTER				
XMT19 XMTA9	X-Band Transmitter ON	Actuates solid state switch within x-band transmitter that cause transmitter to turn ON.	RXMTRW	X-Band Transmitter RF Output analog telemetry will indicate approximately 1 watt output. Transponder/Exciter 1 or 2 must be ON for an output to exist.
			RXMTRT	X-Band Transmitter Temperature conditioned analog telemetry will indicate an increase in temperature.
			PBUSLI	Spacecraft load current increases by approximately 1.5 amperes.
			PLIMTI	Bus voltage limiter current likely decreases by the same amount as PBUSLI had increased.
XMT1Ø XMTAØ	X-Band Transmitter OFF	Converse of above.	RXMTRW RXMTRT PBUSLI PLIMTI	Converse of above.

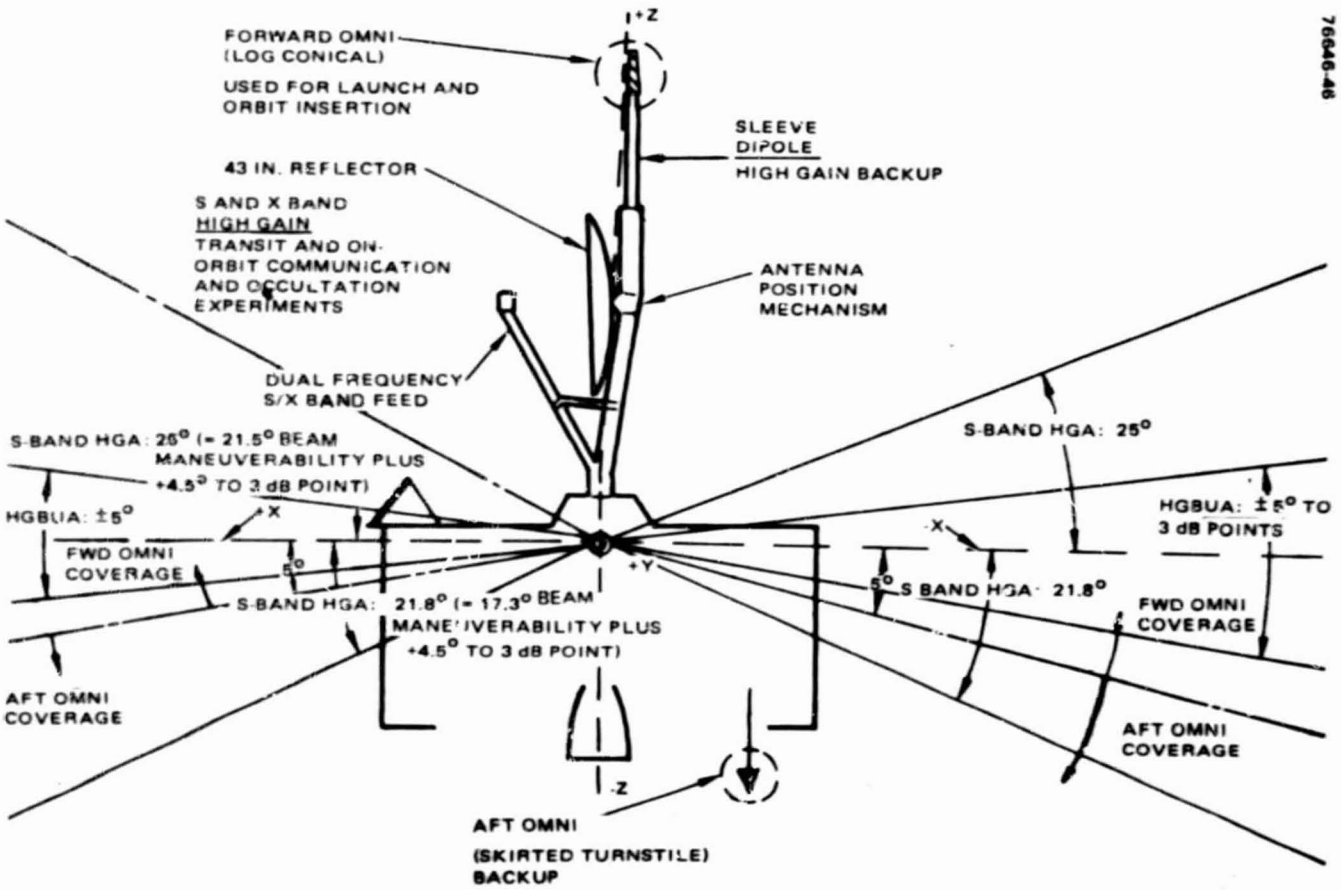
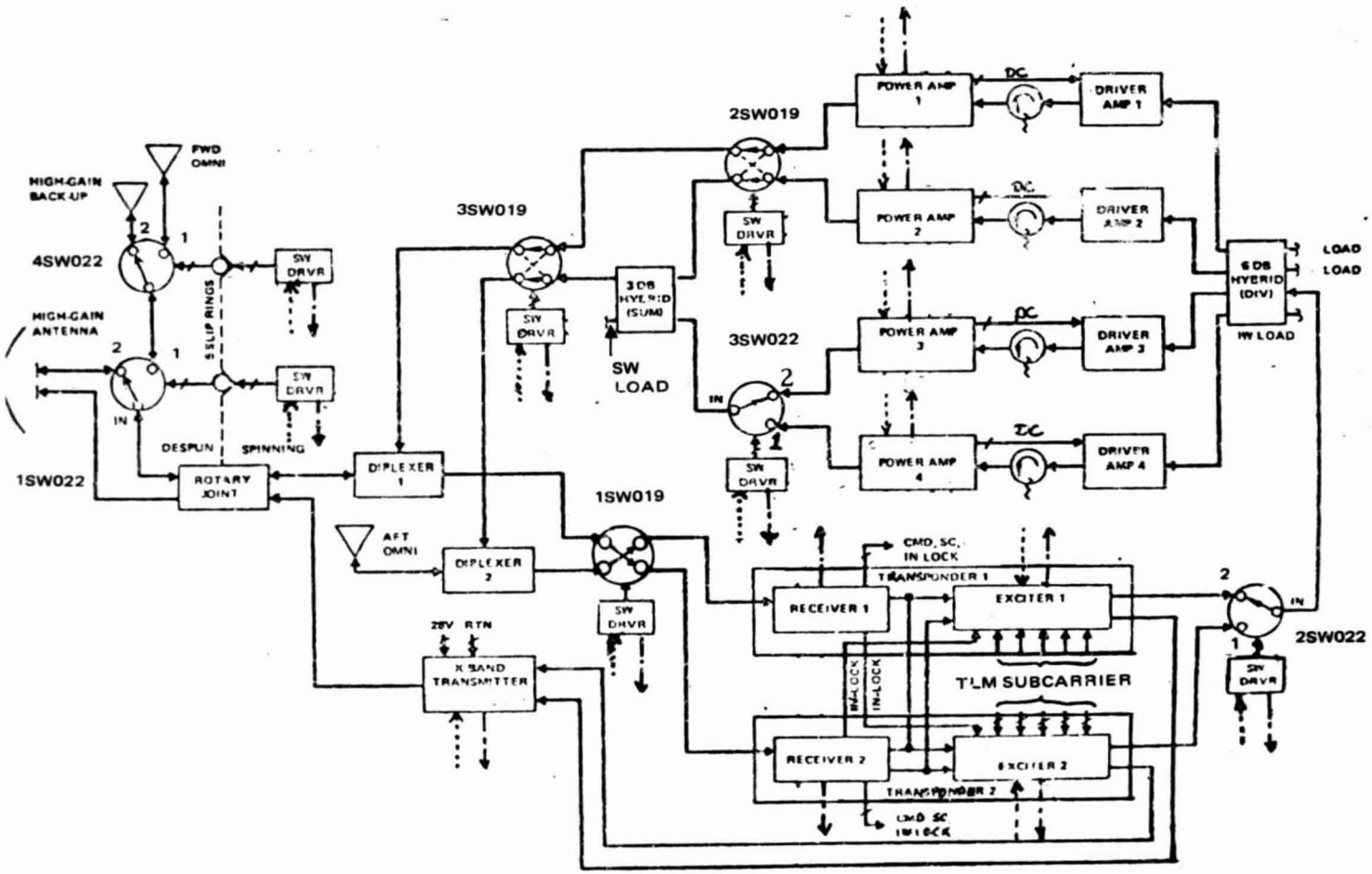


FIGURE 3.7.1-1. ORBITER COMMUNICATION SUBSYSTEM ANTENNA CONFIGURATION



Note: Each RF switch is shown in the "2" position.

Figure 3.7.1-2. Orbiter Communication Subsystem Functional Block Diagram

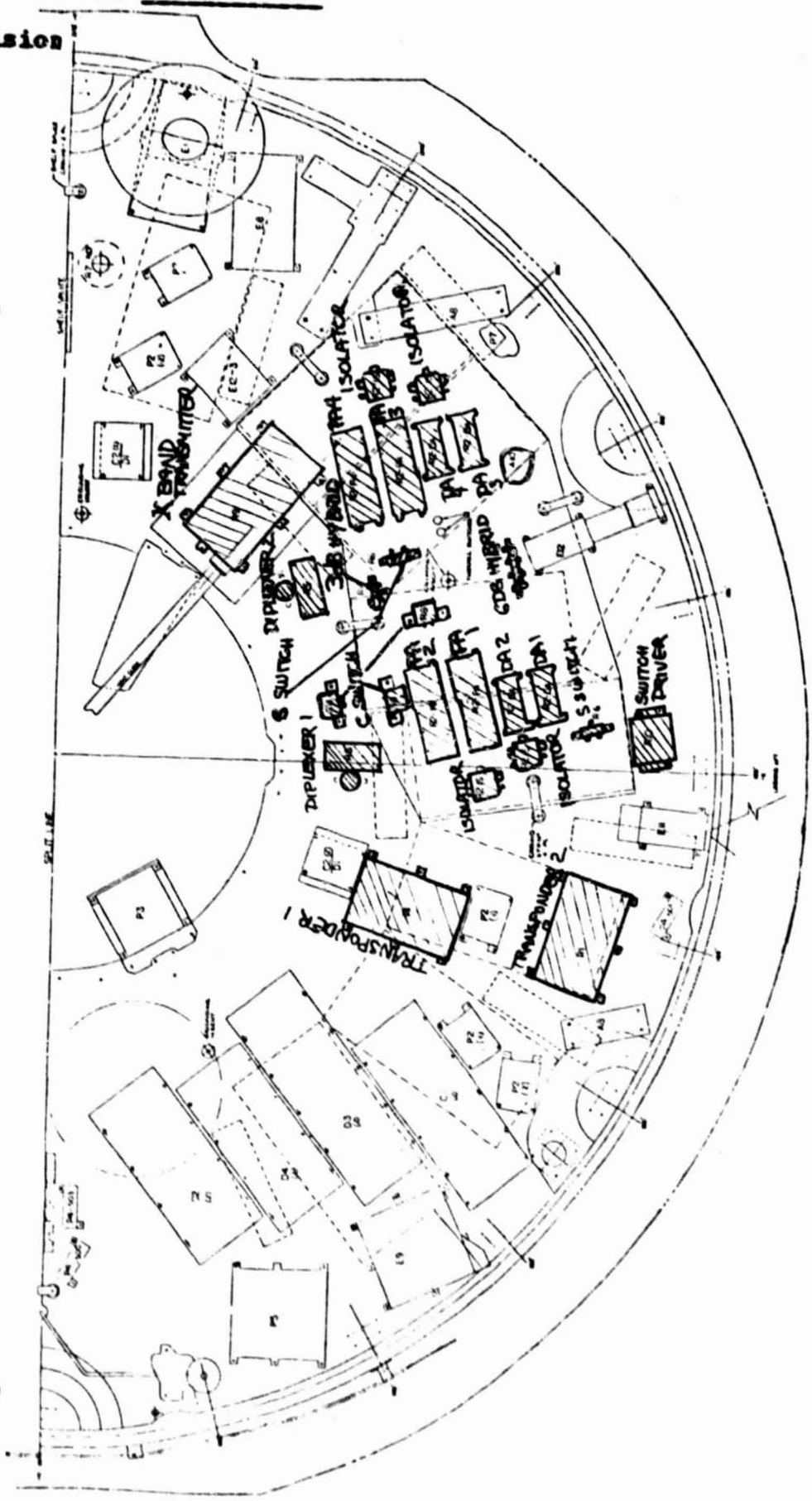


Figure 3.7.1-3. Orbiter Communication Subsystem Shelf Layout

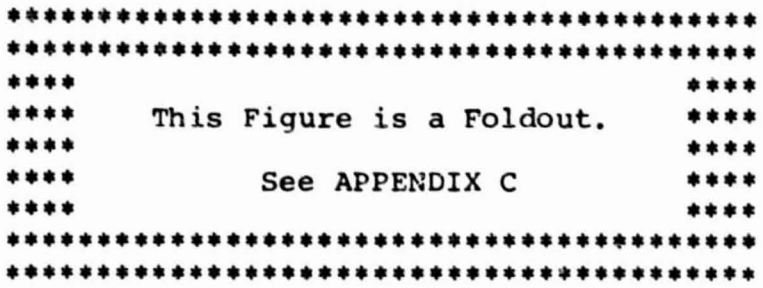


Figure 3.7.1-4.

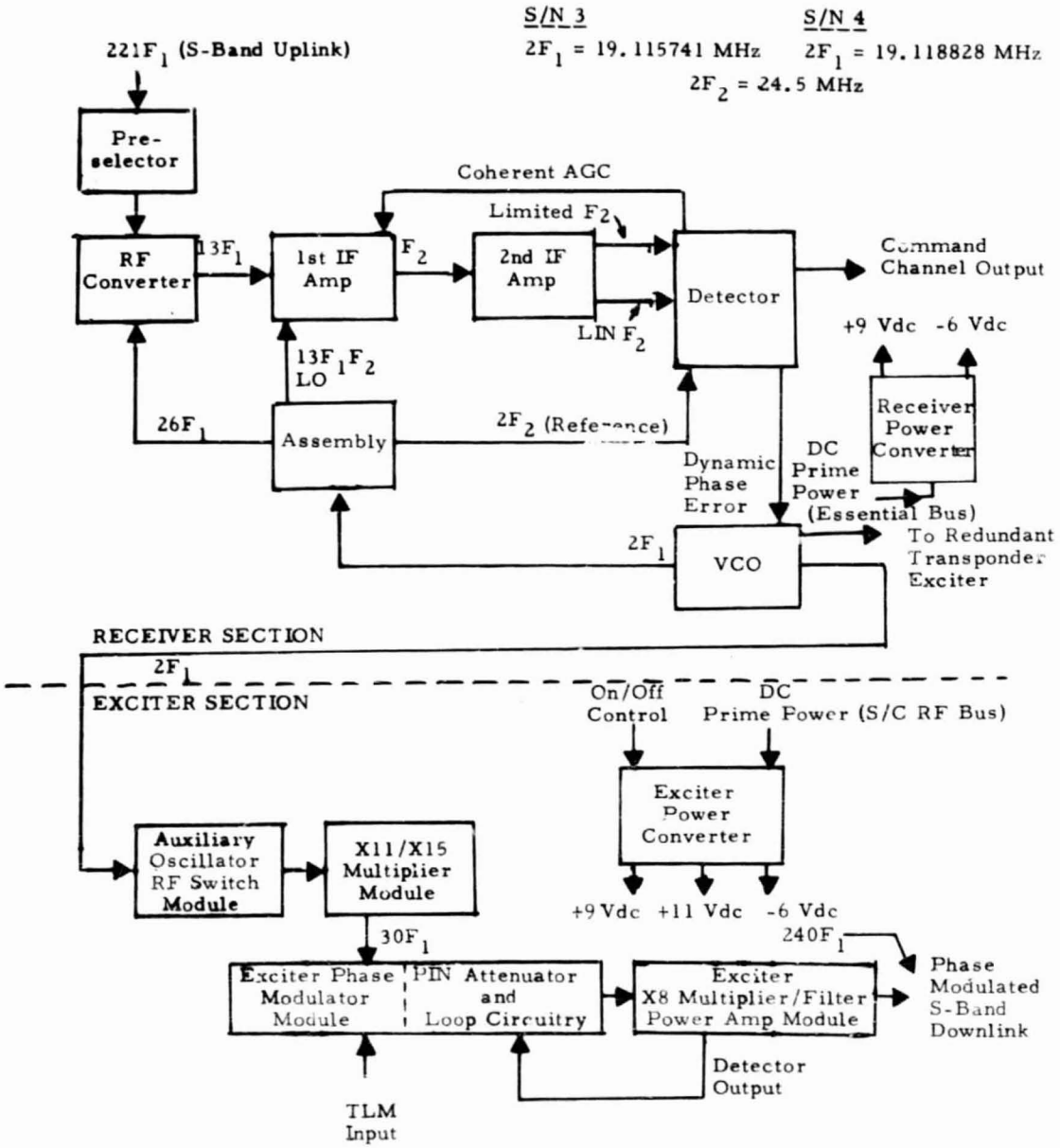


Figure 3.7.2.1-1. Transponder Functional Block Diagram

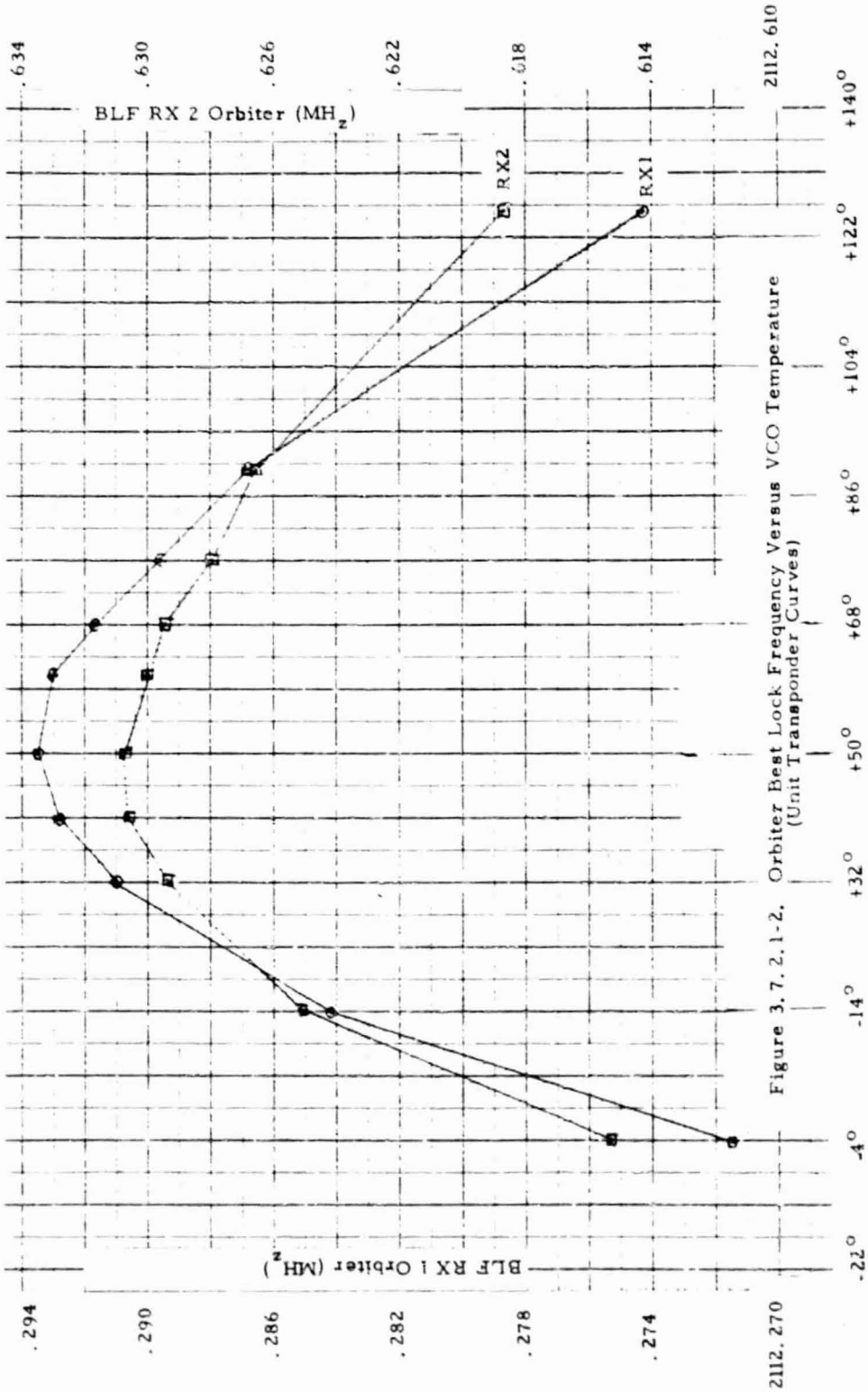


Figure 3.7.2.1-2. Orbiter Best Lock Frequency Versus VCO Temperature (Unit Transponder Curves)

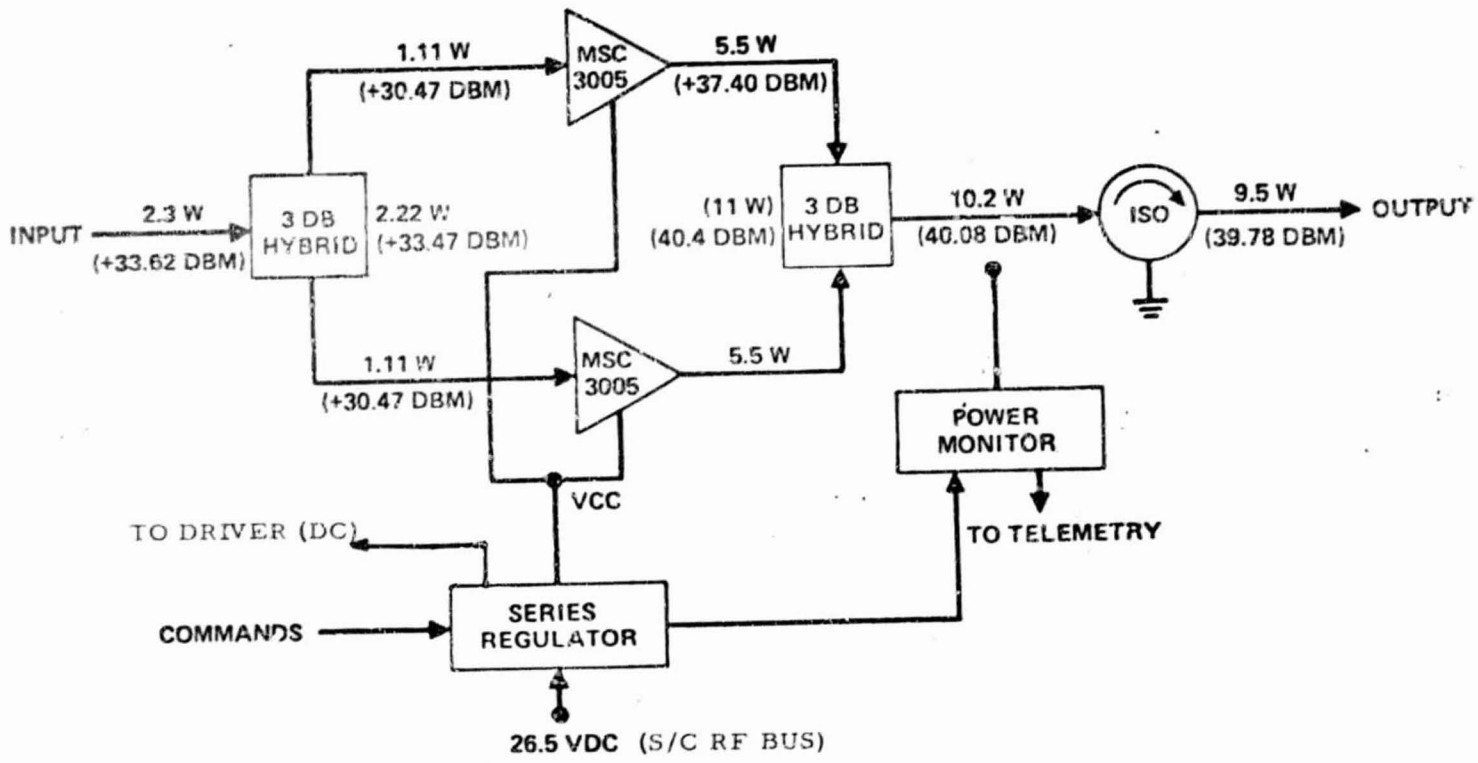


Figure 3.7.2.3-1. Power Amplifier Block Diagram, With Nominal Signal Levels Shown

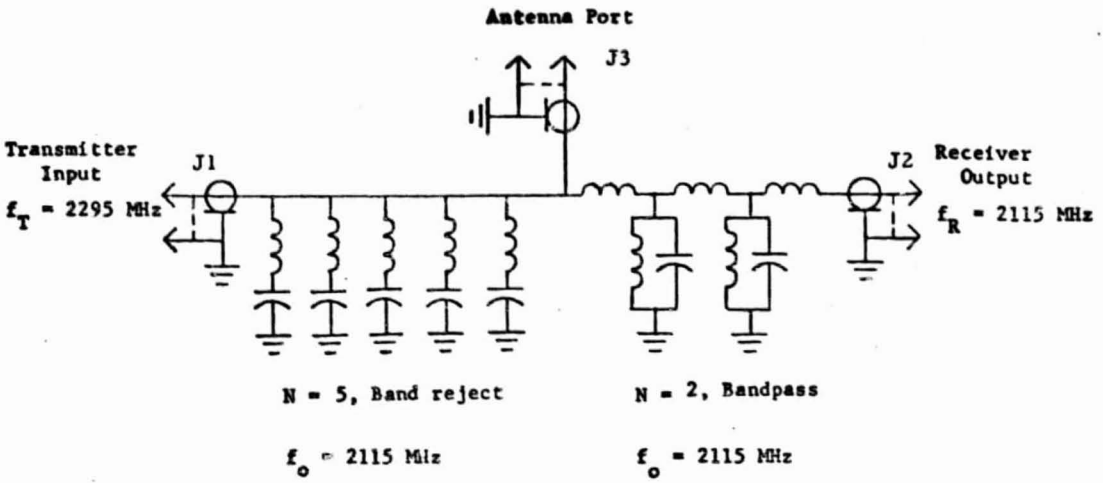


Figure 3.7.2.4-1A. Diplexer Schematic Diagram.

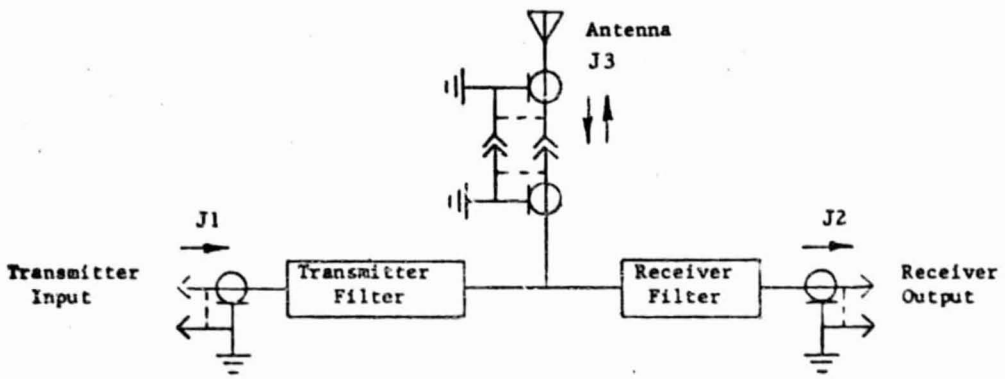


Figure 3.7.2.4-1B. Diplexer Block Diagram.

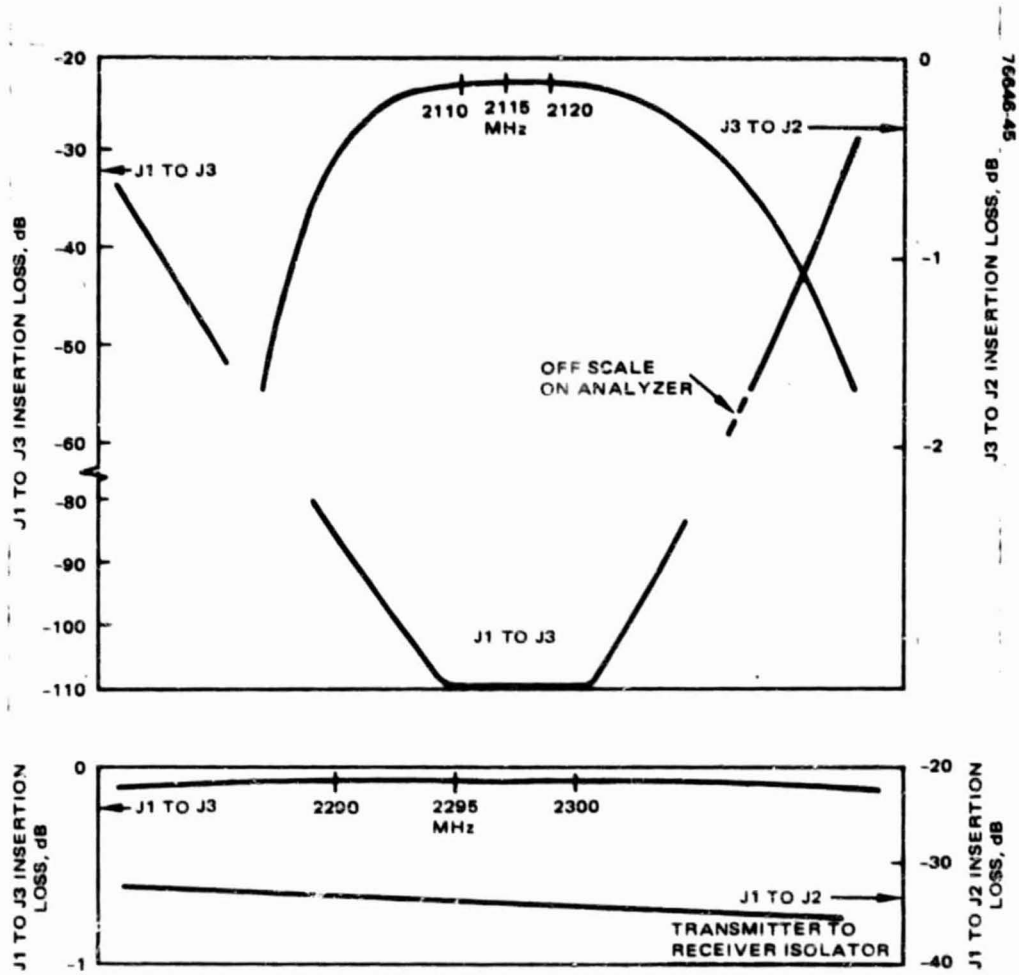


FIGURE 3.7.2.4-2. DIPLEXER MEASURED RESPONSE (AT UNIT LEVEL)

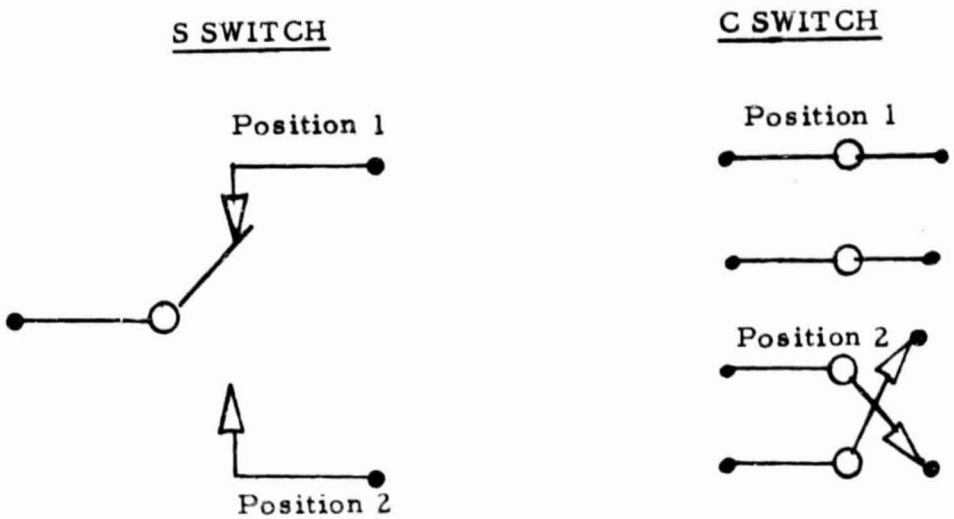


Figure 3.7.2.5-1. RF Switch Schematics

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FIGURE 3.7.2.7-1. ORBITER OMNI ANTENNAS TRANSMIT PERFORMANCE IN TERMS OF RADIATED POWER DENSITY

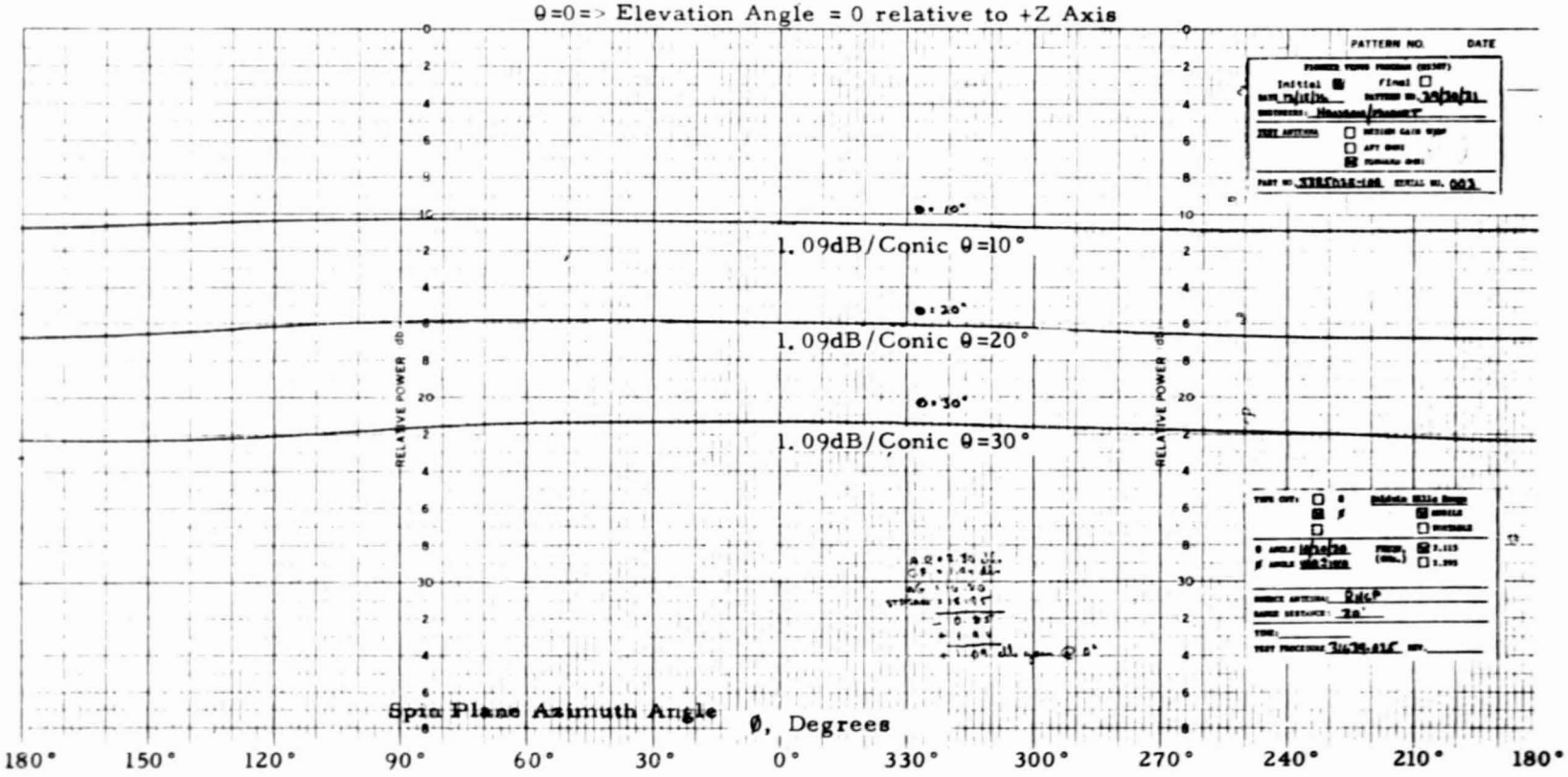


Figure 3.7.2.7-2A. Forward OMNI RHCP Gain Versus Azimuth Angle (Ø) as a Function of Elevation Angle θ

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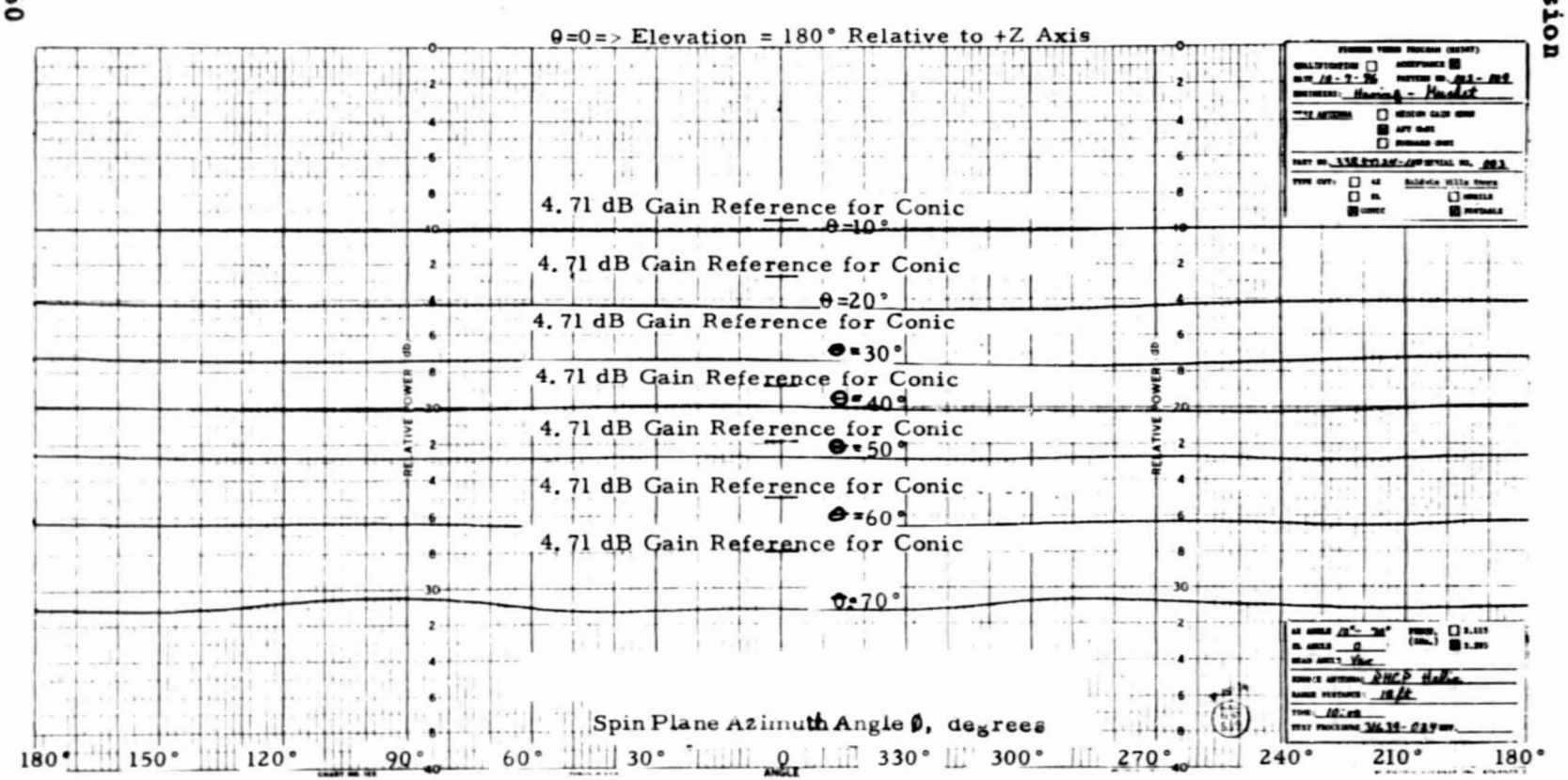


Figure 3.7.2.7-2B. AFT OMNI Right Hand Circularly Polarized Gain Versus Azimuth Angle (Ø) as a Function of Elevation Angle θ

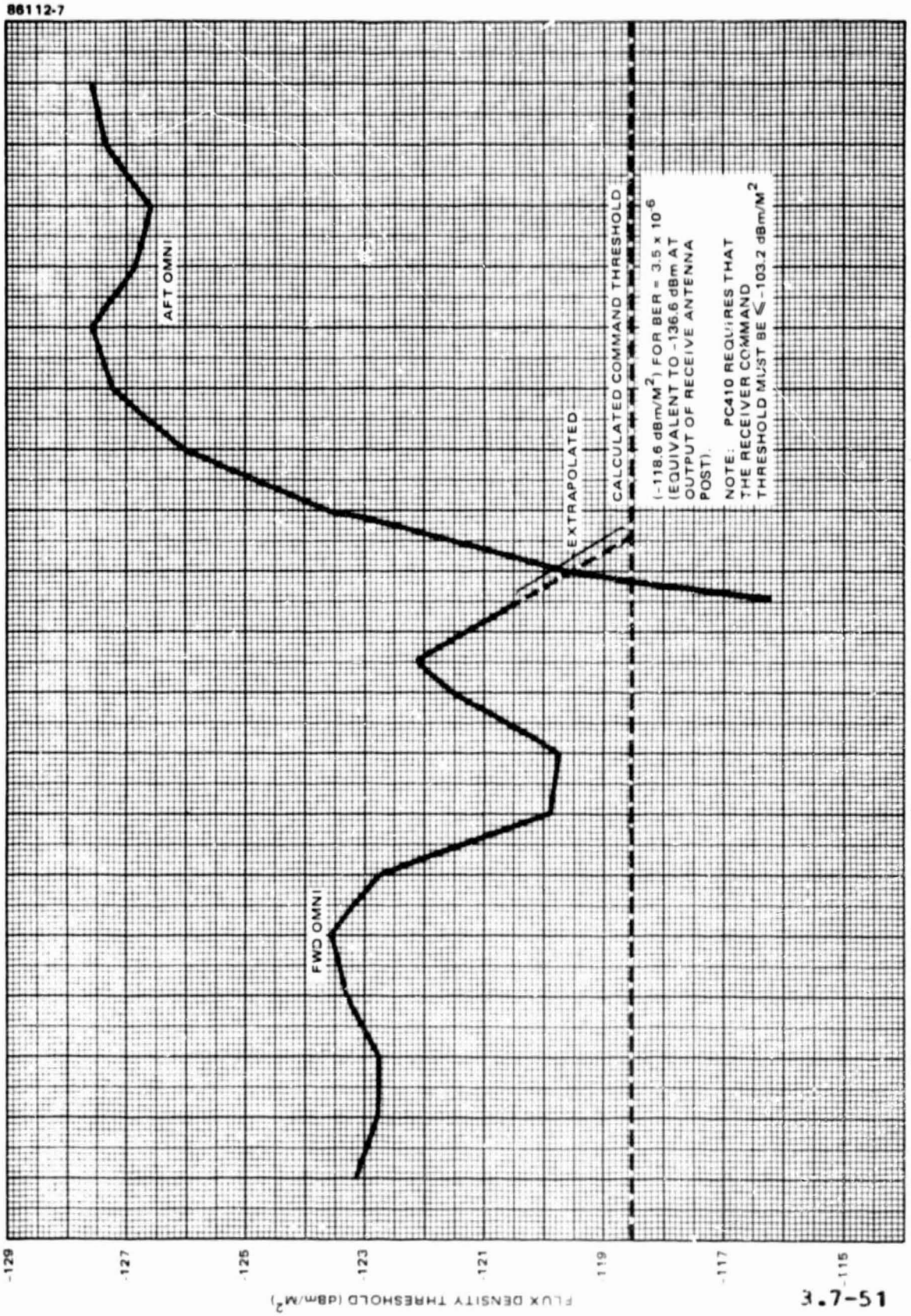


FIGURE 3.7.2.7-3. ORBITER OMNI ANTENNAS RECEIVE PERFORMANCE

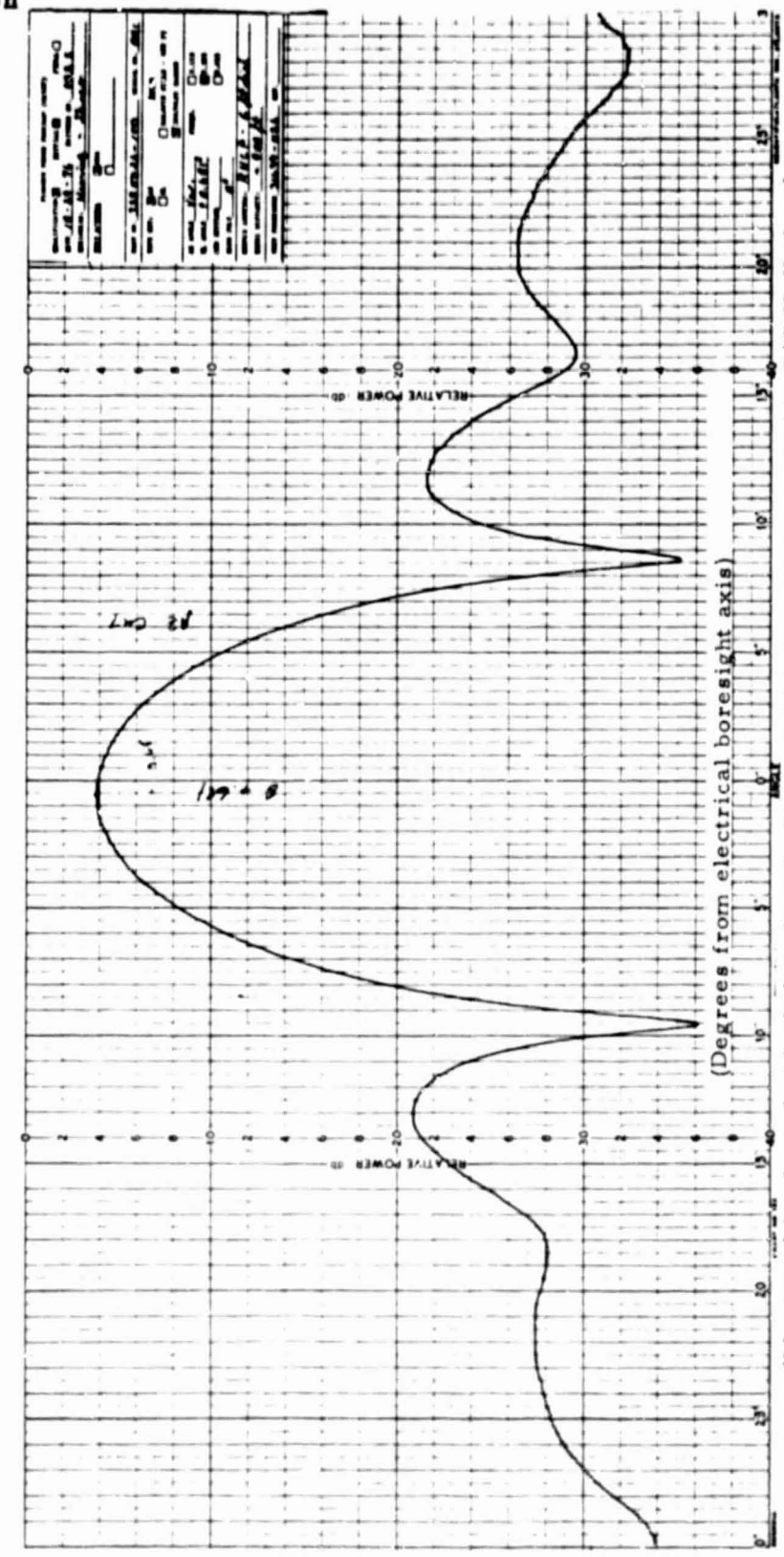
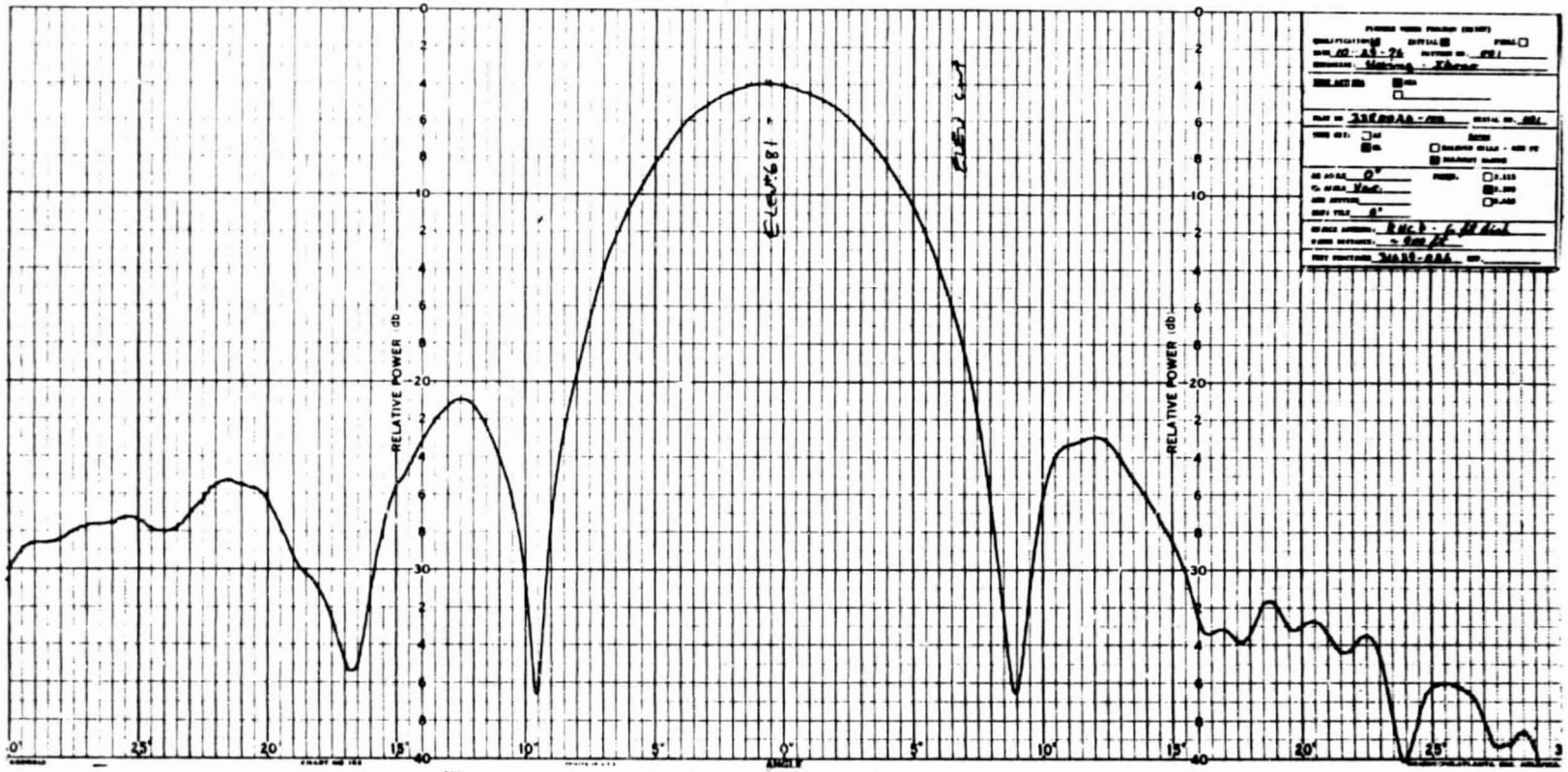


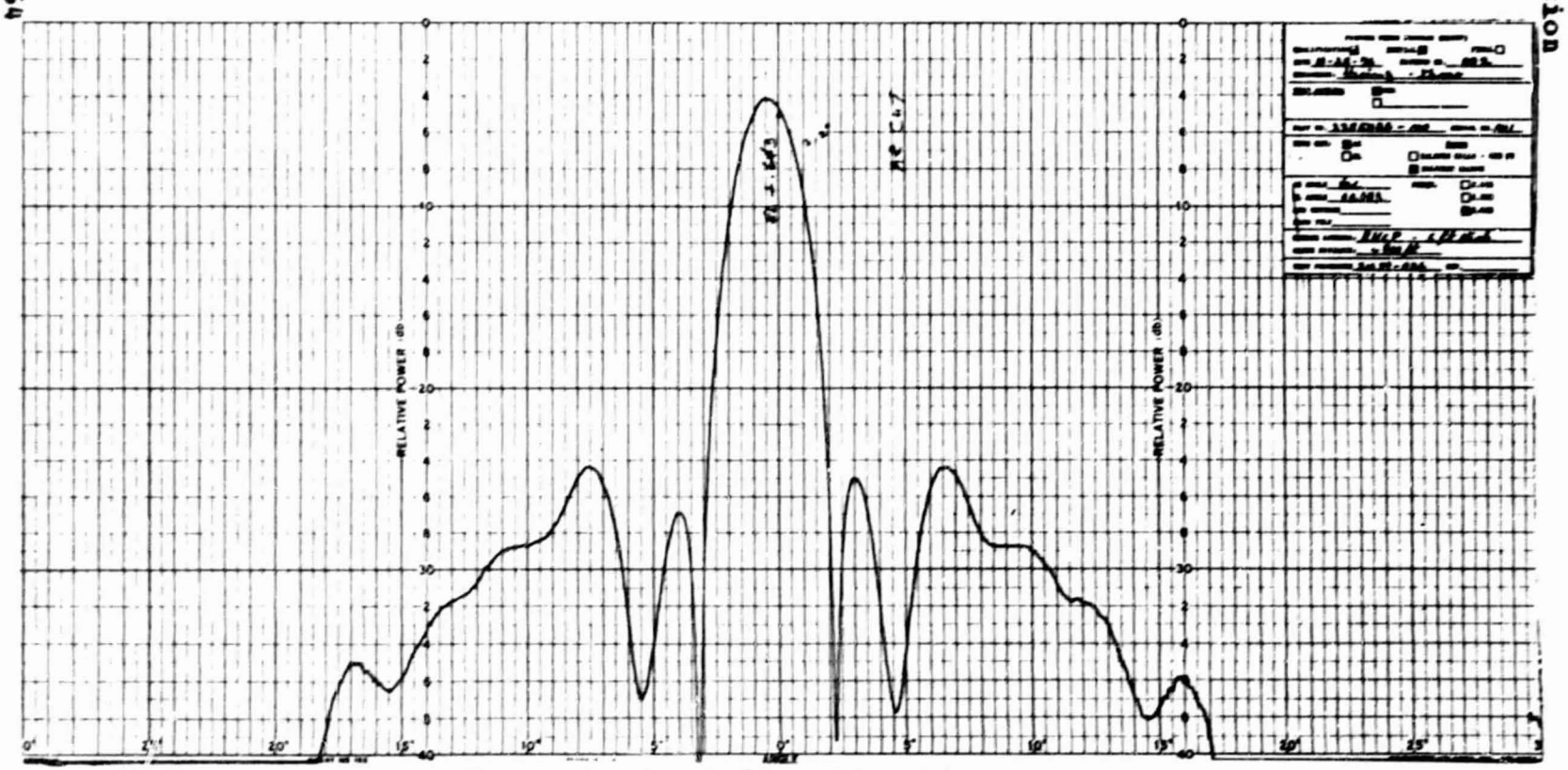
Figure 3.7.2.7-4. HGA Azimuth Pattern, 2.295 GH_z



(Degrees from electrical boresight axis)

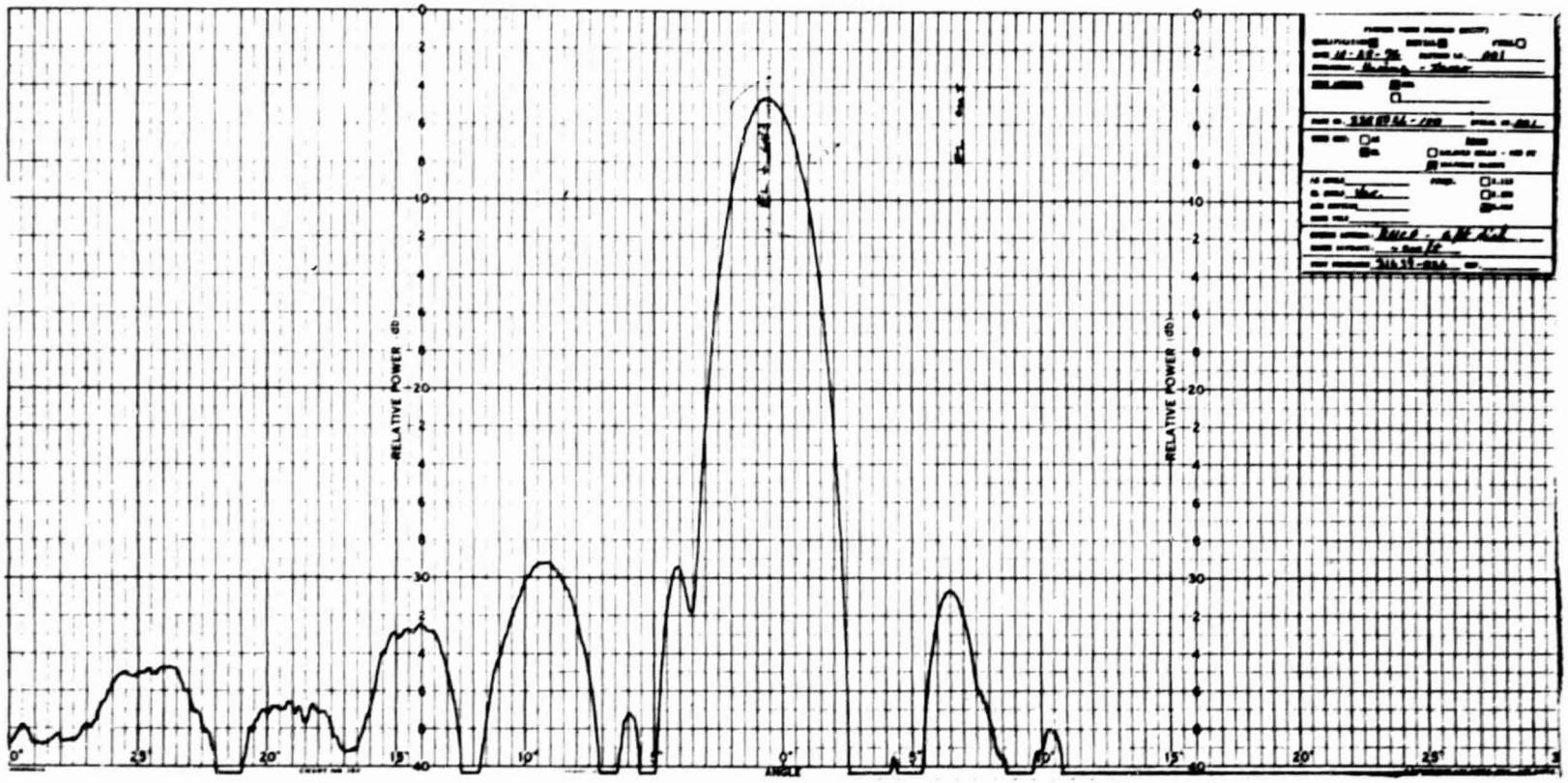
(Measured in plane that includes electrical boresight axis, and is perpendicular to S/C X-Y plane, while boresight axis is parallel to S/C X-Y plane).

Figure 3.7.2.7-5. HGA Elevation Pattern, 2.295 GH_z



(Degrees for electrical boresight axis)
(Measured in plane that includes electrical boresight axis, and is parallel to S/C X-Y plane).

Figure 3.7.2.7-6. HGA Azimuth Pattern, 8.415 GHz



(Degrees from electrical boresight axis)
(Measured in plane that includes electrical boresight axis, and is perpendicular to S/C X-Y plane, while boresight axis is parallel to S/C X-Y plane).

Figure 3.7.2.7-7. HGA Elevation Pattern, 8.415 GH_z

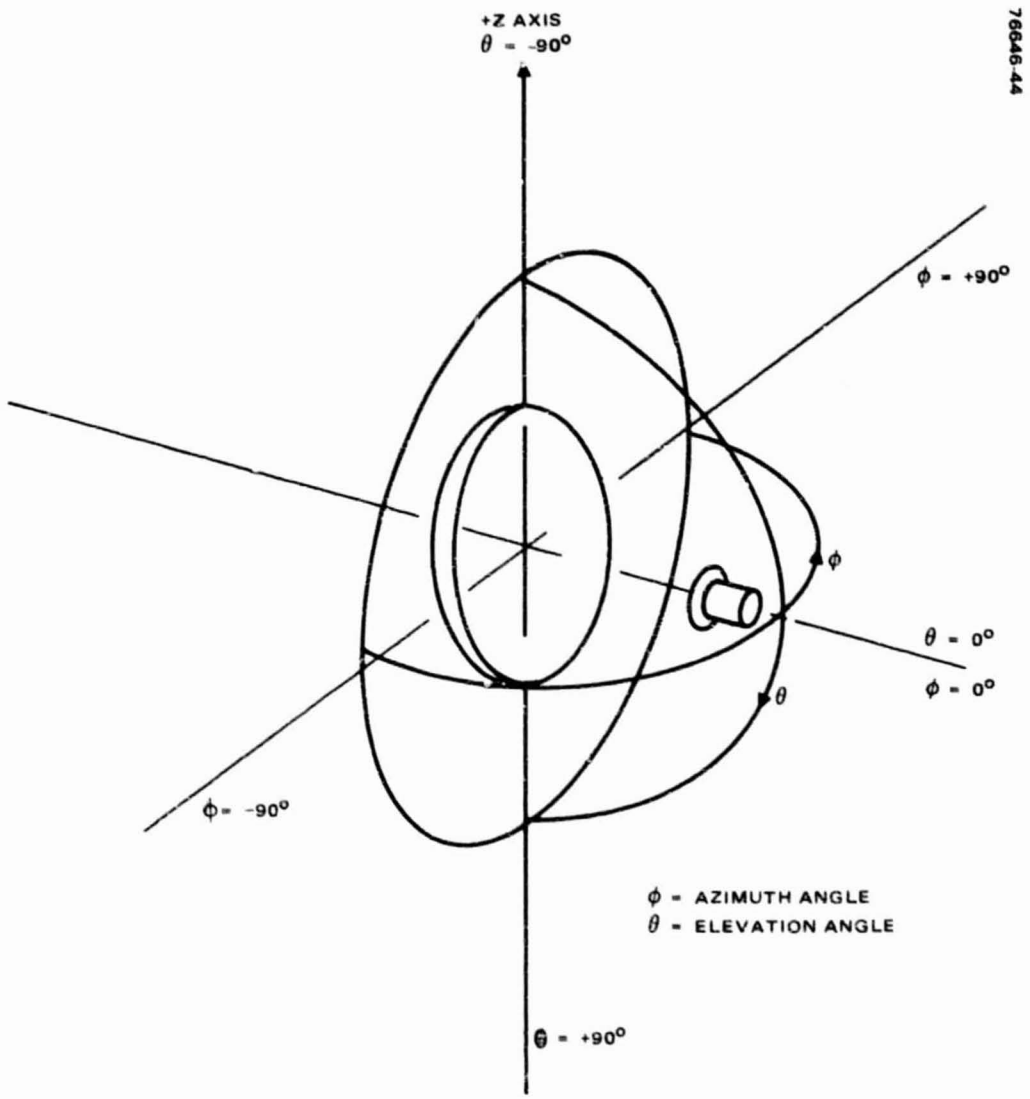
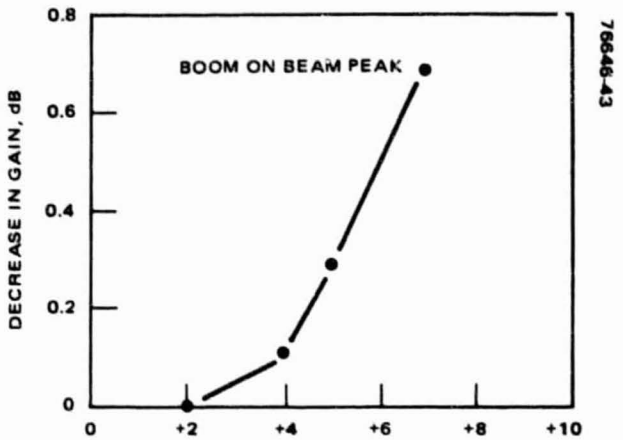
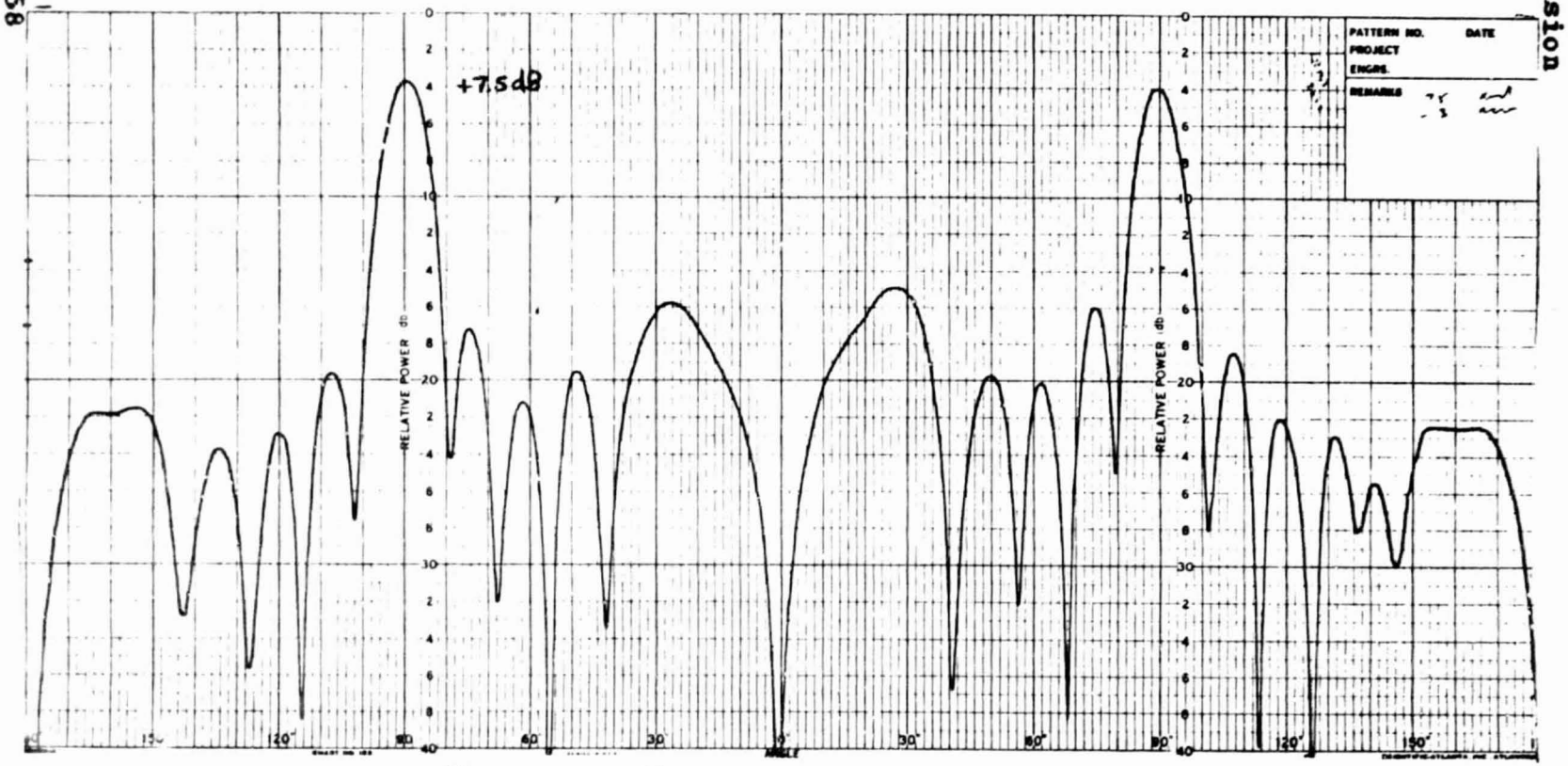


FIGURE 3.7.2.7-8. HGA PATTERN COORDINATES



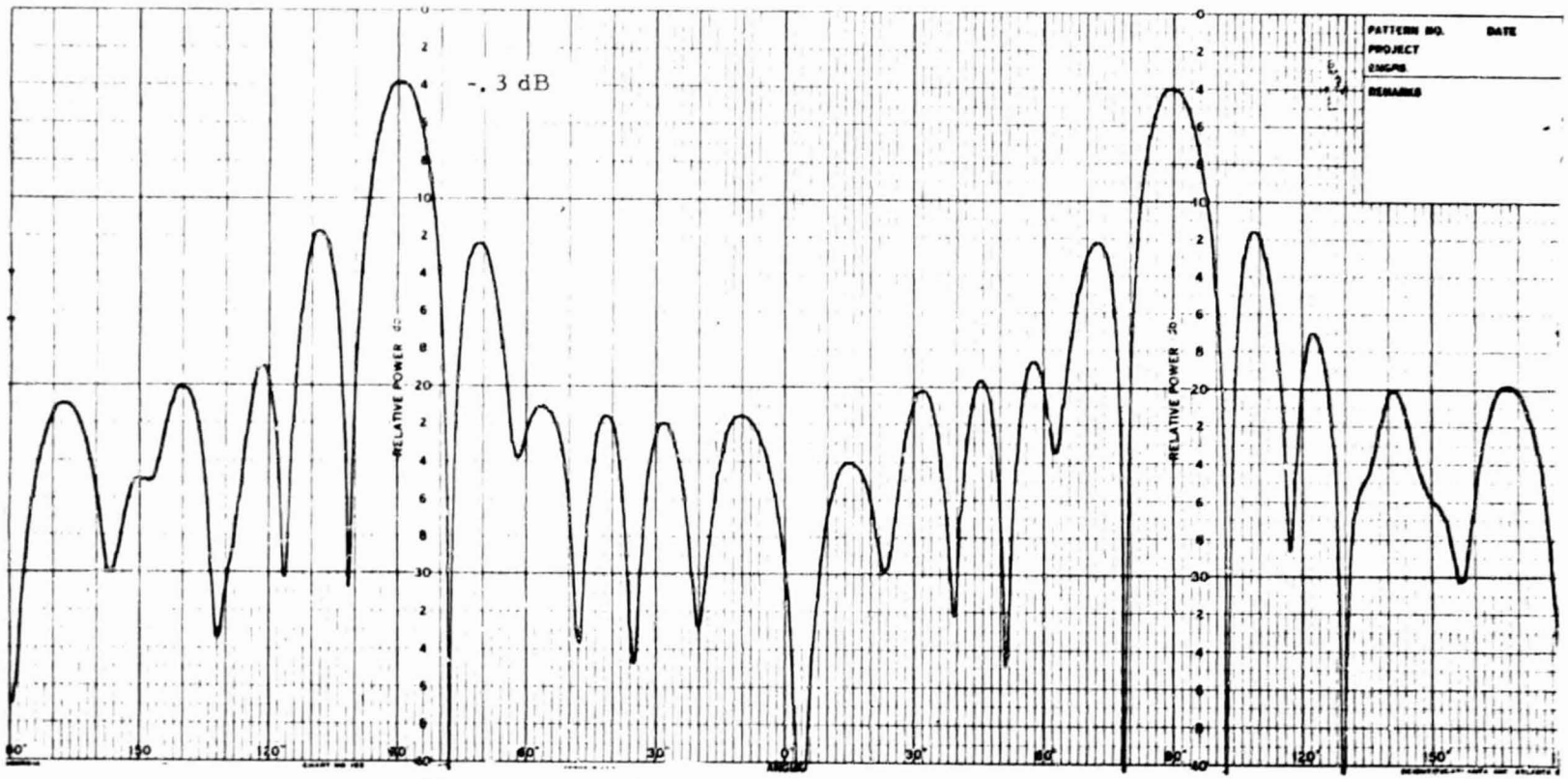
NOTE: DISH BORESIGHT IS LOCATED AZIMUTHLY AT SAME ANGLE AS MAGNETOMETER BOOM FOR EACH OF ABOVE PLOTS.

FIGURE 3.7.2.7-9. DECREASE IN HGA GAIN DUE TO MAGNETOMETER BOOM



(Degrees from S/C X-Y plane)
(Measured in S/C X-Z plane)

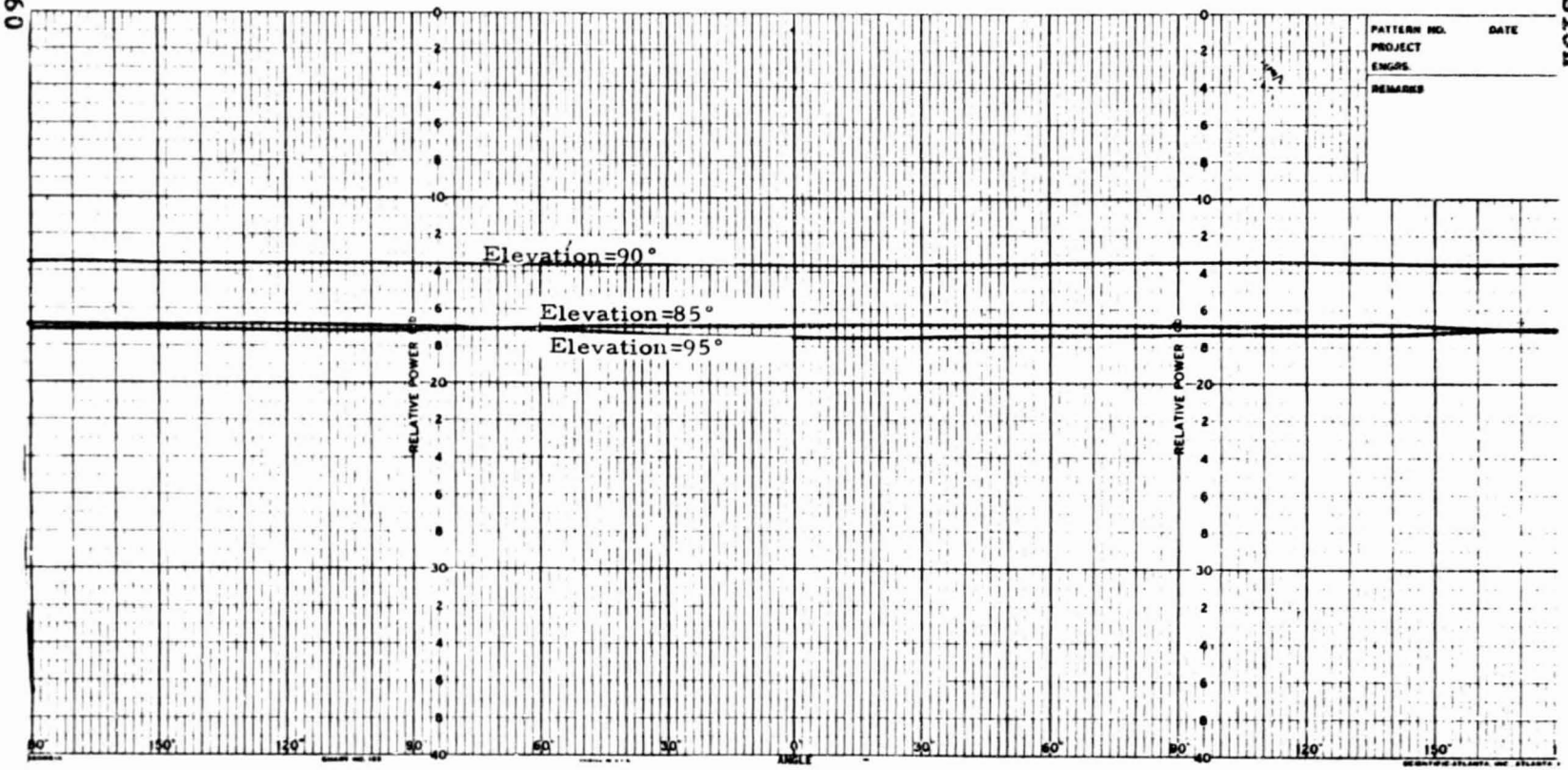
Figure 3.7.2.7-10. HGBUA Elevation Pattern, 2.295 GHz



(Degrees from S/C X-Y plane)

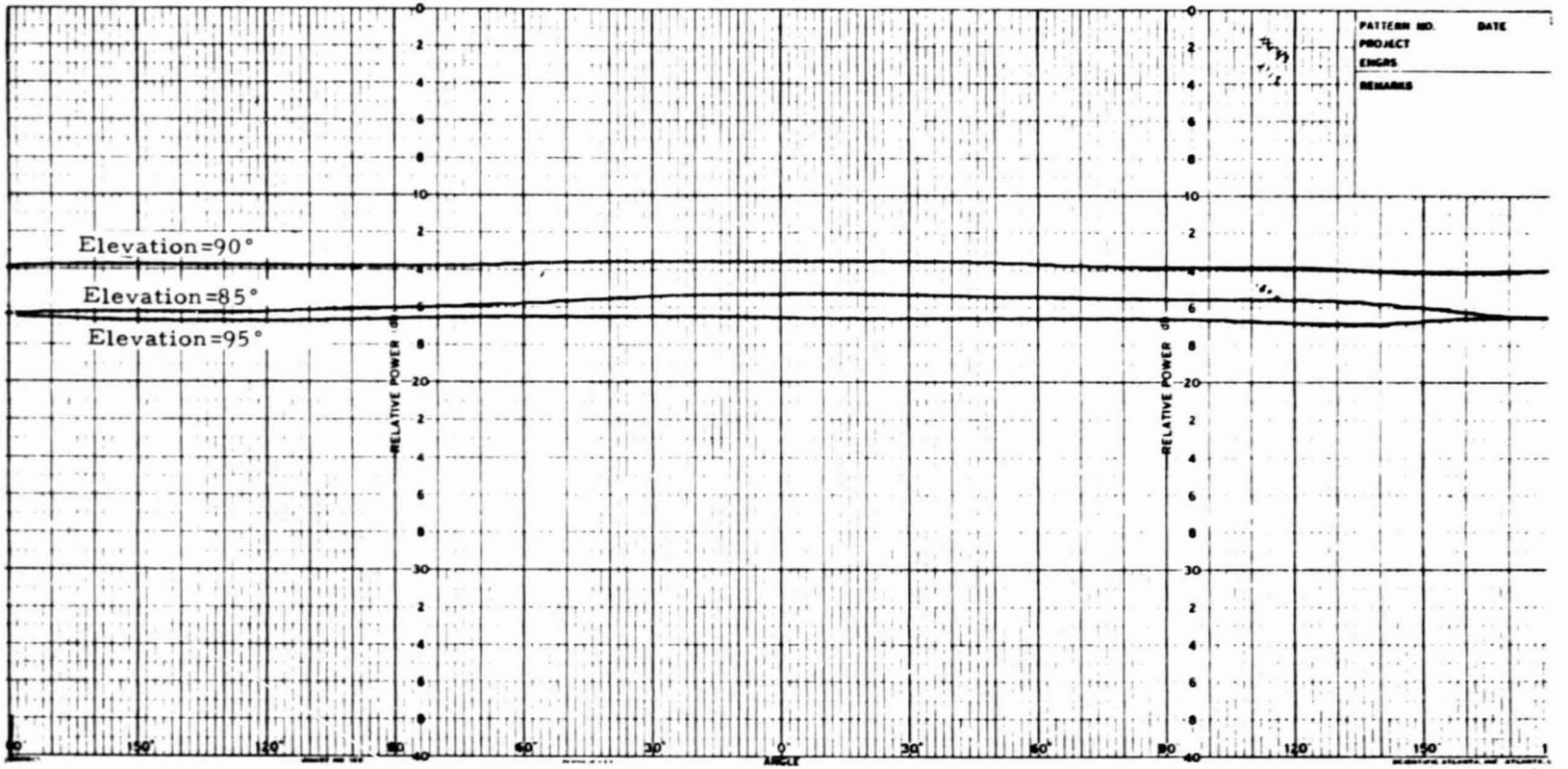
(Measured in S/C X-Z plane)

Figure 3.7.2.7-11. HGBUA Elevation Pattern, 2.115 GHz



Azimuth Angle from S/C +X axis , measured at a constant elevation angle (0°=S/C +Z axis)

Figure 3.7.2.7-12. HGBUA Conic Cut (Azimuth Pattern), 2.295 GHz



Azimuth Angle from S/C +X axis ; measured at constant elevation angle (0°=S/C +Z axis)

Figure 3.7.2.7-13. HGBUA Conic Cut (Azimuth Pattern), 2.115 GHz

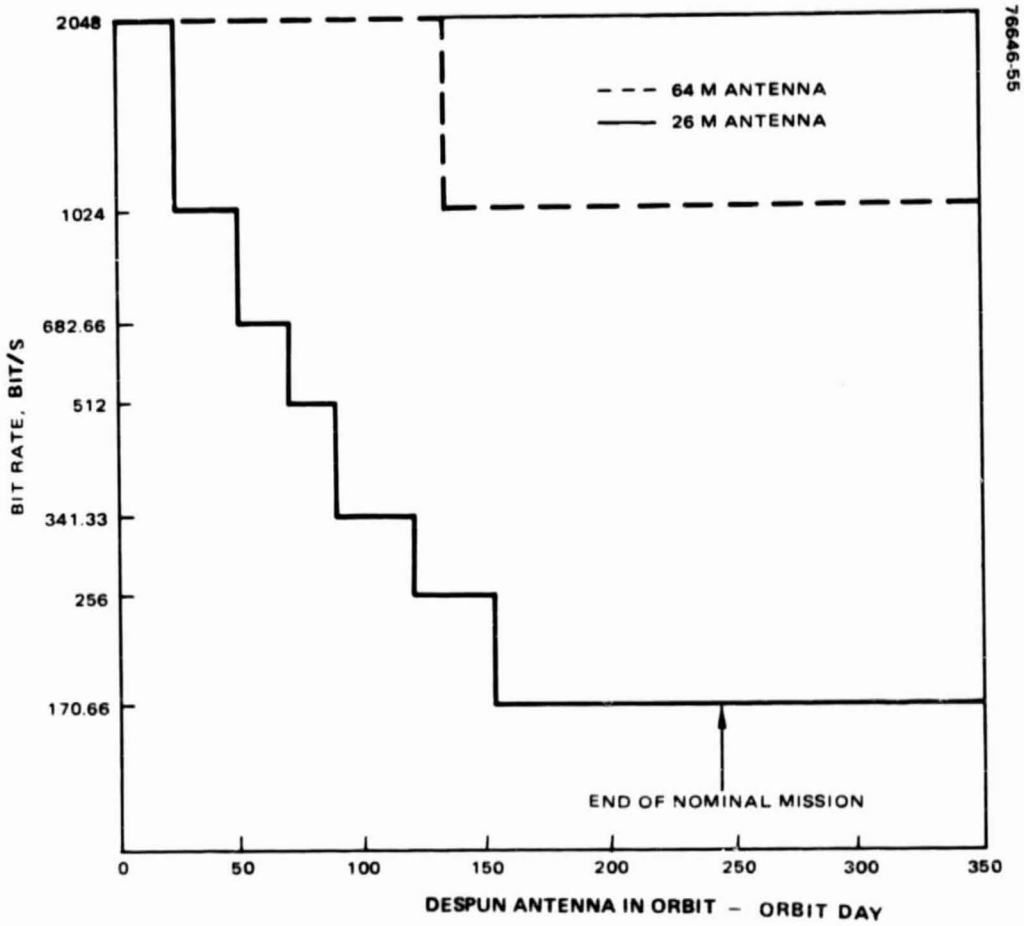


FIGURE 3.7.3-1. ORBITER DATA RATE CAPABILITY

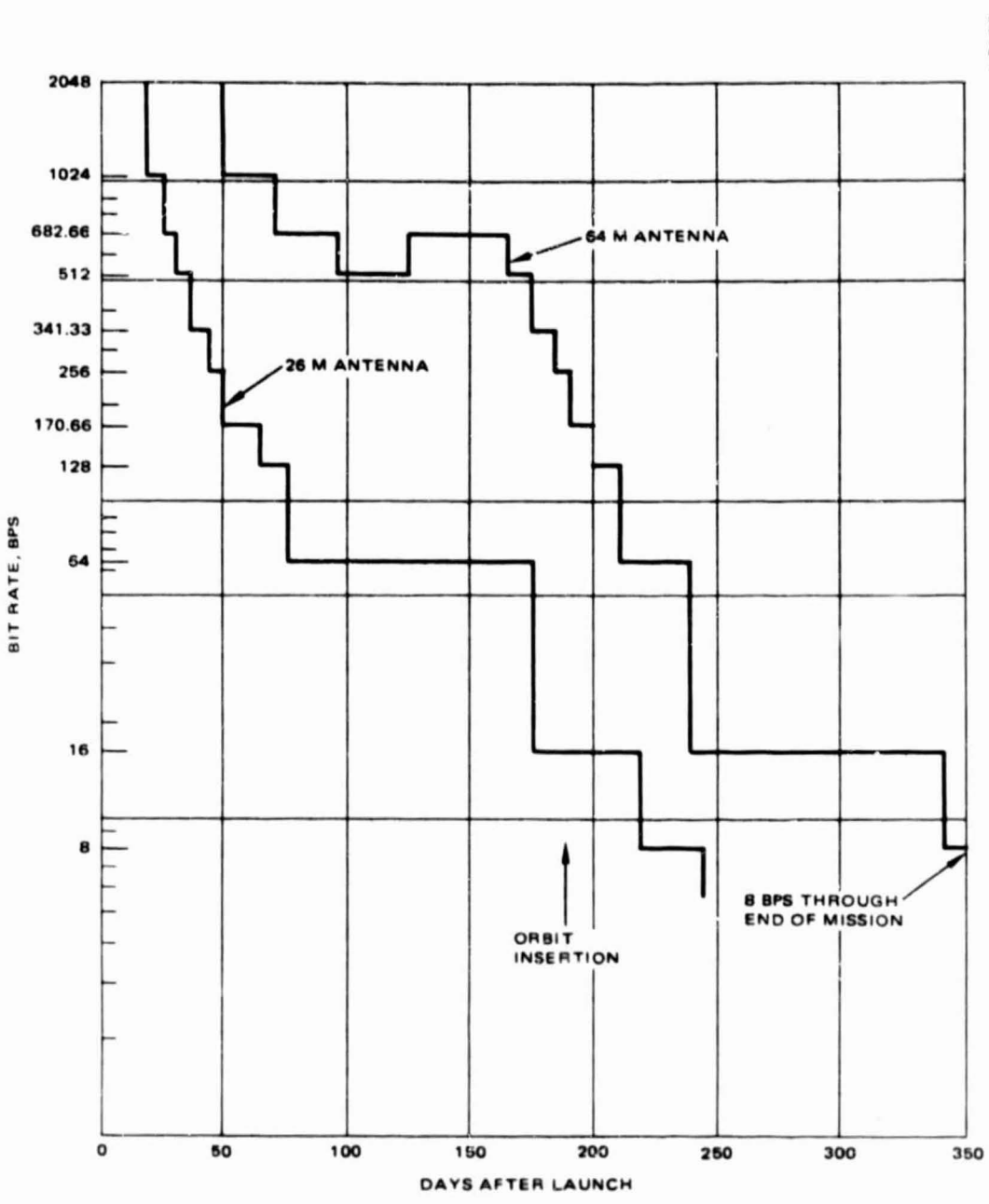


FIGURE 3.7.3-2. ORBITER DATA RATE CAPABILITY
BACKUP (SLEEVE DIPOLE) ANTENNA

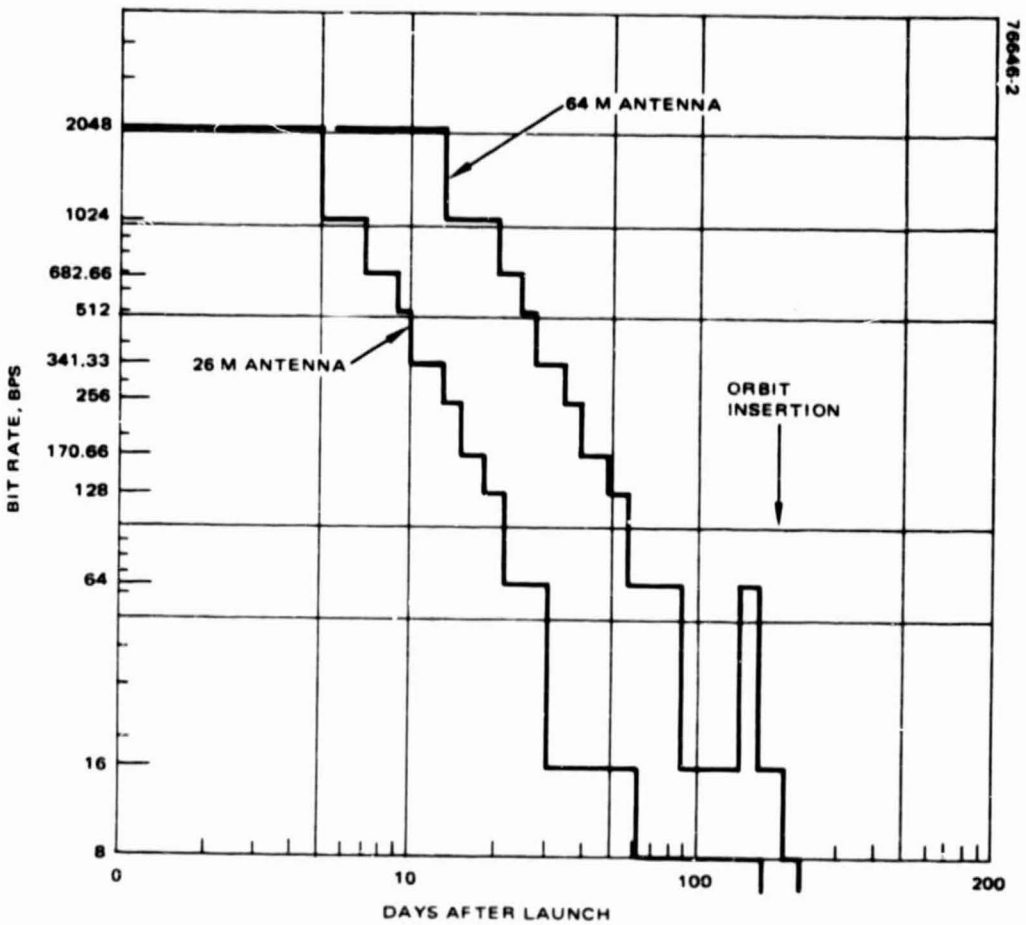
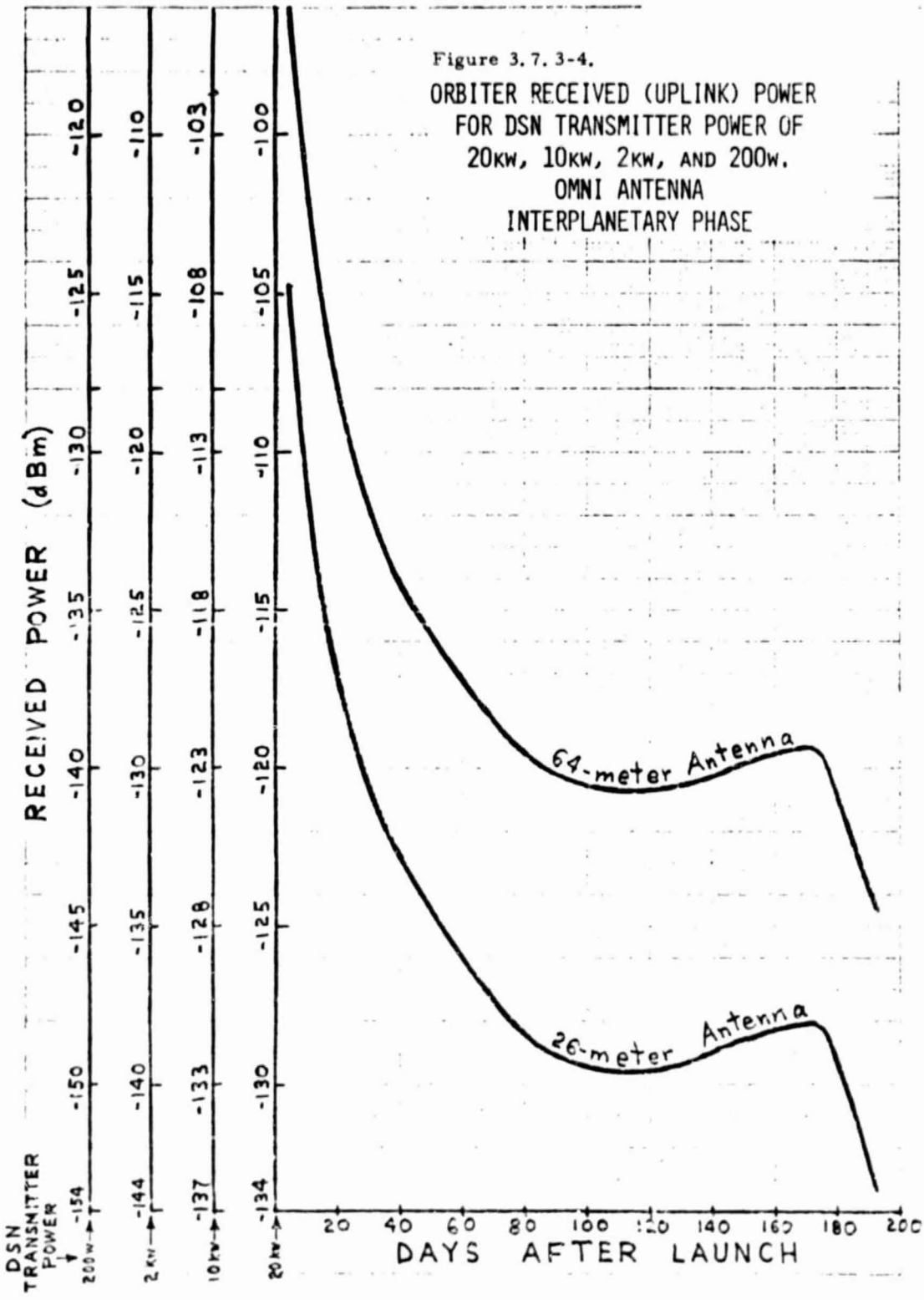


FIGURE 3.7.3-3. ORBITER DATA RATE CAPABILITY
OMNI ANTENNA

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Figure 3.7.3-4.



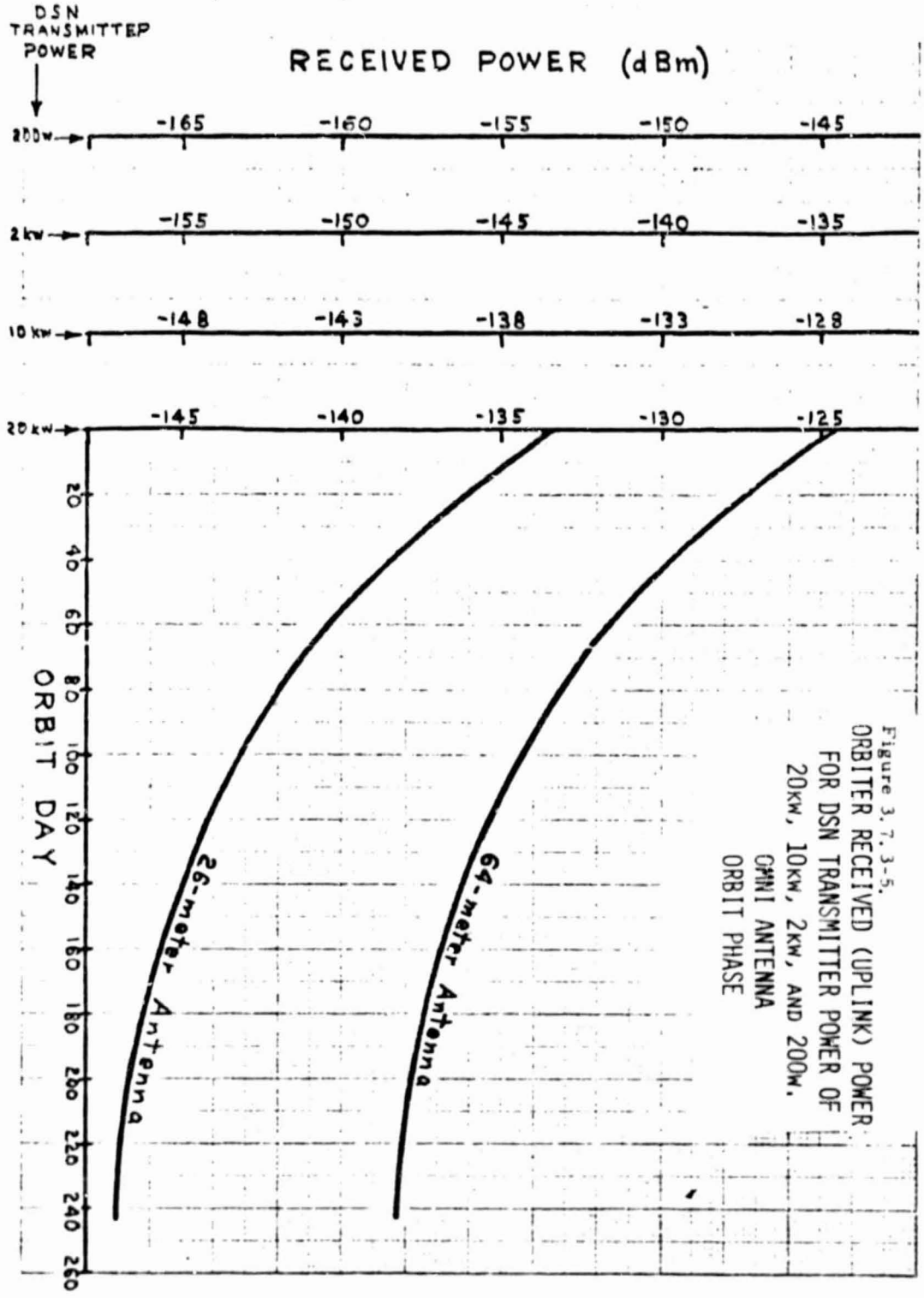


Figure 3.7.3-5.
ORBITER RECEIVED (UPLINK) POWER FOR DSN TRANSMITTER POWER OF 20kw, 10kw, 2kw, AND 200w.
OMNI ANTENNA
ORBIT PHASE

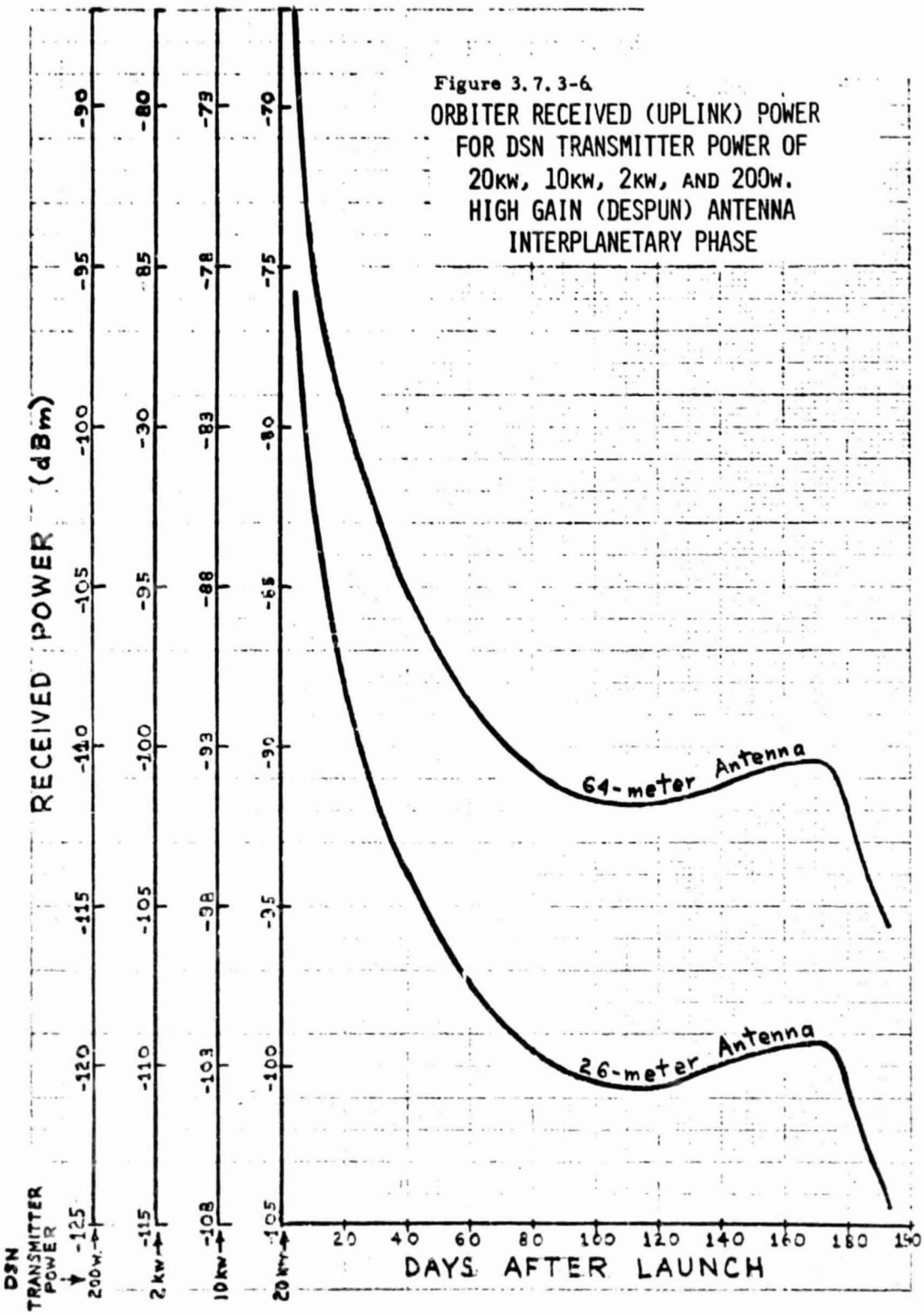


Figure 3.7.3-6
ORBITER RECEIVED (UPLINK) POWER FOR DSN TRANSMITTER POWER OF 20KW, 10KW, 2KW, AND 200W. HIGH GAIN (DESPUN) ANTENNA INTERPLANETARY PHASE

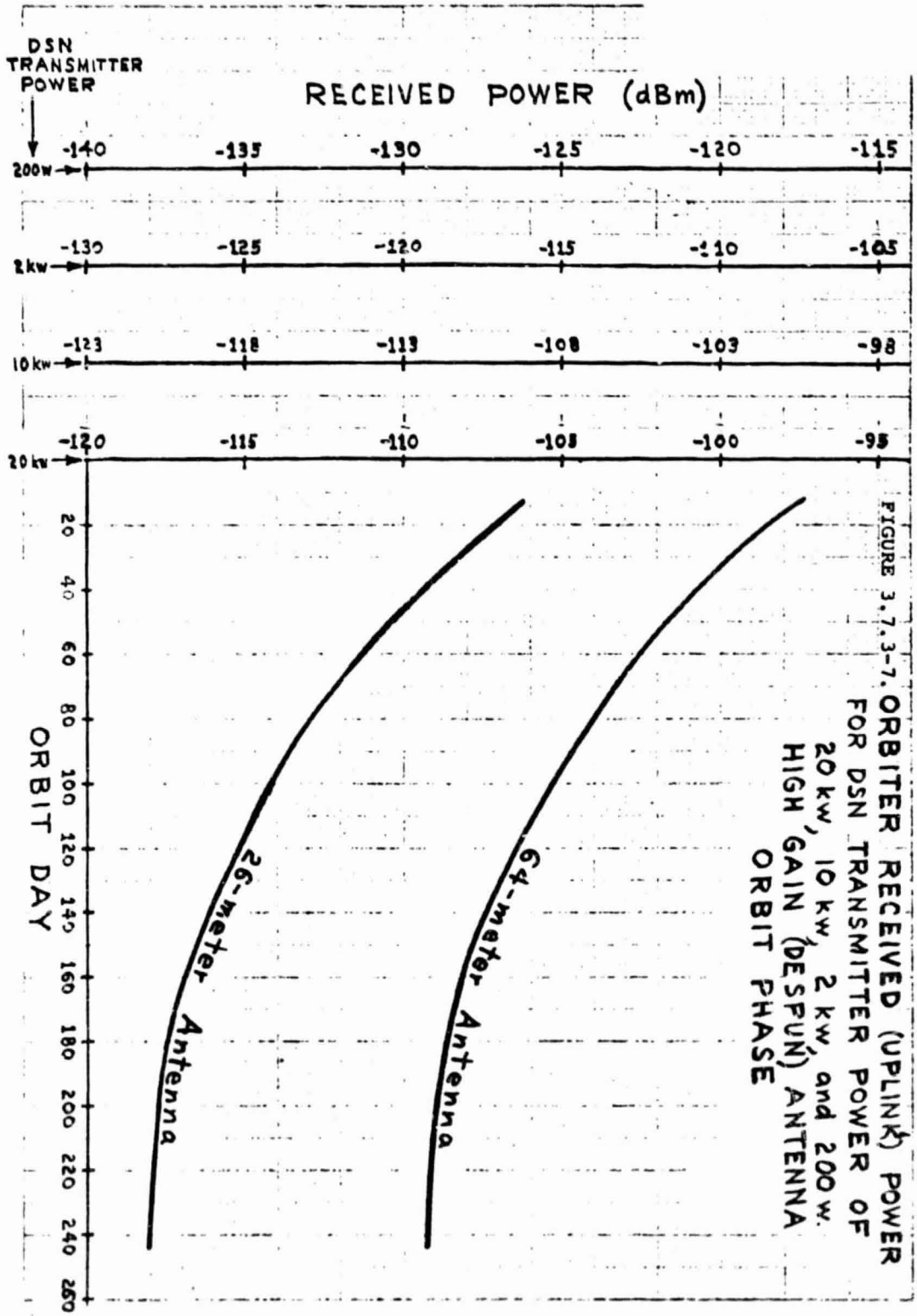


FIGURE 3.7.3-7. ORBITER RECEIVED (UPLINK) POWER FOR DSN TRANSMITTER POWER OF 20 kw, 10 kw, 2 kw, and 200 w. HIGH GAIN (DESPUN) ANTENNA ORBIT PHASE

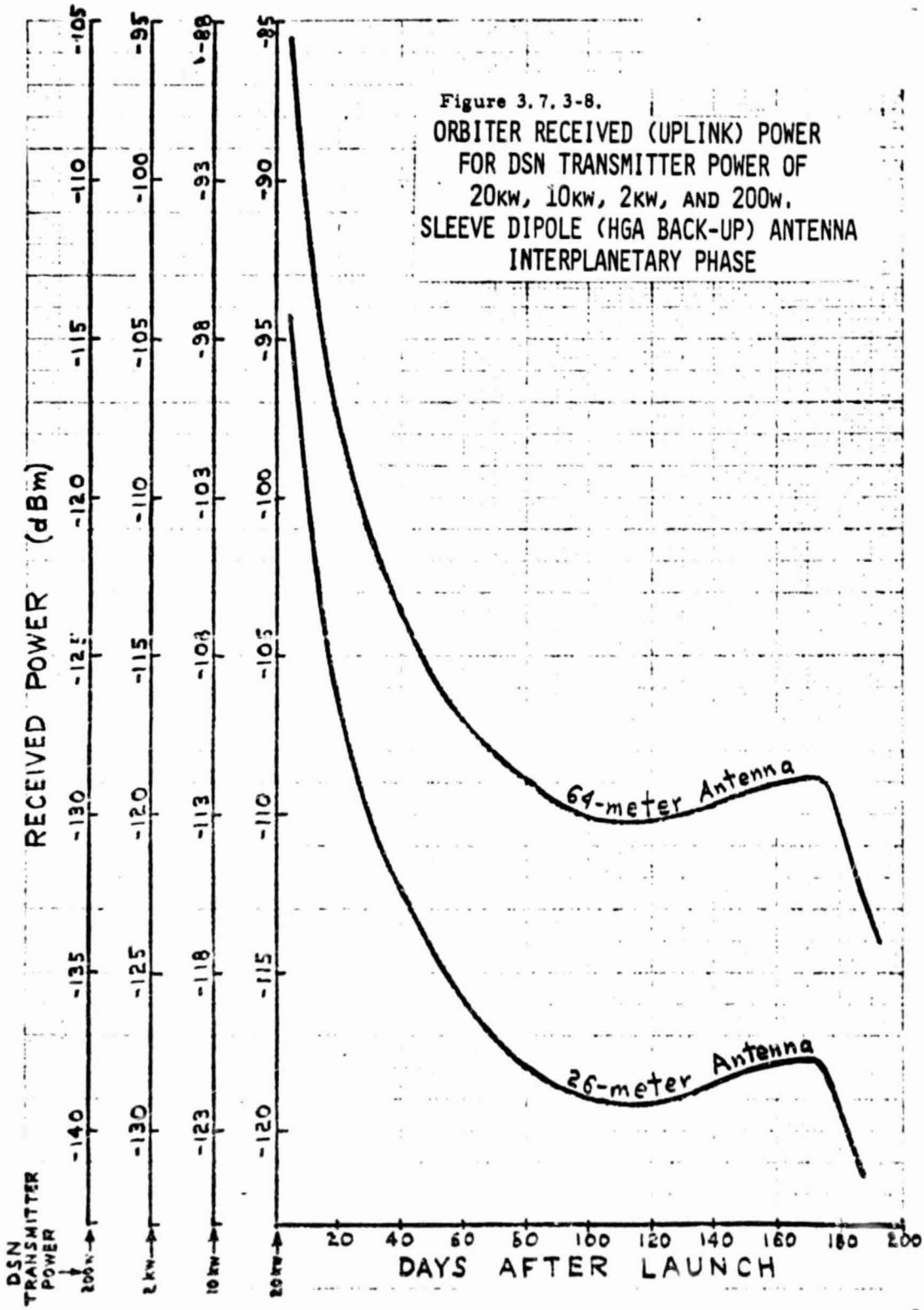


Figure 3.7.3-8.
ORBITER RECEIVED (UPLINK) POWER FOR DSN TRANSMITTER POWER OF 20KW, 10KW, 2KW, AND 200W. SLEEVE DIPOLE (HGA BACK-UP) ANTENNA INTERPLANETARY PHASE

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OF POOR QUALITY

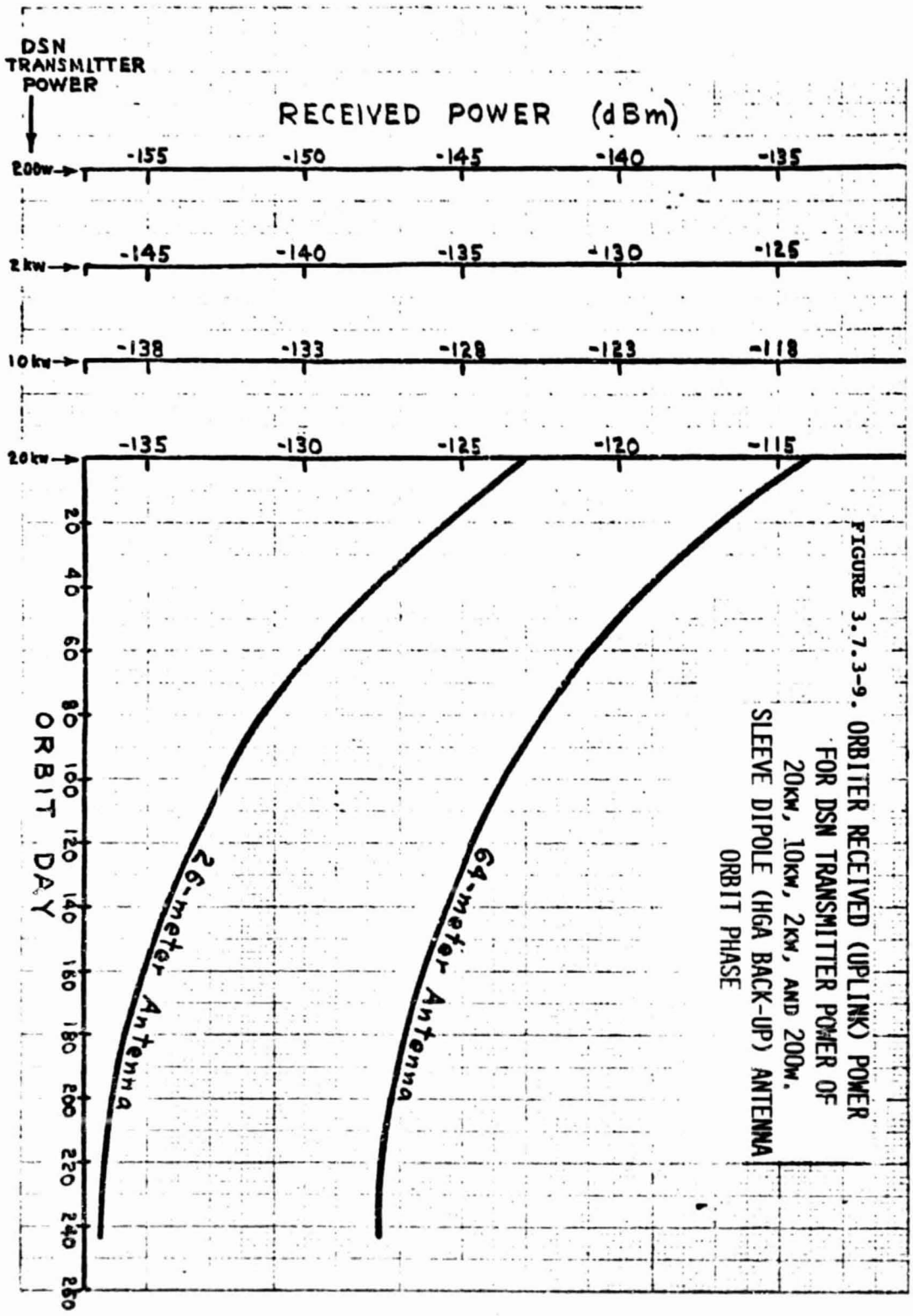


FIGURE 3.7.3-9. ORBITER RECEIVED (UPLINK) POWER FOR DSN TRANSMITTER POWER OF 20KW, 10KW, 2KW, AND 200W. SLEEVE DIPOLE (HGA BACK-UP) ANTENNA ORBIT PHASE

3.8 POWER SUBSYSTEM

3.8.1 Power Subsystem Description. The Orbiter power subsystem provides semi-regulated power at 28 volts ±10% to all spacecraft and science loads. The individual users condition this power, where necessary, by additional regulation, power switching, and dc/dc converters that develop the specific voltages required by that user.

3.8.1.1 Functional Description. Figure 3.8.1.1-1 (Appendix C) shows a functional block diagram of the Orbiter power subsystem. Power is provided to all loads on 4 different power buses:

(a) The essential bus - which cannot be commanded OFF,
(b) The science bus - dedicated exclusively to the science instruments,
(c) The switched loads bus, and
(d) The RF transmitter bus.

The primary power source is the solar panel. The Orbiter panel has a total of 15,988 solar cells connected in a series parallel arrangement that optimizes power output and minimizes stray magnetic fields, ripple, and shadowing effects of the thrusters, star sensor sun shield and magnetometer boom.

During most mission phases there will be a surplus of solar panel power available to meet the load requirements. A power surplus always forces the solar panel voltage upward above the nominal 28 volt value. To prevent excessive power bus voltages, the bus limiters sense the bus voltage and shunt off surplus solar panel power to load resistors mounted on the rear sections of the solar array substrate and the equipment shelves. The bus limiters maintain the bus voltages below 30.0 volts at all times.

When the solar panel cannot provide adequate power for all spacecraft loads at low sun angles and during eclipses, the two nickel cadmium batteries come on line automatically through the discharge regulators. The regulators normally

maintain the bus voltage above 26.7 volts at all times. Battery energy is subsequently replenished through small charge arrays that boost the main array voltage to a voltage level that results in recharging of the batteries.

If a spacecraft overcurrent or undervoltage on either battery occurs, loads are removed by the undervoltage/overload (UV/OL) switch to protect the spacecraft from potential catastrophic failure. Those loads, however, that are deemed vital to the spacecraft survival are hardwired to the essential bus and cannot be switched off. These include all command subsystem units, propulsion heaters, and r.f. subsystem switch drivers.

If a fault occurs that raises the total spacecraft load current above 16.5 ±0.75 amperes, the other 3 power buses (science, switched load, HF transmitter buses) will sequentially trip off. If the spacecraft loads decrease below 16.5 amperes, the tripping sequence is terminated. If a fault occurs that lowers the battery voltage to 27.55 ±0.15 volts, all 3 buses will be switched off. When the battery voltage recovers to 28.3 volts, a UV/OL reset relay command can be sent to power up all three non-essential buses again.

The UV/OL switch has an override mode that inhibits load removal. This feature can be used if the trip circuitry fails or if it becomes necessary to operate selected spacecraft loads even under fault conditions.

Excitation for the pyro bus is derived from a battery tap located 16 cells (of a total of 24) from the ground reference level. This tap is also used to provide large fault clearing currents to the power bus in the event of a heavy fault current that saturates the discharge controllers.

A total of eight current sensors are used to monitor power subsystem performance. Each sensor consists of a 50 millivolt resistive shunt and a

hybrid differential amplifier with a gain of 102.4 that develops a 5.12 volt output signal at full scale. Four of these sensors are integrated into the charge/discharge controllers - they measure battery charge and discharge currents. The spacecraft loads current sensor is integrated into the UV/OL switch because it not only provides a telemetry signal but also an overcurrent trip signal to the UV/OL turnoff sequence generator. The other 3 current sensors shown in Figure 3.8.1.1-1 are packaged as individual units.

The charge controller provides a high rate charge by connecting the boost charge string directly to its associated battery and a low-rate charge by inserting a 47 ohm resistor between the charge string and the battery. Each battery pack has a thermal switch that closes at 95 ±5°F and automatically turns off battery charging. After the battery temperature drops and the thermal switch opens again, battery charging is resumed automatically.

Near Earth low solar heat input conditions or a UV/OL trip can reduce the equipment shelf temperature to dangerously low levels. For this reason two of the seven bus limiters are connected to heater resistors mounted on the back side of the equipment shelves. To guarantee that the first 130 watts of excess solar panel power is shunted into these shelves the operating set voltage of each of these two limiters (#2 and #6) is lowered by closure of thermal switches when the shelf temperature drops to 35 ±5°F.

The power interface unit (PIU) provides power switching for the propulsion heaters. Fuses for the propulsion heaters and science instruments are also located in the PIU.

3.8.1.2 Equipment Summary and Location. The complete Orbiter power subsystem consists of the following units:

- Solar panel (with ten 26.0 ohm load resistor circuits bonded to rear of substrate)

- Four 12-cell battery packs (2 per battery)
- Two charge/discharge controllers
- Four 29.5 volt (-1) bus limiters
- Three 30.0 volt (-2) bus limiters
- One undervoltage/overload switch
- One power interface unit
- Three current sensors, three types: 0-4.5, 0-12, 0-18 amperes (Five additional current sensors are integrated into the charge/discharge controllers and the UV/OL switch).
- Four shelf thermal switches
- Four 26.0 ohm equipment shelf load resistor circuits.

The locations of the shelf mounted units are shown in Figures 3.8.1.2-1 and 3.8.1.2-2.

3.8.2 Units Descriptions. Described below are the individual power subsystem units.

3.8.2.1 Solar Panel. The Orbiter solar panel is designed to supply the power requirements of the loads under varying conditions of solar intensity, temperature and sun angle. A total of 15,988 two x two cm solar cells are used on the Orbiter panel. A thin 8 mil, 20 ohm-cm lightweight cell was selected to minimize panel weight. Each cell has a 6 mil microsheet coverglass to minimize radiation degradation.

The basic layout of the Orbiter panel solar cells are "3 x 85" groups (three cells in a parallel set, 85 such sets in series). There are a total of 52 of these groups (13,260 cells). There is a sufficient number of these "3 x 85" cell groups to meet the power load requirements of Venus orbital operations. However, they do not provide sufficient power to meet the Orbiter cruise requirements during minimum solar intensity conditions. Thirty-six additional parallel cells had to be added to meet estimated requirements 48 days after launch of the Orbiter. Since the additional cells are needed only when the panel is relatively cold, the number of series cells for these groups was reduced from 85 to 62, resulting in a total increase of 36 x 62 = 2232

cells (thirty-six "1 x 62" cell groups). An additional 96 cells were added to compensate for magnetometer boom shadowing when the spin axis is more than 5° from the sun normal. This results in a total of 15,588 cells, exclusive of battery charge strings.

The Orbiter panel has eight individual "1 x 25" battery charge strings for each battery (boost charge array). They are all connected in parallel and spaced approximately 45 degrees apart on the cylindrical substrate. This small number of solar cells (200 per battery) easily provides the required minimum charge rate of 0.5 ampere near Venus.

The Orbiter panel is also designed to withstand wide temperature extremes. During the long 3.8 hour apoapsis eclipse the Orbiter panel temperature will drop to -165°C and during periapsis passes will rise to peak temperatures between +90 and +130°C. Panel temperature is telemetered by PPAN2T, located at θ = 263°, approximately 22 inches from the top of the panel and PPAN1T located at θ = 263°, approximately 9 inches from the bottom of the panel.

The Orbiter panel is designed to survive a massive solar flare. Solar panel degradation computations have been made assuming that this flare can occur at any time. At any point in the mission the equivalent fluence of this solar proton event is scaled according to the inverse of the square of the distance from the sun. If this solar flare occurs when the Orbiter is near Venus, a permanent power loss of approximately 13% will result, see Figure 3.8.3.3-1.

Many of the Orbiter science instruments require that all exposed solar cell terminals and wiring be covered with a non-conductive material to minimize plasma charging of the spacecraft structure. To meet this requirement, the void area between and surrounding each solar cell is filled with RTV 566 adhesive. This encapsulation or grouting process results in a small loss in solar panel power due to blockage of light to the

solar cell active area and is accommodated in solar panel performance predictions.

3.8.2.2 Nickel Cadmium Batteries. Each battery consists of two 12-cell packs. A pack assembly sketch, which illustrates the method of construction, is shown in Figure 3.8.2.2-1. Each pack contains 12 individually insulated 7.5 AH nickel cadmium cells, 11 extruded thermal shunts and two end plates, all held together by two tension bars torqued to compressively load the cells.

The batteries will be fully charged prior to launch at a C/10 (0.75 ampere) rate through the umbilical connector from the blockhouse. The batteries can be charged independently with the battery enable plugs removed and the spacecraft completely powered down. Charging can also continue through the umbilical after the plugs are installed, if desired, until liftoff.

The batteries are charged during the mission with a small boost charge solar array. In order to fully charge the batteries, their voltage must be raised above the nominal 28 volt bus voltage. At a C/10 rate each cell voltage must rise to approximately 1.45 volts/cell or 24 x 1.45 = 34.8 volts for the complete battery. At C/50 (0.150 amperes) each cell voltage must rise to approximately 1.413 volts/cell or 24 x 1.416 = 33.9 volts for the complete battery. This voltage boost is obtained by means of a small array that boosts up the main array voltage to the desired value.

Each of the four packs has a temperature sensor located near the middle of the pack. Each pack also has a thermal switch that turns off battery charging if the pack temperature reaches 95 ±5°F.

Maximum battery depth of discharge (DOD) will be 60% during the apoapsis eclipse season. Figure 3.8.2.2-2 describes battery performance as a function of energy (A-H) removed at 75°F; 4.5 AH represents a DOD of 60% of rated capacity. For a battery load of C/2 or 3.75 amperes, the battery voltage stays above 29.5 volts or 1.23 volts per

cell at the 60% DOD level. At this relatively high battery voltage the discharge controllers will stay in regulation at output voltages exceeding 27.4 volts. This data indicates that with a relatively high discharge rate the batteries will provide sufficient voltage to maintain the regulator output above 27.4 volts, even at 80% DOD.

Although the specified maximum DOD, as quoted above, is 60% there would be very little risk to the batteries if this value were increased to 70% in order to meet apoapsis eclipse science objectives. The apoapsis eclipse season lasts only 6 days and the nickel cadmium batteries are capable of six charge/discharge cycles at this DOD level.

The data shown in Figure 3.8.2.2-2 is for a new battery. Mission simulation data indicates that there will be a small amount of voltage fade due to time aging and charge/discharge cycling during periapsis eclipse. However, the battery current load during apoapsis eclipse will be only 1.35 amperes. Therefore, the discharge characteristics shown in Figure 3.8.2.2-2 are very close to those that will be obtained during apoapsis eclipse.

The nickel cadmium batteries must each provide high peak currents (up to 30 amperes) for 35 milliseconds to actuate pyrotechnic devices. All Orbiter pyrotechnic events occur when the battery is fully charged and the full battery voltage during pyro firings will not drop below 30 volts. This value is well above the set point of the discharge controllers and the bus voltages will not be affected by the pyro firings.

3.8.2.3 Bus Voltage Limiter Electronics. Voltage limiters are used to dissipate excess panel capacity in load resistors in order to maintain the bus voltage below 30.0 volts. The limiters are divided into two groups with different set points to prevent simultaneous operation of all limiters at their maximum internal power levels. The low set point limiters (-1) begin to conduct

after the bus reaches 29.5 volts and achieve full conduction or saturation before the bus voltage becomes 29.6 volts. When the bus reaches 30.0 volts the upper set point limiters (-2) start to operate in a similar manner. When the bus voltage reaches 30.1 volts all of the bus limiters are in full conduction.

A simplified block diagram of a bus limiter is shown in Figure 3.8.1.1-1. Each of the seven bus limiters can be enabled or disabled by command. Each limiter has controlling circuitry that senses the bus voltage and applies drive current to two power transistors which shunt current through load resistors to ground. The two power transistors turn on sequentially, as the bus voltage rises, in order to minimize the maximum thermal dissipation in each limiter. The offset diode forces Q2 (and the B heaters) to go into full conduction before Q1 (and the A heaters) starts to conduct. The dissipation in each power transistor is very low after saturation is reached because the saturation voltage drop is less than 0.15 volts.

The upper set point limiters have two additional terminals. Their nominal 30.0 volt set points can be lowered to either 28.5 (limiter 6) or 29.0 (limiter 2) volts by the closing of shelf thermostatic switches.

Seven voltage limiters are provided on the Orbiter to prevent excessive bus voltages. Four limiters (1, 3, 5, 7) start to conduct at 29.5 volts and dissipate surplus panel capacity in the solar panel substrate. Three voltage limiters (2, 4, 6) start to conduct when the bus voltage reaches 30.0 volts. Limiter 4 is connected to load resistors mounted in the solar panel substrate. Limiters 2 and 6 are connected to load resistors bonded to the rear of the equipment shelves.

Limiter 6 on the battery shelf is connected to battery shelf heater resistors. As explained above, its conduction voltage can be lowered by a pair of thermostatic switches. The battery shelf

switches are located at an angular location of approximately θ = 280°. Closure of the switches at 35 ±5°F lowers the set point of limiter 6 to 28.5 volts. This places its set point below all the other limiters and forces the first 66 watts of available solar panel power into the battery shelf load resistors. The switches open when the shelf temperature reaches 55 ±5°F.

Limiter 2 on the r.f. shelf is also connected to shelf heaters and thermostatic switches. The latter are located near Power Amplifier 3 and 4 (θ = 210°). These switches close when their temperature reaches 35 ±5°F. Closure of the switches lowers the set point of limiter 2 to 29.0 volts and forces 66 watts into the r.f. shelf when solar panel power is available.

3.8.2.4 Bus Limiter Load Resistors. The seven bus limiters each require two 26 ohm load resistors for a total of 14 load circuits. Ten of these are bonded to the rear of the solar array substrate and the other four are bonded to the bottom of the equipment shelves.

Each solar array circuit consists of five parallel 130 ohm sections and each 130 ohm section, in turn, consists of two 65 ohm resistors in series. Each of these ten 65 ohm sections are approximately 1.80 inches high and 26.5 inches wide and cover a complete 1.8 inch band around the entire inside perimeter of the substrate. The wire is non-magnetic and twisted to provide magnetic field cancellation.

Each shelf load circuit consists of four 6.5 ohm sections connected in series. Each load resistor section is 0.8 x 2.3 inches in size.

3.8.2.5 Charge/Discharge Controllers. Figure 3.8.1.1-1 shows a block diagram of the charge/discharge controller associated with each battery. The charge controller section connects a boost charge array in series with the battery through switching relays K1 and K2. High charge rate is selected by commanding both relays to their upper positions (BAT19 or BATA9; and BAT13 or BATA3 for

battery 1. For battery 2: BAT29 or BATB9; and BAT23 or BATB3), which results in a direct connection between the charge array and the battery. Low charge rate is selected by commanding both relays to their lower positions (BAT1Ø or BATAØ; and BAT14 or BATA4 for battery 1. For battery 2: BAT2Ø or BATBØ; and BAT24 or BATB4). This places a 47 ohm resistor between the battery and charge array and forces the operating point of the latter below the knee (constant voltage portion) of its voltage-current characteristics. Battery charging is turned off whenever the two relays are in opposite positions.

The charge rate of one battery is selectable by ground command independent of the charge rate selected for the other battery. An overtemperature trip circuit is included to turn off battery charging if the temperature of either of its associated battery packs reaches 95 ±5°F. The battery thermal switches are connected in a parallel and closure of either switch results in a trip command pulse to the K1 relay driver. If relay K2 is in the high charge rate or upper position, relay K1 will be transferred to the lower position. If relay K2 is in the low-charge rate or lower position, an overtemperature trip signal will drive relay K1 into the upper position. In both cases charging will turn off. When the battery pack temperature decreases by approximately 10°F the thermostatic switches will open again and an onboard automatic reset command pulse will return relay K1 to its previous position and charging will be resumed. Charging can also be restored after a trip by ground command to relay K1.

When the essential bus voltage is maintained above 26 volts, the discharge controllers are in a stand-by mode. When the solar panel cannot provide sufficient current to power the spacecraft loads by itself, the bus voltage will decrease. Either the primary or redundant discharge controller (whichever had been selected by ground command) comes on line automatically at 27.80 ±0.005 volts. The 0.1 ohm resistors in

series with each regulator force load current sharing between the two batteries. These resistors cause the bus voltage to decrease as a function of load current although the regulator output voltage is maintained at 27.80 ±0.065 volts by the regulator feedback loop.

The maximum eclipse load will be 5.5 amperes per regulator, resulting in an output voltage downstream of the 0.1 ohm resistor of 27.80 - 0.55 = 27.27 volts. At this load current the minimum required drop across the regulator is approximately 1.2 volts. The output voltage will continue to be regulated as long as the battery voltage remains above 27.25 + 1.2 = 28.25 volts or 1.185 volts per cell. If the batteries discharge beyond this point the regulator will saturate and the output voltage will decrease in accordance with the decrease in battery voltage. When the battery voltage reaches the UV/OL trip point of 27.55 volts or 1.148 volts/cell the regulator output voltage will drop to 27.55 - 1.20 = 26.35 volts.

If the primary discharge regulator of either controller fails, the redundant regulator associated with that controller can be switched in to replace it. Should a battery fail completely, the remaining battery and its controller can support the entire load requirement within the ampere-hour energy limitations of the one remaining battery. Each discharge regulator is capable of providing a maximum of 11 amperes continuously. In the example cited above, a single regulator would regulate at 27.80 - 1.10 = 26.7 volts at 11 amperes until the battery reached 26.70 plus 2 volts regulator drop for a total of 28.70 volts (1.196 volts/cell).

Two current sensors are incorporated into each controller to measure battery charge and discharge current (PCHG1I and PDIS1I respectively for battery 1; PCHG2I and PDIS2I respectively for battery 2).

3.8.2.6 Power Interface Unit. The Power Interface Unit (PIU) provides centralized switching and fuse protection for the propulsion heaters and science instruments.

Magnetic latching relays are used where load switching is required. Each latching relay is driven by its own hybrid relay driver circuit. In those cases where flight critical functions are controlled, redundant relays are connected together to ensure reliable switching. The status of all relays is monitored with bilevel telemetry signals.

All fuses are connected in a parallel redundant configuration. The series resistor in the redundant leg is sized large enough to force approximately 90% of the current through the primary fuse, yet small enough to ensure a minimum bus voltage of 25.2 volts to the loads, if the primary fuse fails.

All spacecraft heaters are on the essential bus and are fused in the PIU. The four small radial jet and tank heaters are on continuously; they cannot be commanded OFF. The tank heaters are redundant. A latching relay selects either the primary or secondary tank heater group by ground command (HTT19 or HTTA9; HTT29 or HTTB9). No single point failure will prevent one of the two heater groups from being commanded ON.

The OIM heater arrangement has series/parallel redundancy. This guarantees that no single point failure can prevent one of the two heaters from being commanded ON (HTM19 or HTMA9; HTM29 or HTMB9), and that both heaters can always be commanded OFF (HTM1Ø or HTMAØ; HTM2Ø or HTMBØ).

Switching of the two aft axial heaters is parallel redundant. No single point failure will prevent one of the two heaters from being commanded ON (HTJ29 or HTJB9).

The Orbiter PIU has a total of 12 science instrument fuse pairs and 11 propulsion heater fuse pairs.

3.8.2.7 Undervoltage/Overload Switch. The undervoltage/overload switch unit (UV/OL) provides power to the four spacecraft load buses. A 0-18 ampere, 50 mv resistive shunt is externally mounted on the UV/OL unit. It measures the load current (PBUSLI - Spacecraft Loads Current) provided to all four power buses. Three of these buses pass through magnetic latching relay contacts which are tripped open if the spacecraft load current is excessively high or if either battery terminal voltage is too low.

The essential bus loads cannot be switched off because they are vital to the survival of the spacecraft. The other loads can be tripped off if either battery terminal voltage drops to 27.55 ±0.15 volts or the spacecraft load current exceeds 16.5 ±0.75 amperes.

A trip signal is generated to turn off science loads 75 milliseconds after a fault condition occurs (refer to Table 3.8.2.7-1). If removal of the science loads reduces spacecraft loads to a value less than 16.5 amperes within 50 milliseconds, the tripping sequence is inhibited. If spacecraft loads are still above 16.5 amperes, another trip signal is generated 125 milliseconds after the fault occurs. This will trip off the switched loads bus also. If the current still does not recover after trip of the switched loads bus, the RF transmitter bus is also tripped off by a signal generated 175 milliseconds after the fault occurs.

The same trip signals described above are also generated when the voltage of either battery decreases to 27.55 ±0.15 volts. When the battery voltage rises above 28.3 volts after an undervoltage trip, as a result of a transition from battery discharge status to battery charge status, the tripping sequence is inhibited. However, it normally requires several seconds to recover to this voltage. Since the 3 trip signals are generated within 100 milliseconds of each other, a battery undervoltage trip will always switch off all 3 non-essential power buses. After the battery voltage rises to 28.3

TABLE 3.8.2.7-1

RECOGNITION OF UV/OL TRIP

Time of occurance after fault appears	Subsystem Response	Ground Recognition	
		TM mnemonic	TM & Response
75 msec.	Science Bus Relay to off; Power to science loads is cutoff, science loads reset to off.	PSCBUS	Science Bus Relay Status = 0 (OFF)
		PSCICI	Science current decreases to zero
		PBUSLI	S/C loads current decreases by same amount as PSCICI
		PLIMTI	Bus Voltage Limiter current likely increases by same as PSCICI decreased
125 msec.	Switched Loads Bus relay to off; all switched loads reset to off except the star sensor		Loss of downlink telemetry data
175 msec.	Xmtr Bus Relay to off; all RF Xmtr Bus loads reset to off, except Xponder-exciter		Loss of downlink carrier

volts, a "UV/OL relay reset" ground command (PSP11 or PSPA1) will reset all 3 buses.

The three relays, K1, K2 and K3 which provide the bus protection described above, are all reset by the "UV/OL relay reset" command. Relay K4 provides an override capability that can be utilized, at the discretion of ground control, to bypass the trip function and remove the protection feature (PSP1Ø or PSPAØ).

Once the "UV/OL relay reset" command is sent, only the science bus can be powered down again by ground command (INS1Ø or INSAØ) unless an overcurrent or undervoltage condition occurs.

It is possible to power up temporarily after a trip, in order to make a quick spacecraft telemetry assessment, and then power down again in the following manner. First, the override (not protected) feature via relay K4 must be enabled (PSP1Ø or PSPAØ). Then spacecraft loads are turned on, a few major frames of TM data is recorded, and then K4 is disabled again (PSP19 or PSPA9). The override feature is also used if a failure occurs in the trip circuitry (trips when it should not), or if any one of the two batteries fails.

A capacitor precharge circuit is incorporated into the UV/OL switch. The purpose of this circuit is to limit the magnitude of current inrush to the input capacitors of all the loads on the science and switched loads buses through two 390 ohm series resistors. This prevents large switching voltage transients when these buses are connected to the essential bus through relays K1 and K2.

Two reverse connected diodes are wired across each of the three switchable buses. They prevent potentially dangerous negative bus voltages which could be generated by turning off loads with large inductive elements in their input filters.

3.8.2.8 Engineering Instrumentation. The power subsystem telemetry signals are explained in detail in Appendix A.

The temperature sensors on the solar panel are precision wide range platinum resistors. Thermistors are used on the battery packs.

The current sensor uses a dual 50 mv current shunt and a hybrid differential amplifier with a gain of 102.4 to develop a full scale telemetry signal of 5.12 volts (0.05 x 102.4 = 5.12). The current sensor is connected in series with the positive buses. It is powered directly from the bus where the measurement is made.

Four of the current sensors are built into the charge/discharge controllers. One of the current sensors is an integral part of the UV/OL unit. The other three sensors are packaged as individual units.

3.8.3 Operations Description. This section describes how to power manage the spacecraft. It explains how to configure the power subsystem properly by ground command and how to assess its performance from the available telemetry signals.

3.8.3.1 Power Bus Voltage Operating Ranges. Figure 3.8.3.1-1 describes the power bus operating voltage ranges during various mission operating modes. The nominal power bus voltage is 28.0 volts. During those operational phases when the solar panel is fully illuminated, the actual bus voltage will be higher than this value. During eclipses, when the discharge controllers are active, the bus voltage will be under 28.0 volts.

The solar panel is sized to provide power somewhat in excess of that which is required by the loads and the bus voltage is driven upward until clamped by the bus limiters. Bus limiters 1, 3, 5 and 7 have set points at 29.5 volts and bus limiters 2, 4, and 6 have set points at 30.0 volts. During most mission phases there is a surplus of solar panel power that ranges between 0.25 and 6.0 amperes. Bus limiters 1, 3, 5 and 7

together are capable of absorbing over 9 amperes of excess panel capacity. Therefore, the other group of limiters will not become active unless a large number of spacecraft loads are turned off or the shelves are cold.

For the reasons stated above, the bus voltage will be near 29.5 volts during most of the mission. It will rise up to 30.0 volts only for a short period of time during exit from apoapsis eclipse when excess panel capacity will be near 13 amperes. An undervoltage/overload switch unit trip could also activate the 30.0 volt bus limiters.

Bus limiters 2 and 6 are connected to load heater resistors bonded to the rear of the equipment shelves. If the thermostatic switches associated with bus limiter 6 are activated due to a cold shelf, its set point will be lowered to 28.5 volts and the first excess 66 watts of panel capacity will be forced into the battery shelf heaters. If the thermostatic switches associated with bus limiter 2 are activated due to a cold r.f. shelf, its set point will be lowered to 29.0 volts and the next incremental excess 66 watts of panel capacity will be forced into the r.f. shelf heaters.

When the spacecraft passes through a penumbra region before entering full eclipse, the current provided by the solar panel slowly decreases and the power bus voltage also decreases. When the bus voltage drops to 27.80 ±0.065 volts the discharge controller comes on automatically to help share the load. The discharge controller output is regulated very close to 27.80 volts even at high loads. There are 0.1 ohm resistors connected between each regulator output and the power bus to force accurate load share between the two regulators. Since the highest eclipse load will be 5.5 amperes (during periapsis) the lowest bus voltage during eclipse will be 27.80 - (0.1) (5.5) = 27.25 volts. The lightest load will be during apoapsis eclipse - approximately 1.35 amperes/battery which results in a 27.65 volt power bus.

The normal bus voltage during eclipses, therefore, will range between 27.25 and 27.65 volts at the discharge regulator connector. Harness drops will reduce this voltage somewhat but the lowest voltage at any load will not be lower than 26.8 volts. Load voltages will drop by another 0.6 volt if one of the batteries fails. Bus voltages will drop to the minimum allowable value of 25.2 volts only if battery DOD exceeds 80%.

3.8.3.2 Energy Balance Maintenance. The spacecraft must be powered managed to stay in a suitable energy balance situation at all times. During most of the mission, the spacecraft will be in full illumination and the sun line will be near 90 degrees with respect to the spin axis. During this time, the spacecraft loads must not exceed the solar panel capability for any appreciable period of time. The batteries can supplement the solar panel during short transition periods, but a continuous negative energy situation cannot be tolerated.

The spacecraft energy balance condition is best assessed with the current sensors. A simplified diagram of the power sources, loads and current sensor measurements is shown in Figure 3.8.3.2-1. All power sources are shown on the left and all power loads are on the right.

The current provided by the main array for spacecraft load use is measured by the solar panel current sensor. Battery charge current is provided by the main array, plus the boost charge arrays, and is measured by the two battery charge current sensors. If battery charging is turned off or reduced by switching from high rate to low rate, the solar panel current sensor telemetry signal will increase by the same amount as the decrease in charge current. The solar panel current that is utilized by the loads, plus the current supplied by the batteries when they ar in a discharge state, is measured by the spacecraft load current sensor. The excess current that is not needed is shunted off by the bus limiters and is measured in the bus limiter current sensor.

Total Main Array current is the sum of $I_{SP} + I_{CH1} + I_{CH2}$ (refer to Figure 3.8.3.2-1) Solar Panel Current (I_{SP}) measures solar panel current supplied to the spacecraft loads. It measures total main array current only if battery charging is commanded off.

Full illumination conditions can be defined by a current equation that shows the power sources on the left and the loads on the right: $I_{SP} + I_{CH1} + I_{CH2} = I_{SCL} + I_{BL}$ (Batteries not discharging)

where

I_{SP} = solar panel current

I_{SCL} = spacecraft loads current

I_{BL} = bus limiter current

I_{CH1} and I_{CH2} = telemetered battery 1 and battery 2 charge currents

During eclipses, all spacecraft loads are supported by the batteries through the discharge controllers. The battery currents are measured by the two battery discharge current sensors. The spacecraft load current is again measured by the spacecraft load current sensor. However, the small shunt current loss in each discharge controller is not measured directly. Figure 3.8.3.2-2 describes this shunt current loss as a function of load current. The bus limiters, of course, are not active during eclipse. Each bus limiter draws only 5 milliamperes of standby current. Since the bus limiter current sensor full scale range is 0 to 12 amperes, or 47 milliamperes per telemetry bit, the telemetry signal will be very small. The eclipse situation can be summarized by the current equation:

$$I_{BD1} + I_{BD2} = I_{SCL} + I_{BL} + I_{SH1} + I_{SH2}$$

where

I_{BD1} and I_{BD2} = telemetered battery 1 and battery 2 discharge currents

I_{SCL} = spacecraft load current

I_{BL} = bus limiter current (approximately 35 ma)

I_{SH1} and I_{SH2} = discharge controller shunt currents as determined from Figure 3.8.3.2-2

There will be short time periods when the solar panel and batteries must share the load. This includes launch, entering and leaving eclipse, and some periapsis passes. When entering eclipse, the available solar panel current decreases and the bus limiter current also decreases by a corresponding amount and then becomes 5 milliamperes/limiter standby current. After the bus voltage drops to 27.80 volts, the discharge controllers become active. The charge current will then by-pass the batteries and flow directly into the discharge controllers through the battery discharge current sensors. As the solar panel current continues to decrease, the batteries will then start to discharge. The current equation for this situation is somewhat more complex than the two preceding conditions. Once again, all power sources are on the left and the loads are on the right:

$$I_{SP} + I_{TBD1} + I_{TBD2} = I_{SCL} + I_{BL} + I_{SH1} + I_{SH2}$$

where

I_{SP} = solar panel current

I_{TBD1} = True battery 1 discharge current = $I_{TBD1} - I_{CH1}$

I_{TBD2} = True battery 2 discharge current = $I_{TBD2} - I_{CH2}$

I_{SCL} = spacecraft load current

I_{BL} = bus limiter current (approximately 35 ma)

I_{SH1} and I_{SH2} = discharge controller shunt current as determined from Figure 3.8.3.2-2

It should be noted that when loads are being shared by the solar panel and batteries, the true battery discharge currents are $I_{BD1} - I_{CH1}$ and $I_{BD2} - I_{CH2}$: During eclipse, of course, the charge current sensors read zero and I_{BD1} and I_{BD2} indicate the true battery discharge current directly.

The discharge controller shunt loss, shown in Figure 3.8.3.2-2 represents the shunt current required when the battery voltage is high enough to maintain the regulator in an active regulator mode. When the battery voltage drops to a voltage level where the series pass regulator transistor drop gets too low, the latter goes into a saturated non-regulating mode and the output voltage starts to decrease. In the saturated mode, the drive circuitry is attempting to maintain regulation by delivering its maximum drive current capability to the series pass transistor. This value is approximately 230 milliamperes. Therefore, the shunt loss of the regulator will sharply increase from the small values shown in Figure 3.8.3.2-2 to approximately 240 milliamperes when the controllers go out of regulation. Under normal operating conditions,

it is not anticipated that the battery voltage levels will drop down to levels that result in saturated mode operation. This would occur only under failure mode conditions.

3.8.3.3 Solar Panel Performance as Function of Solar Constant. In order to plan power operations, it is very useful to know what the available current will be at the nominal 28.0 volt bus voltage for all solar constants that will be encountered during the Orbiter mission. Figure 3.8.3.3-1 shows plots of solar panel current at 28 volts as a function of solar constant. The lower curve is a worst case predict that assumes a massive solar flare will occur. The upper curve is a maximum predict that assumes that no radiation damage occurs and that the solar cells will perform up to their maximum potential. These curves represent power output with the sun incidence normal to the spacecraft spin axis and include the effects of increasing solar panel temperature at higher solar constants.

3.8.3.4 Battery Usage and Charging. During most of the mission, the batteries will be on trickle charge. High rate charging will be needed only during that portion of the periapsis eclipse season when eclipse times exceed 10 minutes and during the apoapsis season. Table 4.3.5.4-1 contains recommended charging management for the Orbiter Mission.

Section 3.8.2.5 describes how the battery charger operates. The charge current available during the various mission times is a function of a number of factors. This can be most easily explained by the use of battery boost charge array voltage/current curves and load lines for various critical mission times (Figure 3.8.3.4-1).

During high rate charge the load line operating region is always above the knee or on the constant current portion of the curve. Therefore, the charge rate is a direct function of the solar constant. It will vary from 0.3 amp

at L+48 days to approximately 0.7 amperes near Venus.

During low rate charging the load lines intersect the curves beyond the knee of the curve. The charge rate will vary between 130 and 190 ma when the bus voltage is at 29.5 volts and from 100 to 160 ma at a bus voltage of 28.0 volts.

The trickle or low charge rates quoted above assume that the battery is fully charged or near full charge and has a terminal voltage of approximately 34 volts (1.416 volts/cell). When a partially discharged battery is initially placed on charge, its voltage will be somewhat lower by approximately 2 or 3 volts. This moves the abscissa intercept of the load lines to the left and increases the initial charge current to a value that is 40 to 60 ma higher. The charge rate will then slowly decrease as the battery reaches full charge. Charge current increases by approximately 20 ma for each 1 volt reduction in battery terminal voltage or 1 volt increase in power bus voltage.

Battery charging will be commanded to low rate charge at launch and remain in this configuration approximately 7 months until the periapsis eclipse season. In order to stay in energy balance, when the periapsis eclipses exceed 10 minutes, it will be necessary to go to high rate charging for a few hours every orbit. To prevent overheating the batteries, the amount of high rate charge must be time limited. Maximum battery depth of discharge (DOD) during the periapsis season will be approximately 30%.

During the long apoapsis eclipse season, maximum battery DOD will be approximately 60%. At this battery discharge level, it will require approximately 10 hours to recharge at high rate. If the battery temperature does not exceed 85°F under continuous high rate charge during the apoapsis eclipse season, the charge controller can be left in high rate charge during the entire short season.

Battery charging will normally never be turned off. As explained in Section 3.8.2.5 battery charging is turned off when relays K1 and K2 are in opposite positions. If it is necessary to turn off battery charging for some reason, the preferred relay states for charging turnoff are K1 high and K2 low. The reason is as follows: If, instead, relay K2 was left in the high state, and relay K1 was left in the low state, a battery overtemperature condition followed by a battery cool down would generate a reset signal to relay K1 which would transfer the latter to a high position and high rate charging would start automatically. If, on the other hand, charging is turned off in the preferred manner, a battery overtemperature condition followed by a cool down would result in a reset signal to K1 which would transfer the latter to a low position and low rate charging would start automatically. Although, it is extremely unlikely that turning off battery charging would be followed by actuation of the overtemperature switch, it is preferable in this case that low rate rather than high rate charging be turned on. In any case, if charging comes on automatically as a result of a reset signal, the overtemperature circuit will turn charging off again if the batteries get too hot. The overtemperature switch and its associated circuit would continue to automatically turn battery charging on and off and, thus, protect the batteries from excessive temperatures.

The solar panel charge strings are spaced approximately 45 degrees apart on the cylindrical substrate, as explained in Section 3.8.2.1. This results in approximately a 9 percent ripple of the current over the constant current portion of the V/I curve. At high charge rates this ripple current will be seen on the charge current telemetry signals. Low charge rate operation is performed below the knee of the V/I curve. Since the open circuit voltage of a solar cell is relatively insensitive to solar illumination levels, there will be a much lower charge current ripple during low rate charge.

To get an accurate measurement of charge current, approximately 10 measurements should be averaged.

3.8.3.5 Use of Precharge Circuit. As explained in 3.8.2.7, a precharge circuit has been incorporated into the Orbiter undervoltage/ overload unit to slowly precharge the input filter capacitors of the instruments on the science bus and the spacecraft loads on the switched loads bus.

The precharge circuit should be used as a first step in powering up the switched loads bus again after an undervoltage or overcurrent fault has resulted in a trip of this bus. All of the loads on the switched loads bus (with one exception) are turned on and off by electronic latch circuits. Only the star sensor is turned on and off with a mechanical latching relay. Therefore, if a UV/OL trip occurs, the star sensor load will still be on the switched load bus after it is reactivated. When the precharge circuit is turned on, the star sensor will draw a continuous current of approximately 25 milliamperes through the 390 ohm isolation resistor. This results in a switched loads bus voltage of only 16 to 18 volts. Fortunately, at this low voltage star sensor OFF command circuitry still functions. After sending a "Precharge on" command, both star sensor channels must be commanded off to allow the switched loads bus to be precharged to approximately 28 volts before the "UV/OL relay reset" or "power system protection OFF" commands are sent.

The science bus is also precharged by the circuit described above. During system test, however, it was determined that the instruments present a relatively low input capacitance and do not require precharging. Therefore, if the science bus is turned off by sending "All science OFF" command, it is not necessary to precharge the science bus before sending "UV/OL relays reset". On the Orbiter spacecraft, the precharge circuit is used only if the switched loads bus is tripped off. In that event, the command PRECHARGE ON should always precede the commands UV/OL RELAY

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RESET and/or POWER SYSTEM PROTECTION OFF. These should always be followed by the command PRECHARGE OFF.

3.8.3.6 Power Bus Fault Protection and UV/OL Trip Recovery. The spacecraft has a number of features that protect it from power bus faults. Potential fault conditions are overvoltage, undervoltage and overcurrent.

The bus limiters provide overvoltage protection by absorbing excess solar panel capability and restricting transient overshoot voltages to values less than 31 volts.

Each spacecraft load has protective fuses and most spacecraft units also have electronic current limiting to protect against overcurrent conditions. It is almost impossible to protect against harness shorts, but internal unit short circuits will be effectively isolated by the fuses and current limit circuits.

It is possible for the spacecraft to go out of energy balance if the spacecraft spin axis drifts off the sun line normal, if excessive loads are commanded on inadvertently by equipment faults or mission operations errors, or a spacecraft unit develops a fault current that is of insufficient magnitude to blow a fuse. The net result would be that batteries would start to discharge at a time when they normally should be on charge or, if problems occur during eclipse, the battery discharge current would exceed its nominal value. These operating conditions could deplete the battery and threaten the spacecraft with marginally low voltage conditions. To remedy these battery undervoltage and/or overcurrent conditions, the UV/OL switch will trip off science, switched load, and HF transmitter bus loads as explained in Section 3.8.2.7.

In most instances, an analysis of the telemetry data, prior to a UV/OL trip, will reveal the nature of the difficulty and thereby suggest corrective action. However, if a trip results in sudden loss of the downlink carrier and/or

digital telemetry signals with no clues to the nature of the problem, it will be necessary to make an assessment of the spacecraft operating conditions to formulate corrective action. In this case, it would be prudent to power up the switched loads and RF transmiter buses temporarily to get a few major frames of data and then power down again to the minimum essential bus loads while the data is being evaluated.

When it is desired to power up the buses temporarily after a trip, it should be accomplished by commanding to an override or "power system protection off" mode. Execution of the "UV/OL relay reset" command is undesirable in this case, since the switched load and RF transmitter reset relays, K2 and K3, can subsequently be powered down only by another undervoltage or overcurrent trip signal.

The command sequence to temporarily power up the spacecraft after a UV/OL trip is as follows:

- Precharge on (wait one second)
- PSI* off and PSI2* off (star sensor off commands)
- Power system protection off (override on)
- Precharge off
- Commands to restore RF downlink and telemetry
- Power system protection on (override off) to power down again.

After corrective action has been implemented to remove the fault, the power buses should be commanded back into the "protected" mode with relay K4 off and K1, K2 and K3 reset relay enabled by the "UV/OL relay reset" command.

Almost all non-essential spacecraft and science loads are turned on with electronic latch circuits. Removal of bus power to these loads also turns off these latches. After power is restored to their respective buses, it is necessary to send pulse turn-on commands to turn them on again. There are only two non-essential

power bus units that do not have electronic latch circuits. As explained above, one of them is the star sensor. The other is the transponder exciter. They are both turned on by mechanical latching relays. When power is restored to their respective power buses after a trip, they will turn on again. The star sensor can be turned off through the precharge circuit as explained above. The transponder exciter can be turned off (if it is necessary to do so) only after RF Xmtr bus power is restored. However, it does not interfere with the normal operation of the precharge circuits on the science and switched load buses.

A battery undervoltage trip will remove all 3 non-essential power buses even if the battery starts to charge after the Science and/or Switched Loads buses are tripped off. Battery voltage does not rise fast enough to interrupt the trip sequence. An overcurrent condition that does not reduce battery voltage below 27.55 volts will be interrupted when load removal drops total spacecraft loads current under 16.25 amperes; this could result in removal of 1, 2 or 3 non-essential power buses (Reference: Paragraph 1.5.18).

The all science OFF command will be ineffective whenever the spacecraft is operating with the power system protection OFF. Thus, in this configuration, instruments must be turned OFF by utilizing individual OFF commands (Reference: Paragraph 1.5.18).

3.8.3.7 Use of Bus Voltage Limiters. The seven bus limiters will be enabled prior to launch and they should stay enabled throughout the mission. A bus limiter should be disabled only if it is defective.

Six bus limiters are required to absorb the large excess panel capacity that is available immediately after the spacecraft has passed through a long apoapsis eclipse. Spacecraft loads are approximately 2.7 amperes at this time and the panel capability at 30.8 volts (maximum

allowable bus voltage) can be as high as 16.6 amperes for a few minutes as the panel heats up to its normal operational temperature near 158°C. Therefore, there will be an excess of 13.9 amperes that must be shunted off by the bus limiter loads at this time. Each bus limiter is capable of absorbing 2.35 amperes or 16.45 amperes total for the seven limiters. This provides a margin of 2.55 amperes. If one bus limiter fails, however, the margin is reduced to only 0.2 ampere. Therefore, in order to lighten the burden on the bus limiters, the batteries must be commanded into high charge rate before the spacecraft enters eclipse. This will add a load of approximately 1.4 amperes upon exit from eclipse and increase the bus limiter safety margin by an equal amount. This will bring the total spacecraft load up to 4.1 amperes. It will also prevent the bus from rising above the maximum safety level of 32.8 volts even if two bus limiters fail.

The above analysis pertains to the time when the solar panel output is maximum and loads are minimum. At all other times, the four (-1) bus limiters will be able to handle all excess capacity and maintain the bus voltage below 29.5 volts.

3.8.3.8 Propulsion Heater Performance Evaluation. The Orbiter has the following complement of jet thrusters: four radial, two aft axial and one forward axial. Each thruster and its associated line segments have heaters. All of the radial line heaters and the four radial jet thruster heaters are on continuously. The axial jet thruster heater and associated line heaters can be commanded on and off.

All seven jet tarusters have temperature sensors; the three axial lines also have temperature sensors. However, the long radial lines do not have temperature sensors. Radial line 1/2 heater status (VJ12HS) is monitored by a small series 38.4 ohm resistor with a scale factor of 26.05 ma/volt. At 28 volts, this heater requires 103.4 ma and results in a telemetry signal of 3.97

volts (198 data units). Radial line 3/4 heater status (VJ34HS) is monitored by a small series 42.25 ohm resistor with a scale factor of 23.66 ma/volt. At 28 volts, this heater requires 76.4 ma and results in a telemetry signal of 3.23 volts (151 data units). Improper operation of the radial line heaters will be denoted by a sharp change in the TM signal.

The propulsion tank heater switch in the PIU controls operation of not only the tank heaters themselves but also a large number of associated heaters for propellant, fill and drain, and gas lines; fill and drain valves; latch valves; and a pressure transducer. The propulsion tank heaters and their associated heaters are all upstream of the latch valves. All of these heaters are redundant. The PIU switch selects either the primary or secondary group of heaters (HTT19 or HTTA9; HTT29 or HTTB9). No single point failure will prevent one of the two heaters from being commanded on. Either the primary or secondary group is on continuously.

The two tanks and their propellant lines, have temperature sensors to monitor the operation of their heaters. At launch, the primary heaters will be commanded on. If any of the tank or propellant line temperature sensors indicates heater failure, the secondary group of heaters will be switched on and the primary group will be switched off.

There is not a sufficient number of temperature sensors to monitor all of the other small heaters in parallel with the tank and propellant line heaters. The resolution of the 0 to 18 ampere spacecraft loads current sensor is inadequate to verify their proper operation. Therefore, two additional current sensor measurements, with good low current resolution, were added to monitor two primary tank heater subgroups. The following group of heaters and their nominal current at 28 volts is monitored by a 16.5 ohm current sensor (VLVPHS - Latch Valve Primary Heater Status):

Latch valve 1A	10.46 ma
Latch valve 2A	26.16 ma
Pressure transducer	10.46 ma
Fill and drain line 1	8.20 ma
Fill and drain line 2	8.20 ma
	63.48 ma

This TM channel will read 1.047 volts at 28 volts. The scale factor is 60.61 ma/volt or 1.21 ma per data unit. Failure of any one of the above heaters will result in a minimum change of 6 or 7 data units.

The second group of primary heaters and their nominal current at 28 volts is (VFDVHS - Fill and Drain Valve Primary Heater Status):

Fill and drain valve 1	18.42 ma
Fill and drain valve 2	18.42 ma
Fill and drain valve 3	18.42 ma
Fill and drain 3 gas line	19.36 ma
	74.62 ma

This TM channel has an 8 ohm current sensor that develops a 0.597 volt signal when the voltage is 28 volts. The scale factor is 125.1 ma/volt or 2.50 ma per data unit. Failure of one of the above heaters will result in a minimum change of 7 or 8 data units.

The telemetry values of these 4 propulsion heaters will fluctuate slightly as the bus voltage fluctuates. Also the propulsion heaters and the small current sensors each have a 5% tolerance. In addition, heater resistance increases by approximately 10% when the propulsion heater temperatures are near their maximum levels. Therefore, the actual measured data deviates slightly from the above nominal values. Shown below are actual measured values at a bus voltage of 29.5 volts at approximately +68°F:

Latch valve heater group	63.0 ma
Fill and drain valve heater group	17.5 ma
R1/R2 line heaters	107.8 ma
R3/R4 line heaters	80.0 ma

3.8.3.9 Failure Modes, Redundancy and Corrective Action. The power subsystem has redundant bus limiters, discharge regulators, and battery cells which allow normal or near normal spacecraft operation to continue in case of failure. The UV/OL protective circuits have an override mode that allows by-passing of the reset relays. The OIM and tank heater circuits are redundant and their operation cannot be compromised by any single point failure. A summary of potential failure modes and corrective action is shown in Table 3.8.3.9-1 below.

TABLE 3.8.3.9-1

FAILURE MODES & CORRECTIVE ACTION

FAILURE MODE	CORRECTIVE ACTION/COMMENTS
Bus voltage limiter fails	Send disable command to remove from power bus; 1 limiter is redundant.
Discharge controller fails	Each battery has redundant discharge regulator - select redundant unit by command.
Current sensor fails	Current sensor elements consist of 2 parallel power resistors (shunts) to prevent open in power bus - if sensor electronics fails, unknown current can be derived from other current sensor measurements with aid of equations in Section 3.8.3.2.
Charge controller relay K1 fails in high state	By means of relay K2 can command to high charge and off only - if battery gets too hot in high charge state, must periodically command on and off by ground or memory command.
Charge controller relay K1 fails in low state.	By means of relay K2 can command to low charge or off only; would not be able to completely recharge one of the batteries during longest eclipse periods but sufficient capacity will be available from 2 batteries.
Charge controller relay K2 fails in high state	By means of relay K1 can command to high charge and off only - if battery gets too hot in high charge state, must periodically command on and off by ground or memory command.

TABLE 3.8.3.9-1 (Continued)

FAILURE MODE	CORRECTIVE ACTION/COMMENTS
Charge controller relay K2 fails in low state	By means of relay K1 can command to low charge or off only; would not be able to completely recharge one of the batteries during longest eclipse periods but sufficient capacity will be available from 2 batteries.
Overtemperature circuit inadvertently turns off relay K1	Relay K1 can be commanded back to previous positon by ground command.
UV/OL trips off inadvertently	Bypass UV/OL reset relays by turning on relay K4 with "Power subsystem protection off" command
Precharge Circuit fails	If UV/OL trip occurs and loads unlatch after reset command, send pulse commands to turn them on again.
Radial Jet Heater 1, 2, 3 or 4 fails	Redundant radial jets available if defective heater results in excessively cold jet.
Cannot turn on or turn off aft axial heater	Use redundant axial jet with suitable temperature as verified by temperature sensor.
Cannot turn forward axial jet heater on or off	If forward axial jet temperature not suitable, use redundant aft axial jet.
Open cell in a battery pack (very low probability event)	Loss of complete battery - turn off charge to failed battery. During long apoapsis eclipses minimize spacecraft loads to prevent complete discharge of remaining battery. (See Section 4.3.5.1)
Shorted cell in a battery pack	Battery will continue to operate with one or two shorted cells with reduced capacity.

TABLE 3.8.3.9-1 (Continued)

FAILURE MODE	CORRECTIVE ACTION/COMMENTS
One or more heaters of primary tank group fails	Transfer to secondary tank heater group
One or more of three OIM group of heaters fails	Select redundant OIM group of heaters.
Radial 1 and radial 2 line heaters defective as verified by R1 and R2 Line Heater Status TM.	Use radial 3 and radial 4 jets where possible.
Radial 3 and radial 4 line heaters defective as verified by R3 and R4 Line Heater Status TM.	Use radial 1 and radial 2 jets where possible.

3.8.4 Power Subsystem Command Response. Table 3.8.4-1 below explain the function of each power subsystem command and how the associated telemetry signals measure the response to these commands. The table assumes that commands will be sent during full solar panel illumination. Normally power subsystem commands will not be sent during eclipses.

During most full illumination time periods there is excess panel capacity that forces current through the bus limiter load resistors. Whenever this is the case, an increase in solar panel current results in an increase of bus limiter current by the same amount. Also, when a load is turned off, as reflected in a decrease in the spacecraft load current, the bus limiter current must increase by this same amount.

TABLE 3.8.4-1

POWER SUBSYSTEM COMMAND RESPONSE

Command Mnemonic	Command Title	Command Response	Telemetry Mnemonic	Telemetry Title/Comments
BAT10 BATA0	Battery 1 - Relay 1 Lo Rate Charge	Commands battery 1 to lo charge rate if K2 in lo state; turns battery 1 charge off if K2 in hi state	PB1RIS	Battery 1, Relay 1 Status goes to lo state (= "0")
			PCHG1I	Battery 1 Charge Current changes value to low charge rate level or to zer
			PPANLI	Solar Panel Current changes by same quantity as battery charge current but change is in opposite direction
BAT19 BATA9	Battery 1 - Relay 1 Hi Rate Charge	Commands battery 1 to hi charge rate if K2 in hi state; turns battery 1 charge off if K2 in lo state	PB1RIS	Battery 1, Relay 1 Status goes to hi state (= "1")
			PCHG1I	Battery 1 Charge Current changes value to high charge rate level or to zero.
			PPANLI	Solar Panel Current changes by same quantity as battery charge current but change is in opposite direction
BAT11 BATA1	Battery 1 Primary Discharge Regulator Select	Input of primary discharge regulator connected to battery 1 and output to essential bus	PREG1S	Primary/Redundant Discharge Regulator 1 Status goes to primary state (= "1").
BAT12 BATA2	Battery 1 Redundant Discharge Regulator Select	Input of Redundant discharge regulator connected to battery 1 and output to essential bus	PREG1S	Primary/Redundant Discharge Regulator 1 Status goes to "redundant" state (= "0").

TABLE 3.8.4-1 (Continued)

Command Mnemonic	Command Title	Command Response	Telemetry Mnemonic	Telemetry Title/Comments
BAT13 BATA3	Battery 1 - Relay 2 Hi Rate Charge	Commands battery 1 to hi charge rate if K1 in hi state; turns battery 1 charge off if K1 in lo state	PB1R2S	Battery 1, Relay 2 Status goes to hi state
			PCHG1I	Battery 1 Charge Current changes value to high charge rate level or to zero
			PPANLI	Solar Panel Current changes by same quantity as battery charge current but change is in opposite direction
BAT14 BATA4	Battery 1 - Relay 2 Lo Rate Charge	Commands battery 1 to lo charge rate if K1 in lo state; turns battery 1 charge off if K1 in hi state	PB1R2S	Battery 1, Relay 2 Status goes to lo state (= 0).
			PCHG1J	Battery 1 Charge Current changes value to low charge rate level or to zero
			PPANLI	Solar Panel Current changes by same quantity as battery charge current but change is in opposite direction
BAT2Ø BATBØ	Battery 2 - Relay 1 Lo Rate Charge	Commands battery 2 to lo charge rate if K2 in lo state; turns battery 2 charge off if K2 in hi state	PB2R1S	Battery 2, Relay 1 Status goes to lo state (= 0).
			PCHG2I	Battery 2 Charge Current changes value to low charge rate level or to zero
			PPANLI	Solar panel current changes by same quantity as battery charge current but change is in opposite direction.

TABLE 3.8.4-1 (Continued)

Command Mnemonic	Command Title	Command Response	Telemetry Mnemonic	Telemetry Title/Comments
BAT29 BATB9	Battery 2 - Relay 1 Hi Rate Charge	Command battery 2 to hi charge if K2 in hi state; turns battery 2 charge off if K2 in lo state	PB2R1S	Battery 2, Relay 1 Status Goes to hi state (= 1)
			PCHG2I	Battery 2 Charge Current changes value to high charge rate level or to zero
			PPANLI	Solar Panel Current changes by same quantity as battery charge current but change is in opposite direction
BAT21 BATB1	Battery 2 Primary Discharge Regulator Select	Input of primary discharge regulator connected to battery 2 and output to essential bus	PREG2S	Primary/Redundant Discharge Regulator 2 Status goes to primary state (= 1).
BAT22 BATB2	Battery 2 Redundant Discharge Regulator Select	Input of Redundant discharge regulator connected to battery 2 and output to essential bus	PREG2S	Primary/Redundant Discharge Regulator 2 Status goes to redundant state (= 0).

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TABLE 3.8.4-1 (Continued)

Command Mnemonic	Command Title	Command Response	Telemetry Mnemonic	Telemetry Title/Comments
BAT23 BATB3	Battery 2 - Relay 2 Hi Rate Charge	Command battery 2 to hi charge if K1 in hi state, turns battery 2 charge off if K1 in lo state	PB2R2S	Battery 2, Relay 2 Status goes to hi state (= 1)
			PCHG2I	Battery 2 Charge Current changes value to high charge rate level or to zero
			PPANLI	Solar Panel Current changes by same quantity as battery charge current but change is in opposite direction.
BAT24 BATB4	Battery 2, Relay 2 Lo Rate Charge	Command battery 2 to lo charge if K1 in lo state; turns battery 2 charge off if K1 in hi state	PB2R2S	Battery 2, Relay 2 Status goes to lo state (= 0)
			PCHG2I	Battery 2 Charge Current changes value to low charge rate level or to zero
			PPANLI	Solar Panel Current changes by same quantity as battery charge current but change is in opposite direction
PSP19 PSPA9	Power System Protection ON	Commands relay K4 of UV/OL into OFF position	PSPROS	Power System Protection on/off (redundant) goes to ON state (UV/OL Protection active) (= 1)
PSP1Ø PSPAØ	Power System Protection Off	Commands relay K4 of UV/OL into ON position	PSPROS	Power System Protection on/off (redundant) goes to OFF state (= 0); applies power to science, switched load and r.f. transmitter buses (UV/OL Protection overridden).
INS1Ø INSAØ	All Science Off	Commands relay K1 of UV/OL into off position	PSCBUS	Science Bus Reset Relay Status goes to OFF state (= 0); power to science bus removed.
			PBUSLI	Spacecraft loads current will decrease
			PSCICI	Science current will decrease to zero

TABLE 3.8.4-1 (Continued)

Command Mnemonic	Command Title	Command Response	Telemetry Mnemonic	Telemetry Title/Comments
PSP1! PSPA1	UV/OL Relays Reset	Commands relays K1, K2 and K3 into on position	PSCBUS PLDBUS PRFBUS	Science Bus Reset Relay Status, Switched Loads Reset Relay Status, and Xmtr Bus Reset Relay Status signals all go to hi state (= "1")
			PBUSLI	Spacecraft loads current will increase from last observed value if science bus and switched loads bus had tripped off.
PCG19 PCGA9	Precharge On	Commands precharge switch on	PRCHGS	Precharge on/off status goes to ON state (="1")
PCG1Ø PCGAØ	Precharge Off	Command precharge switch off	PRCHGS	Precharge on/off Status goes to OFF state (="0")
HTT19 HTTA9	Primary Propellant Tank Heaters Select	Switches essential bus power from secondary to primary tank heaters	PHTTKS	Primary/Secondary Propulsion Tank Heater Status goes to primary state (="1").
			VLVPHS	L/V primary htr status is nom. 164 ma; F&D val.
			VFDVHS	Primary htr status is nom. 75 ma.
HTT29 HTTB9	Secondary Propellant Tank Heaters Select	Switches essential bus power from primary to secondary tank heaters	PHTTKS	Primary/Secondary Propulsion Tank Heater status goes to secondary state (="0").
			VLVPHS VFDVHS	LV primary htr status & F &D valve Primary Heater status both go to zero ma.
HTJ19 HTJA9	Forward Axial Jet Heaters On	Turns on forward axial jet 5 heater	PHTFJS	Forward Axial Jet Heaters on/off goes to ON state (="1"). S/C load current sensor resolution inadequate to measure 15 ma current change
			VJET5T	Forward Axial Jet Temperature rises
HTJ1Ø HTJAØ	Forward Axial Jet Heaters Off	Turns off forward axial jet 5 heater	PHTFJS	Forward Axial Jet Heaters on/off goes to OFF state (="0").
			VJET5T	Forward axial jet temperature decreases.

Table 3.8.4-1. (Continued)

Command Mnemonic	Command Title	Command Response	Telemetry Mnemonic	Telemetry Title/Comments
HTJ29 HTJB9	Aft Axial Jet and Line Heaters On	Turns on Aft axial jet 6 and 7 heaters and lines	PHTAJS	Aft Axial Jet Heaters on/off goes to ON state (="1")
			VJET6T	Aft Axial Jet 6 temperature rises
			VJET7T	Aft Axial Jet 7 Temperature rises
			PBUSLI	Spacecraft Loads Current increases by 1 or 2 data bits
HTJ2Ø HTJBØ	Aft Axial Jet and Line Heaters Off	Turnoff Aft axial jet 6 and 7 heaters and lines	PHTAJS	Aft Axial Jet Heaters on/off goes to OFF state (="0")
			VJET6T	Aft Axial Jet 6 Temperature decreases
			VJET7T	Aft Axial Jet 7 Temperature decreases
			PBUSLI	Spacecraft Loads Current decreases by 1 or 2 data bits
HTM19 HTMA9	OIM Heater 1 On	Turns on OIM heater 1	PHT1MS	OIM Heater 1 on/off goes to on state (="1")
			PBUSLI	Spacecraft Loads Current increases by approximately 0.6 amperes
			VTHROT	OIM Throat Temperature verifies heater on
			VCASET	OIM case temperature verifies heater is ON.
			VSAFET	Safe & Arm Unit temperature verif. htr is ON.
HTM1Ø HTMAØ	OIM Heater 1 Off	Turns off OIM heater 1	PHT1MS	OIM Heater 1 on/off goes to OFF state (= "0").
			PBUSLI	Spacecraft Loads Current decreases by approximately 0.6 amperes
			VTHROT	OIM Throat Temperature decreases
			VCASET	OIM Case Temperature decreases
			VSAFET	Safe & Arm Unit temp. verifies htr is OFF

Table 3.8.4-1. (Continued)

Command Mnemonic	Command Title	Command Response	Telemetry Mnemonic	Telemetry Title/Comments
HTM29 HTMB9	OIM Heater 2 On	Turns on OIM heater 2	PHT2MS	OIM Heater 2 on/off goes to ON state (= "1").
			PBUSLI	Spacecraft Loads Current increases by approximately 0.6 amperes
			VTHROT	OIM Throat Temperature verifies heater on
			VCASET	OIM Case Temperature verifies htr ON.
			VSAFET	Safe & Arm Unit Temp. verifies htr ON.
HTM2Ø HTMBØ	OIM Heater 2 Off	Turns off OIM Heater 2	PHT2MS	OIM Heater 2 on/off goes to OFF state (=0)
			VSAFET	Safe & Arm Unit temp. verifies htr ON.
			PBUSLI	Spacecraft loads current decreases by approximately 0.6 amperes.
			VTHROT	OIM Throat Temperature decreases
			VCASET	OIM Case Temperature decreases
LIM19 LIMA9	Bus Limiter 1 and 2 Enable	Latching relay connects bus limiters 1 and 2 to essential bus	PLIM1S	Bus Limiter 1 Enable/Disable goes to enabled state (= "1").
			PLIM2S	Bus Limiter 2 Enable/Disable goes to enabled state (="1").
LIM1Ø LIMAØ	Bus Limiter 1 Disable	Latching relay disconnects bus limiter 1 from essential bus	PLIM1S	Bus Limiter 1 Enable/Disable goes to disabled state (="0").
LIM2Ø LIMBØ	Bus Limiter 2 Disable	Latching relay disconnects bus limiter 2 from essential bus	PLIM2S	Bus Limiter 2 Enable/Disable goes to disabled state (="0").
LIM39 LIMC9	Bus Limiter 3 and 4 Enable	Latching relay connects bus limiters 3 and 4 to essential bus	PLIM3S	Bus Limiter 3 Enable/Disable goes to enabled state (="1").
			PLIM4S	Bus Limiter 4 Enable/Disable goes to enabled state (="1").

TABLE 3.8.4-1 (Continued)

Command Mnemonic	Command Title	Command Response	Telemetry Mnemonic	Telemetry Title/Comments
LIM3Ø LIMC0	Bus Limiter 3 Disable	Latching relay disconnects bus limiter 3 from essential bus.	PLIM3S	Bus Limiter 3 Enable/Disable goes to disabled state (="0").
LIM4Ø LIMDØ	Bus Limiter 4 Disable	Latching relay disconnects bus limiter 4 from essential bus.	PLIM4S	Bus Limiter 4 Enable/Disable goes to disabled state (="0").
LIM59 LIME9	Bus Limiter 5 and 6 Enable.	Latching relay connects bus limiters 5 and 6 to essential bus.	PLIM5S	Bus Limiter 5 Enable/Disable goes to Enabled State (="1").
			PLIM6S	Bus Limiter 6 Enable/Disable goes to Enabled State (="1").
LIM5Ø LIMEØ	Bus Limiter 5 Disable	Latching relay disconnects bus limiter 5 from essential bus.	PLIM5S	Bus Limiter 5 Enable/Disable goes to Disabled State (="0").
LIM6Ø LIMFØ	Bus Limiter 6 Disable	Latching relay disconnects bus limiter 6 from essential bus.	PLIM6S	Bus Limiter 6 Enable/Disable goes to Disabled State (="0").
LIM79 LIMG9	Bus Limiter 7 Enable.	Latching relay connects bus limiter 7 to essential bus.	PLIM7S	Bus limiter 7 Enable/Disable goes to Enabled State (="1").
LIM7Ø LIMGØ	Bus Limiter 7 Disable.	Latching relay disconnects bus limiter 7 from Essential Bus.	PLIM7S	Bus Limiter 7 Enable/Disable goes to Disabled State (="0").

This Figure is a Foldout.

See APPENDIX C

Figure 3.8.1.1-1.

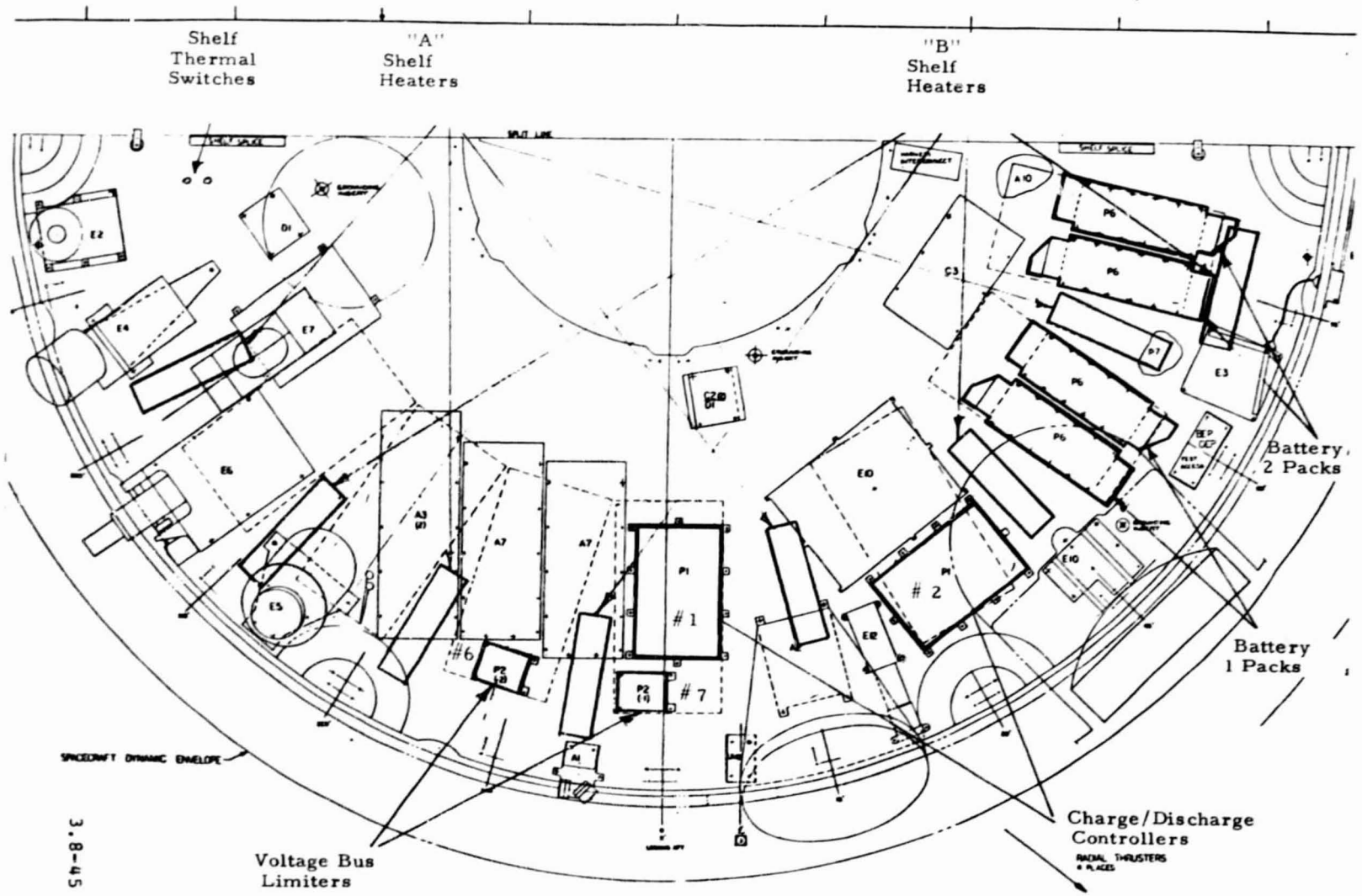


Figure 3.8.1.2-1. Orbiter Shelf No. 1 (Battery) Power Subsystem Unit Locations.

3.8-45

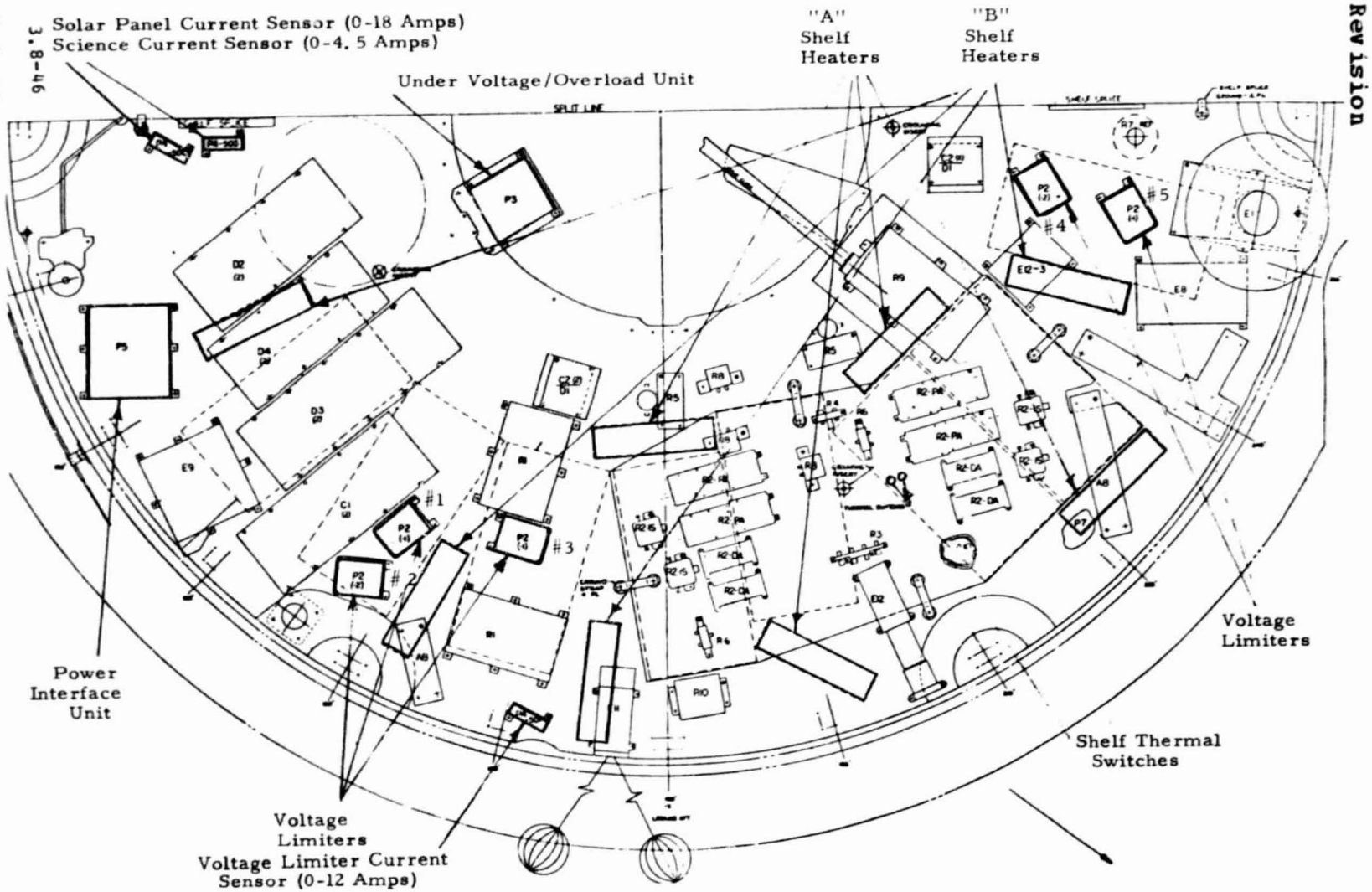


Figure 3.8.1.2-2. Orbiter Shelf No. 2 (R.F.) Power Subsystem Unit Locations.

3.8-46

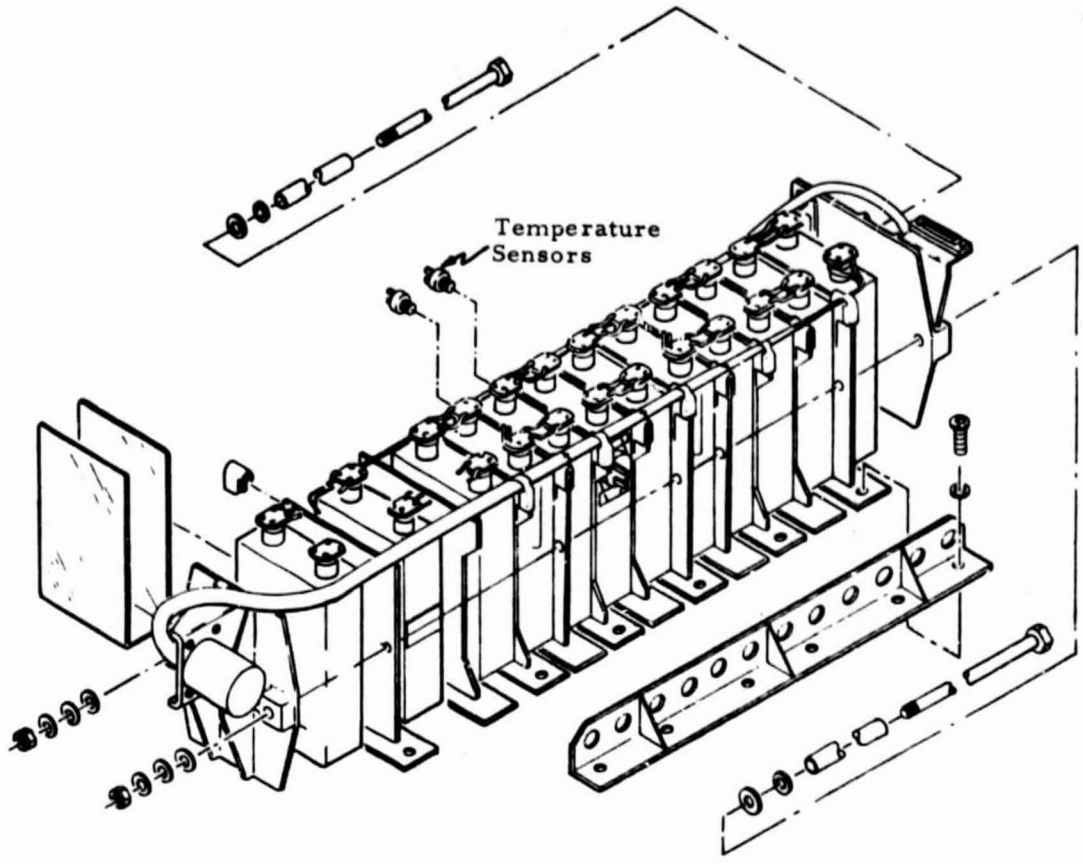


Figure 3.8.2.2-1. Twelve-Cell Battery Pack.

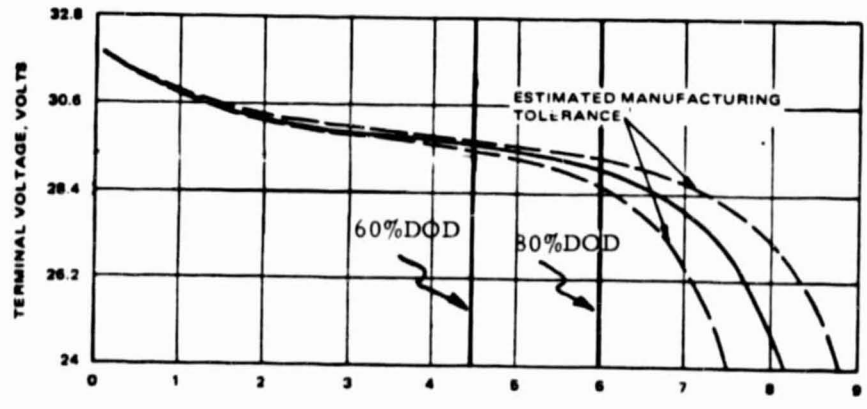


Figure 3.8.2.2-2. First Cycle Performance (75°F and 3.75A Discharge Rate).

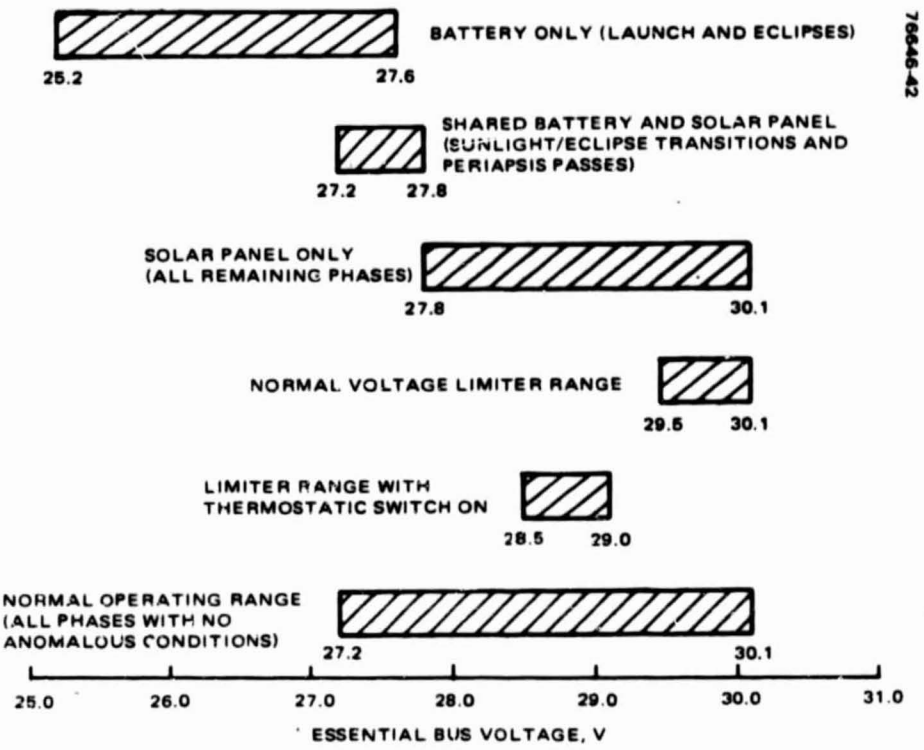


FIGURE 3.8.3.1-1. ORBITER BUS VOLTAGE RANGES

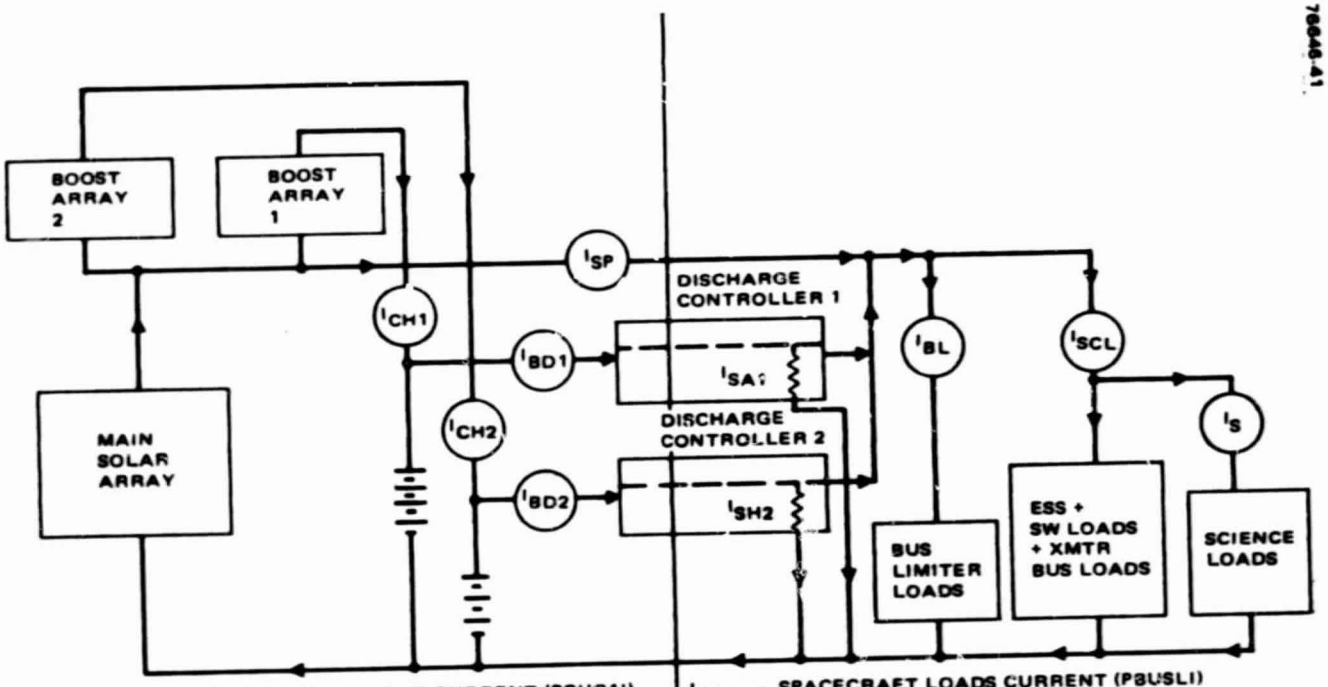


FIGURE 3.8.3.2-1. POWER SUBSYSTEM CURRENT MEASUREMENTS

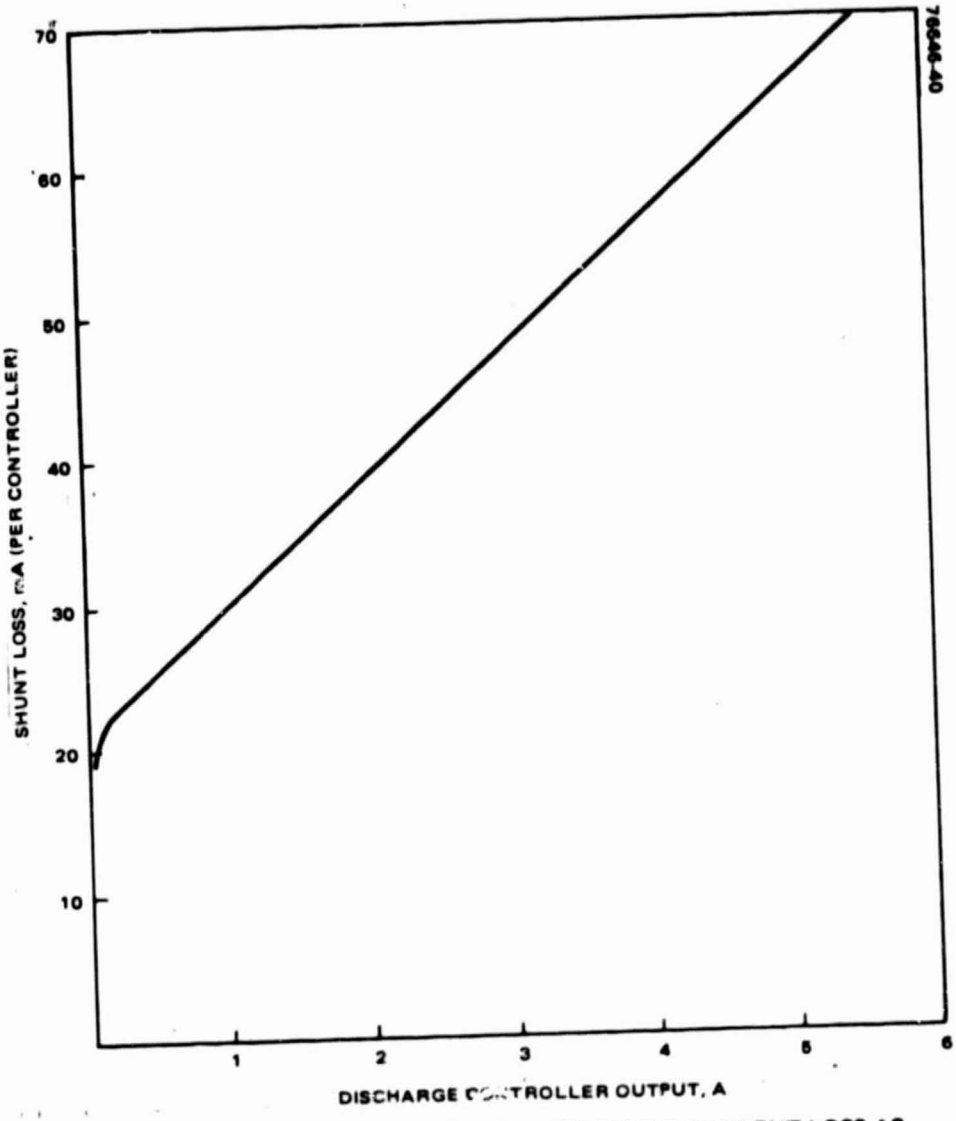


FIGURE 3.8.3.2-2. DISCHARGE CONTROLLER SHUNT CURRENT LOSS AS FUNCTION OF OUTPUT CURRENT

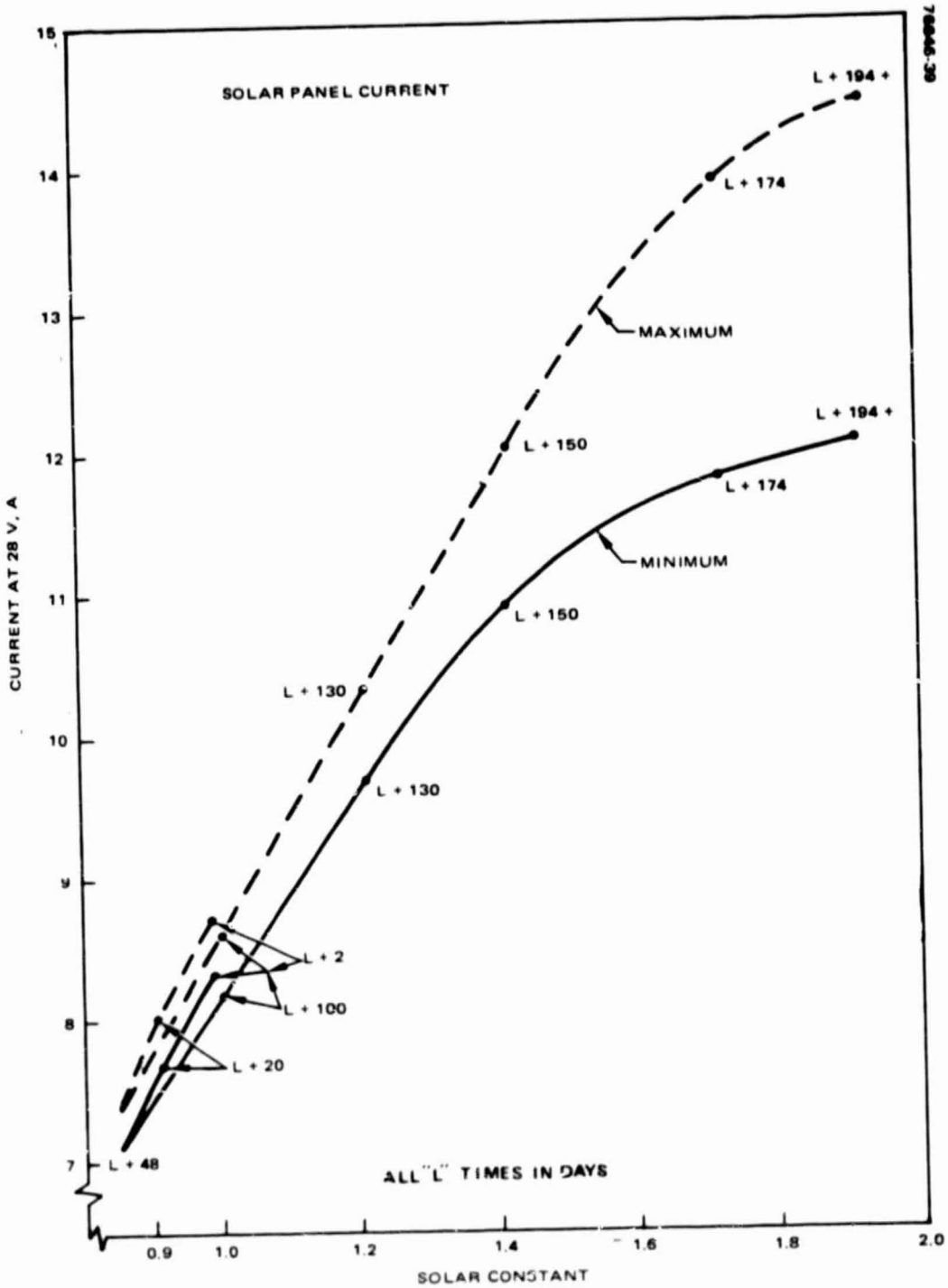


FIGURE 3.8.3.3-1 SOLAR PANEL CURRENT AS FUNCTION OF SOLAR CONSTANT

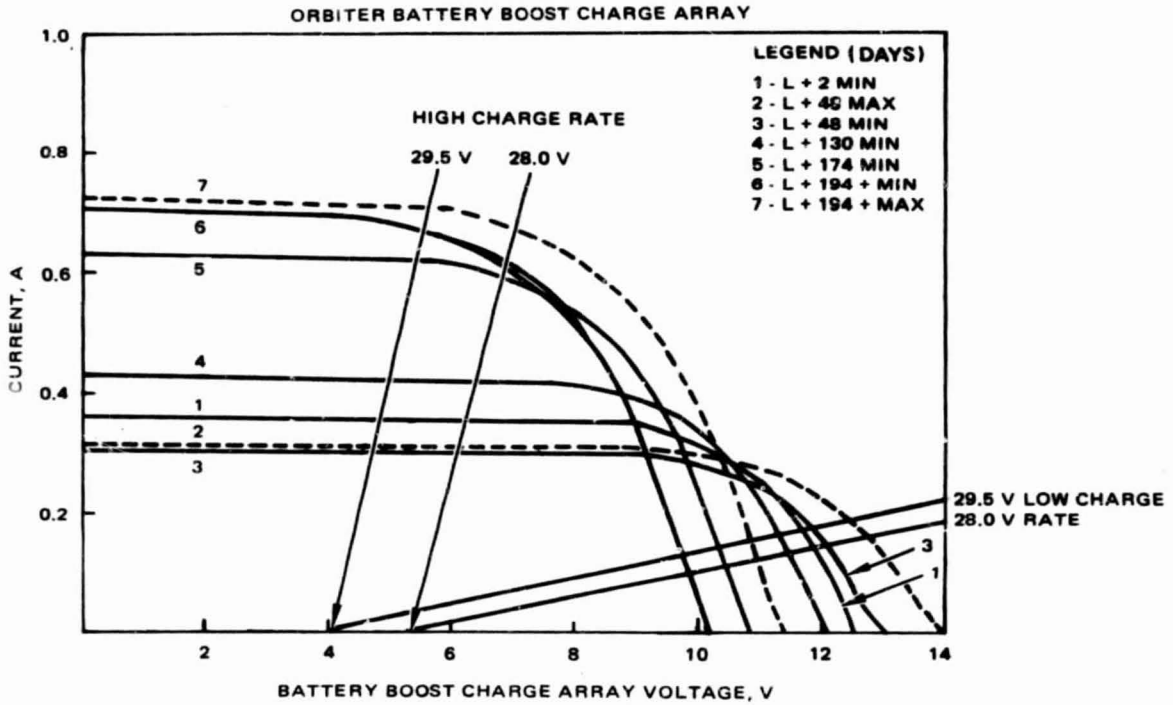


FIGURE 3.8.3.4-1 BATTERY CHARGING LEVELS DUPING VARIOUS MISSION TIMES

4.0 SPACECRAFT MISSION OPERATIONS

Detailed performance of the Orbiter Spacecraft in normal operating modes and backup modes is presented below.

4.1 MISSION MECHANICS

The mechanics of the orbiter mission are delineated by descriptions of (i) the nominal mission profile (4.1.1), (ii) disturbances to nominal spacecraft configurations (4.1.2), and (iii) changes in spacecraft configurations (4.1.3).

4.1.1 Nominal Mission Profile. The nominal mission profile is introduced by a comprehensive mission description from lift-off to end of life (4.1.1.1). A detailed attitude profile (and options) follows (4.1.1.2), with illustrations of spin axis attitude, sun angle and communication angle when appropriate. The spin rate profile (and options) are presented in 4.1.1.3. The maneuver profile propellant budget, maneuver characteristics and constraints are discussed in 4.1.1.4.

4.1.1.1 Mission Description

4.1.1.1.1 Launch Phase (Lift-off to Spacecraft Separation). The Orbiter spacecraft will be launched with an Atlas Centaur launch vehicle from Cape Canaveral. Launch will take place during a one hour window on one of ten successive days in the period from 20 May through 2 June 1978. The launch vehicle will place the orbiter on the desired interplanetary trajectory after a 6 to 18-minute coast period in a 90 nmi earth parking orbit.

After Centaur second burn (MECO 2) and prior to spacecraft separation, the Centaur will reorient the spacecraft +Z axis to an approximately normal to the ecliptic attitude in the direction of the north ecliptic pole. The north ecliptic polar direction has been selected for the orbiter spin axis cruise attitude because of its proximity to the Venus orbit insertion attitude and its resultant propellant saving relative to using the

south ecliptic polar direction. The spacecraft +Z axis is initially tilted away from the north ecliptic toward the local sunline in order to obtain an average sun look angle less than 80 deg required to insure thermal control of the magnetometer boom prior to deployment, some four or more hours later. The amount of tilt selected allows for an attitude uncertainty of ≈9 deg after spinup due to Centaur control errors (5 deg attitude uncertainty, 0.5 deg/sec pitch/yaw rates) and separation tip-off rate error (0.3 deg/sec). After reorientation, the Centaur sequencer initiates spacecraft separation.

4.1.1.1.2 Near Earth Phase (Separation to 36 Hours After Lift-off). During the near earth phase, the Orbiter spin axis, spin rate, attitude and earth pointing states are sequentially established in preparation for the cruise phase. The process begins when spacecraft separation switches initiate redundant command sequences, each sequence being stored in one of the two spacecraft command processors prior to launch. Two spinup thrusters are fired to impart a 6.5 ±0.3 rpm spin rate and the spacecraft is then configured for initial ground station acquisition (Section 4.2.1). It is expected that ground station acquisition at Canberra, DSS-42, will occur during the first four hours after launch. Cebreros, DSS-62, provides nominal coverage during the next six to eight hours while Goldstone, DSS-12, should acquire about 12 hours after lift-off.

The spacecraft is designed so that the nominal spin (+Z) axis will be the axis of maximum moment of inertia when the magnetometer boom is deployed. Before boom deployment, the axis of maximum moment of inertia (true spin axis) is tilted about 9 deg from the +Z axis toward the stowed magnetometer boom. Therefore, sun and earth look angles with respect to the +Z axis will have a ±9 deg variation at spin frequency prior to boom deployment.

Centaur reorientation/spacecraft separation/ spinup nominally tilt the angular momentum vector

(i.e., the average attitude of the true spin axis as well as the +Z axis) 19 deg toward the sunline. Centaur, separation and spinup errors contribute to a 9 deg uncertainty in the average attitude so that the average sun angle lies between 62 and 80 degrees. In the absence of nutation, the true spin axis would be aligned to the angular momentum vector, and the sun look angle with respect to the +Z axis would range from 62 ±9 to 80 ±9 degrees. However, the spinup to the low momentum state required for boom deployment yields ≈10 deg of residual nutation and generates an additional ±10 deg variation in sun and earth look angles at inertial nutation frequency (approximately 20% greater than spin frequency). Therefore, the sun look angle would range from 62 ±19 to 80 ±19 deg immediately after spinup. The residual nutation after spinup is slowly damped out by the nutation damper (≈1 hour time constant for 6.5 rpm spin rate).

The spacecraft is initially below the ecliptic after departure from the earth. If the +Z axis were oriented toward the north ecliptic pole, the initial earth look angle would be 70 to 80 degrees. Because the average attitude after spinup is tilted toward the sun and the sunline and earthline are within 30 deg of one another, the average earth look angle lies between 43 and 71 degrees. Therefore, the earth look angle to the +Z axis would range from 43 ±19 to 71 ±19 deg immediately after spinup.

Spacecraft engineering telemetry will be transmitted via the spacecraft forward omni antenna during the acquisiton period. The sun sensor is used to initially assess spin speed, sun angle and residual nutation. In the presence of nutation, the average value of $\Psi\Psi$ time intervals represents the spin period (nominally 6.5 ±0.3 rpm) while the average value of $\Psi\Psi_2$ time intervals divided by spin period is a measure of sun aspect angle (nominally 62 to 80 deg). The 9 deg tilt of the +Z axis from the true spin axis causes different attitude sensor geometry with respect to the spin axis; this results in an abnormal relationship of $\Psi\Psi_2$/spin period ratio

with sun aspect, and in abnormal fields of view, but no variation in sun aspect angle (angle between sunline and true spin axis) if nutation is absent. Nutation generates a quasi-sinusoidal variation in Ψ_2 about the average value at body nutation frequency (inertial nutation frequency minus spin frequency), i.e., ≈ 20% of spin frequency. The amplitude of the variation in Ψ_2 time intervals is a measure of residual nutation (nominally <10 deg).

The range of potential attitudes, limited number of stars and large residual nutation could frustrate any initial attitude measurements with the star sensor. Satisfactory spin speed and sun angle are both essential criteria for magnetometer boom deployment. Average sun and earth look angles will remain the same after deployment; post deployment nuation could introduce as much as ±30 deg of nutational variation in these angles. Attitude and spin speed trims will be executed, if required, to meet any of these criteria.

The magnetometer boom will be deployed by ground command. Upon command, pin pullers release the caged boom assembly, and centrifugal force developed by the orbiter's spin drives the three hinged boom segments outward. Detent pins at each hinge automatically lock the deployed assembly _10 seconds after start. Since the magnitude and direction of angular momentum is conserved, the average attitude remains the same. The spin rate will decrease to about 5 rpm because of the larger spin moment of inertia when the boom is deployed. The +Z axis is established as the axis of maximum moment of inertia (true spin axis). The average sun angle recorded by the sun sensor will be the same although the average value of Ψ_2 will differ since the attitude sensors now possess nominal geometry with respect to the spin axis. The deployment process does not align the new instantaneous spin axis to the constant angular momentum vector, h, and a large residual nutation results. The +Z axis cones about the angular momentum vector at an average angle of 22 deg, the actual nutation

angle varying between 14 and 30 deg due to asymmetrical moments of inertia. Residual nutation is once again recorded as a quasi-sinusoidal variation in Ψ Ψ_2.

The nutation damper performance after deployment is poor because of large nutation angles and low spin speeds. The nominal mission plan calls for spinup to 15 rpm after deployment. Increasing the spin speed by a factor of three reduces the average nutation angle from 22 to 7 degrees. The higher spin speed also generates an excitation frequency (body nutation frequency) closer to the damper natural frequency. This results in a nutation damping time constant <30 minutes even at the larger nutation angles and the residual nutation is <0.1 deg within two hours of the secondary spinup.

After the steady state spin about the +Z axis is achieved, the attitude will be nominally reoriented to within 2 deg of the north ecliptic pole in preparation for the cruise phase. Prior to this 19 ±9 deg reorientation, star sensor measurements are obtained to define attitude as well as maneuver parameters. While the star Vega is nominally visible, it is possible that a small maneuver may be executed on sun angle information only in order to acquire a star. The maneuver to the north ecliptic pole will nominally be based on attitude information so that when attitude measurements are obtained after the maneuver, an initial calibration of axial jet(s) pulsed performance can be made.

After spacecraft attitude and spin rate have been adjusted to their cruise values, the high gain antenna can be despun to establish earth pointing. The spacecraft trajectory carries it below the ecliptic plane so that the earth's elevation with respect to the spacecraft spin plane will be positive. The earth's elevation starts at +10 to +20 deg, decreases slowly through the cruise phase and ultimately becomes negative ≈ 180 days after launch. In order to utilize the high gain antenna (HGA), the reflector must be initially stepped up in

elevation (toward the +Z axis), i.e., the negative slew direction) some 5 to 10 deg at acquisition and slowly decreased through the cruise phase. Antenna elevation open loop adjustment will nominally take place after closed loop azimuth pointing of the HGA is achieved. The latter will employ the selected roll reference (SRR), nominally the sun, as its standard inertial reference and a commanded azimuth position relative to the sun based on available trajectory and attitude information. Both commanded antenna azimuth and elevation angles may be incrementally adjusted to find the center of the beam. It is recommended that a command to switch back to forward omni at a designated time be stored onboard, in the event of an anomaly during the initial attempt to point the HGA.

4.1.1.1.3 Interplanetary Cruise Phase (36 Hours After Lift-off to Venus Encounter). For a type II earth-to-Venus trajectory, the angle between the sun-Earth line at launch and the sun-Venus line at arrival (heliocentric transfer angle) is greater than 180 deg. As shown in Figure 4.1.1.1-1, this trajectory will remain outside of the Earth's orbit until 87 to 105 days after launch, pass between the Earth and sun (inferior conjunction) at 157 to 179 days, and proceed to intercept Venus at 0.72 AU from the sun at 183 to 202 days after launch. This trajectory allows a 2-1/2 month separation between Orbiter and Multiprobe (Type I) launch opportunities, while being compatible with scientific objectives and maximum in-orbit payload.

Trajectory correction maneuvers (TCM) will be needed to compensate for launch vehicle injection errors and maneuver execution errors. The corrections are expected to total not more than 8.2 m/s (3σ) in spacecraft velocity change (ΔV). Three trajectory correction maneuvers are planned, for approximately five and twenty days after launch, and twenty days before Venus encounter. The first maneuver is the largest, requiring a (99.5% probability of not exceeding) ΔV of 7.5 m/sec to correct launch vehicle

injection errors. This maneuver will nominally be performed in the north ecliptic polar attitude using successive firings of axial and radial jets. Since either radial jet pair generates thrust essentially normal to the spin axis and the aft and forward axial jets generate thrust parallel to the spin axis, the required ΔV can be delivered in any direction by vector combination of individual axial and radial jet impulses. The second and third TCMs, expected to be require not more than a ΔV of 0.6 m/s and 0.1 m/sec respectively, are also executed with the vector mode; these maneuvers correct execution errors resulting from the previous maneuver, and modelling errors compared with the actual solar pressure encountered in the mission. The vector mode has been nominally selected for orbiter TCMs because it does not appreciably change the steady state cruise configuration while incurring a small propellant penalty for the worst case ΔV magnitude and direction.

Command and control of the orbiter will normally be exercised utilizing the DSN 26-meter net except that when maneuvers are being performed the 64-meter stations will be used. Prior to the start of the first two TCMs, the earth look angle is 70 to 80 deg; prior to the third TCM, the look angle is $\approx 90^{\circ}$. The combination of forward and aft omni's will ensure command function in the event of an anomalous attitude change. The HGA will nominally remain despun during any interplanetary trajectory correction maneuver. Prior to and following and axial or radial jet firing, attitude, spin rate and nutation will be determined from sun and star sensor data. Earth look angles during the first two TCMs cause 20 to 30% of the ΔV imparted by axial jets to be reflected in range rate change and thus doppler frequency shift; negligible shift would be observed during the third TCM. Thus, near-real-time measurement of performance during the TCMs is available via doppler shift. Doppler change associated with any ΔV imparted by a radial jet pair is dependent on the direction of the impulse with respect to the spacecraft/earth line in the ecliptic plane.

The order of axial and radial jet firing can be significant. For radial jet pair pulsed maneuvers of five meters/sec, the accumulated precession and spin speed change is on the order of 2.5 deg and 1.5 RPM respectively; the residual nutation is negligible. For a single axial jet continuous maneuver of the same magnitude, precession is negligible, spin speed change is ≈0.6 RPM and nutation is several deg. For a two (aft) axial jets firing, both precession and nutation are negligible while spin speed change is ≈0.6 RPM. In order to minimize the effect of coupling errors due to the first part of the vector maneuver on the performance of the second part, the following sequence is recommended: axial jet maneuvers employing both aft jets always have first priority; radial jet pair pulsed maneuvers have second priority when the ΔV correction is small such as for the second and third corrections; forward axial jet maneuvers have second priority when the radial ΔV is more than several m/sec, but at least a half hour should be allocated between both parts for nutation damping. An attitude touchup and spin speed trim will be executed if required after the maneuver is complete and nutation is damped.

The initial reorientation to the north ecliptic pole affords an opportunity to calibrate the axial jet(s) in the pulsed mode. A radial jet pair in the pulsed mode has potentially strong spin and precession coupling during a velocity maneuver. A calibration maneuver (perhaps a day earlier than the first TCM) on the cruise radial jet pair is recommended to anticipate and plan for any large coupling errors that might be experienced during the first TCM.

Star and sun sensors will be used to provide attitude data for maneuvers as well as steady state operations. The star sensor with its redundant silicon detectors is expected to be responsive to 25 stars. The star sensor field of view is the sector between 44 and 68 deg from the spacecraft spin axis. This field sweeps out a 24 deg wide band of the sky as the spacecraft spins. The stars which fall in the sensor field of view

depend on the spacecraft attitude. For the nominal cruise attitude (north ecliptic polar), four stars are potentially visible: Arcturus, Altair, Mirac, and Capella. Sometime during the interplanetary cruise, the relative celestial longitude of each one of these stars with respect to the sun will cause sun interference. However, at least two of the four stars are useful attitude references at all times. Sun angle measurements are available for sun look angles greater than 15 deg and less than 165 deg, when the appropriate mid-range or extended range sensor is selected.

Attitude determination in its simplest form requires a sun measurement and a star measurement. Multiple stars permit estimation of sensor biases, and therefore improve attitude accuracy and evaluation of solar disturbance torques. Star sensor data on the available star set will be obtained periodically throughout cruise to improve attitude estimation. Assessment of attitude changes in association with TCMs or attitude touchups employs the minimum data set with updated sensor biases. Spin period determination is routinely made by averaging sun sensor $\Psi\,\Psi$ time intervals. Nutation transients and decay can be observed in the quasi sinusoidal variation in $\Psi\,\Psi_2$ time intervals at one quarter of spin frequency. Telemetry will be transmitted primarily via the high gain antenna with the forward omni as backup during the first ≈180 days.

4.1.1.1.4 Venus Arrival Phase (Venus Encounter to Completion of Orbit Insertion). The Orbiter approaches Venus with a relative velocity of 3.6 km/sec. As illustrated in Figure 4.1.1.1-2, the trajectory carries the spacecraft over the northern hemisphere at Venus. The gravitational attraction of Venus causes the velocity to increase 10.7 km/sec at the point of closest approach, nominally 450 km altitude. Approximately 8 minutes before closest approach, the spacecraft will disappear from earth view behind Venus. Retrovelocity (orbit insertion motor impulse) is directed nearly parallel and

opposite to the spacecraft velocity at closest approach to obtain a near 24-hour orbit with a baseline periapsis altitude of 450 km. The Venus approach trajectory will be targeted to produce an orbit inclined 105 deg to the ecliptic plane, as shown in the figure. Initial periapsis latitude increases from 15 to 32 deg as liftoff is delayed through the launch opportunity. Orbit insertion will always occur in sunlight, the injection point being at least 20 deg in longitude from the terminator.

The nominal insertion attitude depends on the day of launch and time of arrival. However, the range of the spin axis attitude is within 20 to 35 deg of the north ecliptic pole. The sun look angles range from 85 to 95 deg so that no time limit due to thermal constraints exists. The earth look angles range from 78.5 to 64.5 deg so that the up and down link can be maintained using the forward omni antenna and the 64 meter DSN net. For launches prior to May 27, Mirach is the primary stellar reference; Capella is the primary stellar reference for the second half of the launch period. Both stars will be visible during the cruise period.

The actual insertion attitude depends on the mismatch between the spacecraft weight used for design of the orbit insertion motor and actual spacecraft weight. The excess capability must be dissipated by a non-optimal attitude. It may be dissipated in useful ways such as raising periapsis latitude, employing an attitude with two stellar references, or biasing the orbital period low to minimize the maximum initial orbit trim ΔV. Sun and Earth look angles would not be significantly different for any of these approaches.

Prior to orbit insertion, transmit and receive functions are switched from the HGA to the forward omni antenna. The spin axis is precessed to the target attitude. Attitude measurements are then made on the available star(s) under cruise conditions, i.e., 15 RPM and antenna despun. This is particularly useful for Mirach,

one of the weakest stars in the reference star set. With the attitude within several degrees of the target attitude, despin control is terminated and the antenna allowed to spin up. The spacecraft is then spun up to 30 rpm to provide the proper gyroscopic stiffness during OIM burn. The attitude perturbation due to this maneuver is small. After verification of proper spin speed change and a spin speed trim, if necessary; the antenna is again despun. (Note that since the antenna is somewhat statically and dynamically imbalanced, the resulting tilt of the true spin axis from the +Z axis when the antenna is spinning (≈1 deg) would bias attitude data). Attitude measurements are again initiated and an attitude trim is planned and executed. After the attitude is properly defined, despin control is again terminated in preparation for retrofire. The antenna is allowed to spin up prior to injection in order to minimize the effect of the transverse torque resulting from antenna static imbalance and nominal thrust along the +Z axis. Since motor firing occurs behind the planet relative to an Earth observer, the necessary commands will be stored and verified in the command memories well before initial occultation.

The two most significant errors associated with the orbit insertion maneuver are uncertainties in the altitude at closest approach (90 km (3σ)) and orbit insertion ΔV magnitude. The latter is a function of uncertainty in motor impulse (±0.6%) and cosine loss. An initial attitude error of 2 deg would not cause significant errors in initial orbit parameters.

4.1.1.1.5 <u>Venus Orbit Phase (Orbit Insertion to 243rd Day of Orbit)</u>. Shortly after orbit insertion, the spacecraft will be reacquired and undergo engineering checkout. Attitude, spin rate and residual nutation will be monitored to verify no significant changes have occurred. Tracking data is obtained to define orbital period and periapsis altitude. If spacecraft status is satisfactory, the spacecraft can be despun to 15 RPM in preparation for the inversion maneuver to position the spin axis pointing toward the south

ecliptic pole near the first apoapsis. The attitude perturbation due to the spin down maneuver is small. The reorientation to the south ecliptic pole sets up the nominal angle of attack and spacecraft viewing geometry for the initial periapsis pass. The sun look angle during this one hour maneuver can range between 75 and 100 deg while the Earth look angle can decrease to a low of 30 deg before increasing to a nominal 89 deg. After the attitude inversion, the HGA is despun and attitude data are taken to plan and execute an attitude trim, if required. After the latter maneuver, a HGA elevation trim is made and telemetry and command functions are switched to the HGA.

The initial orbit trim consists of orbit correction maneuvers near periapsis and apoapsis to modify the initial Venus orbit (resulting from orbit insertion) to a 24-hour orbit period and a 200 km periapsis altitude. This is required because of the insertion altitude bias, and errors in the approach trajectory and orbit insertion. Optimum velocity changes with axial jets could be made if the spin axis were aligned parallel to the periapsis and apoapsis velocity vectors. Axial thruster corrections are made in the nominal ecliptic normal attitude when the orbital velocity vector subtends a near minimum angle with respect to the spin axis. Corrections near apoapsis will occur about one hour after apoapsis and have a 22% loss in efficiency compared to the optimum. Corrections near periapsis are constrained to take place at least 21 minutes after periapsis to reduce interference with scientific measurements there; periapsis corrections incur a 31% loss in efficiency, but represent only 30% of the total orbit trim ΔV budget over the mission.

If sufficient confidence in periapsis altitude estimation exists after 6 hours of tracking, the second half of the inversion can be done with a single aft axial jet to take advantage of ΔV coupling which causes periapsis lowering. Any additional periapsis lowering is obtained by continuously firing the two aft axial jets after

inversion. If insufficient confidence in periapsis altitude exists, the periapsis lowering maneuver is performed with the aft axials after an additional 24 hours of tracking (i.e., second apoapsis).

If the period is less than 24 hours, the spacecraft can be despun to 5 RPM following periapsis adjustment. Then, after 12 hours at an altitude of 4000 km on the exit side of periapsis, both aft axial jets can be fired continuously to increase apoapsis altitude and achieve a nearly 24 hour period. If the period is greater than 24 hours, it will be necessary to fire the forward axial jet after the initial periapsis pass in order to decrease apoapsis altitude and achieve a nearly 24 hour period. If the spacecraft were despun to 5 RPM prior to a period-shortening maneuver, a large attitude error (≈6 deg) and residual nutation (≈15 deg) could result. It could be considered in this case to make the initial periapsis pass at 15 RPM and despin to 5 RPM prior to the next pass.

After the initial orbit corrections, periodic orbital corrections are required principally to compensate for perturbations to the spacecraft orbit due to solar gravity and atmospheric drag near periapsis. The main effect of the sun is to gradually decrease orbital eccentricity, causing the periapsis altitude to rise and apoapsis altitude to drop. The main effect of atmospheric drag is to reduce the spacecraft periapsis velocity and thus lower apoapsis altitude while reducing orbital period. The baseline orbit control objectives ar to maintain periapsis altitude between 150 km and 260 km, and period near 24 hours. The requirement to lower periapsis altitude and increase apoapsis alittude (and increase orbital period) is satisfied with the aft axial jets when the spin axis is directed toward the south ecliptic pole. Periapsis altitude is lowered by firing the two aft axials continuously about an hour after apoapsis; periapsis corrections are typically 6 m/sec and occur about 8 times in the mission. Apoapsis altitude is increased by firing the two aft

axials continuously at an altitude of 4000 km (no science interference) on the exit side of periapsis; apoapsis corrections are typically 1 m/sec and occur about 10 times in the mission. The pair of aft axials are balanced to minimize attitude perturbations; they are also redundant to one another in that if one fails, the other can execute all required corrections in either the continuous or the twice spin frequency pulsed mode (180 deg bang-bang) mode.

The spacecraft attitude will nominally be maintained within 2 deg of the south ecliptic pole in Venus orbit. Within this attitude band, seven stars are potentially visible: Sirius, Rigel Kentaurus, Rigel, Hadar, Beta Grus, Formalhaut and Gacrux (the latter two are not visible at the pole). The sun will periodically prevent several visible stars from serving as useful references anywhere in orbit. However, at least two stars are useful at any time and at least three stars are useful 90% of the time. Venus itself will interfere with the sensor's view of reference stars as spacecraft comes over the north pole on its approach to periapsis and possibly when the spacecraft approaches the orbit ascending node. During the 12-hour interval between periapsis exit and apoapsis approach, candidate stars for attitude determination need only avoid sun interference to be useful references.

Multiple stars permit estimation of sensor biases, which, after orbit insertion, may differ from those estimated during cruise. After the initial bias estimation during the first few orbits, star sensor data on diverse sets of usable stars will be obtained periodically in orbit to improve attitude estimation accuracy and evaluation of solar and aerodynamic disturbance torques. The solar torque is quasi-static while the aerodynamic torque is the impulsive type, occurring near periapsis when the altitude is below 160 km. The minimum attitude data set with sensor biases updated by periodic estimation will be used to define magnitude and direction of these torques. Weekly attitude corrections can

then be biased such that the attitude will deviate no more than 2 deg from south ecliptic normal during weekly or longer intervals.

The HGA will nominally provide both uplink and downlink capability throughout the nominal in-orbit mission. Since the orbiter's despun antenna must be pointed toward earth for optimum communications, azimuth and elevation position of the antenna beam will be periodically updated to reflect Venus and earth motion about the sun. During the initial 75 to 90 days in orbit, the spacecraft will pass behind Venus and be occulted from earth view for periods up to 24 minutes. An occultation experiment will be conducted employing both S and X-band frequencies radiated toward earth through the fringe of the Venus atmosphere during both entrance into and exit from occultation. The orbiter's despun antenna will be controlled to follow the virtual earth at the limes of Venus for up to 17 deg of refraction. Antenna pointing is controlled by adjusting antenna azimuth position with respect to the selected roll reference and by slewing the position of the antenna reflector relative to the fixed feed in elevation. The required azimuth and elevation profiles for any occultation are executed on-board via stored commands. The 64-meter DSN stations will be utilized for this experiment. A second, shorter occultation season (≈10 days) occurs near the middle of the in-orbit mission, when occultations are longer (one to three hours) because they are nearer to apoapsis.

Two seasons of eclipses occur for the Pioneer Venus orbit: a first season of short eclipses (<25 minutes) near periapsis and a second season of long eclipses (up to 3.5 hours) near apoapsis. The first seaon can begin as early as 24 days after insertion and end as late as 128 days after insertion. Periods of combined solar eclipse and occulation will occur, then, during the last ≈50 days of the first occulation season. Since the eclipse duration is at most 24 minutes near periapsis, a full spacecraft operations can be maintained using energy supplied by the battery. The absence of sun implies switch-over to a

stellar roll reference (nominally Kigel Kentaurus) or to a simulated roll reference prior to eclipse. During the initial 10 to 20 days of the first eclipse season, transfer time between the sun and a usable stellar reference is on the order of two minutes. An option to employ the onboard simulated roll reference pulse to buy more transfer time, or to "flywheel" through the short eclipse, may be exercised, pending in-orbit studies of sun/Venus interference with star detection, and subject to anticipated roll phase error buildup due to speedup of the spin rate induced by thermal shrinking. The long eclipse season starts about 170 days after insertion and lasts for ≈10 days; the spacecraft must be powered down to a survival mode during this season. The long occultation season will always occur between the two eclipse seasons.

Extensive use will be made of the data storage capability and the command memory during orbital operations in order to establish a more efficient schedule of real-time ground operations. The two data storage units will be used to store scientific instrument data recorded around periapsis and for other periods during the orbit for playback at later times. The 26-meter DSN net will be utilized for data gathering and spacecraft command. A 24-hour command sequence of a maximum of 256 commands may be stored in the command memory and might be usable for several days, as the orbital geometry does not change rapidly. Nominal operations, however, will include some real-time commanding during any 24-hour period.

4.1.1.2 Attitude Profile. The spacecraft spin axis is the axis of maximum moment of inertia about which the spacecraft is spun for attitude stabilization. After the magnetometer boom is deployed, the positive spin axis is nominally coincident with the spacecraft +Z axis. In the absence of nutation, the positive spin axis is also coincident with the spacecraft angular momentum and angular velocity vectors. Spacecraft attitude is the direction of the spacecraft angular momentum vector. In the

absence of nutation, spacecraft attitude is identical with spin axis attitude. With nutation present, spacecraft attitude represents the average direction of the positive spin axis. Attitude will be defined by the celestial latitude and longitude of the "average" direction of the positive spin axis, as shown in Figure 4.1.1.2-1. A detailed attitude profile expressed in these coordinates is discussed below.

Sun look angle constraints for the mission are illustrated in Figure 3.2.3-1.

4.1.1.2.1 Launch and Near Earth Phase. Prior to spacecraft separation, the Centaur will orient the spacecraft +Z and +X-axes such that, after separation and spinup, spacecraft attitude will nominally be 19 deg from the north ecliptic pole and the average sun look angle will be less than 80 degrees. The latter condition maintains thermal control of the magnetometer boom prior to deployment some four or more hours later. Figure 4.1.1.2-2 shows that the Centaur orients the +Z axis to 80.9 deg latitude, θ_s deg longitude and the +X axis to -8.7 deg latitude and θ_s -16.9 deg longitude. θ_s is the celestial longitude of the sunline prior to separation; it varies from 56 to 75 deg over the orbiter launch opportunity. Separation and spinup nominally establish spacecraft attitude (angular momentum vector) at 71.1 deg latitude, θ_s longitude. The cone of possible attitudes indicated in the figure reflects a possible attitude error of 9 deg due to Centaur control, separation and spinup errors. Therefore, the average sun look angle can range between 62 and 80 degrees. The earthline is also delineated in the figure with latitude δ_e and longitude θ_e. Both angles vary rapidly during the first two hours following injection. The latitude, δ_e, then settles to 10 to 20 deg depending on launch date. The longitude, θ_e, becomes quasi steady at 20 to 35 deg as a function of launch date. The resulting average earth look angle lies between 43 to 71 degrees.

The spinup process generates ≈10 deg of residual nutation and causes a ±10 deg variation in sun

and earth look angles as the positive spin axis cones around the spacecraft attitude at inertial nutation frequency (approximately 20% greater than spin frequency). Since the positive spin axis is tilted 9 deg from the +Z axis before boom deployment, an additional ±9 deg variation in sun and earth look angles is generated as the +Z axis cones around the positive spin axis at spin speed. Figure 4.1.1.2-3 illustrates the motions of the +Z axis and positive spin axis and how the look angle to an inertial target will vary in the presence of nutation and spin axis tilt from the +Z axis. The nutation damper will slowly (≈1 hour time constant at this spin rate) damp out the ±10 deg variation in look angles, whereas only boom deployment can eliminate the ±9 deg variation in look angles.

The sun and star sensors have different transfer functions and fields of view when the boom is stowed. Figure 4.1.1.2-4 illustrates the angle between the sunline and the positive (true) spin axis versus the measured $\Psi_1\Psi_2$ time interval. Note that, when the boom is stowed, the field of view on the midrange sun sensor is increased while those of the extended range sun sensors are decreased. The sun aspect angle measured by either the upper extended or midrange sun sensor at acquisiton would range from 62 ±10 to 80 ±10 deg in the absence of nutation damping. The stars that are visible near the nominal spacecraft attitude with the boom stowed are illustrated in Figure 4.1.1.2-5. Note that Altair would be the primary stellar reference for the locus of nominal attitudes with no nutation. The 9 deg attitude uncertainty and 10 deg max nutation could make any of the three available stars visible at some time in the nutation cycle. Initial stellar measurements would then be intermittent. As nutation decayed in time, a continuous series of star measurements could be observed. It is also possible that no star would be observed if the attitude were located in the region where all three stars are outside the sensor field of view; however, a reor of the spin axis several degrees away from the sunline would generate a reference star.

The boom will nominally be deployed when sun angle, communications angle, spin speed and residual nutation are satisfactory. For an indefinite attitude dwell during the orbiter mission, power requires an average sun look angle greater than about 40 deg while thermal considerations constrain the average sun look angle to be less than 110 degrees. Telemetry and command via the forward omni antenna require instantaneous earth look angles less than 100 degrees. The spin rate required prior to deployment is 6.5 ±0.3 rpm, while residual nutation should be less than 10 degrees. Since the magnitude and direction of the spacecraft angular momentum is conserved during deployment, the average sun and earth look angles remain the same after deployment. Post deployment nutation could introduce as much as ±30 deg variation in these look angles. Therefore, the initial conditions that nominally exist at acquisiton (average sun angle range 62 to 80 deg, average earth angle range 43 to 71 deg, spin rate 6.5 ±0.3 rpm) are consistent with those required for boom deployment. The sun sensor (mid-range or upper extended) can be used to verify satisfactory sun look angle, spin speed and nutation. Because of spacecraft/earth/sun geometry, this sensor can also be used with ground received signal strength to help verify satisfactory earth look angle. Star sensor data prior to deployment can then be considered optional.

After boom deployment, the sun sensor transfer function and field of view become nominal, as illustrated in Figure 4.1.1.2-6. Although the average sun look angle will not change, the average value of $\Psi_1 \Psi_2$ time intervals observed by the midrange and upper extended sensors will change; this is one indicator of successful deployment. The choice of which sensor to observe during the actual deployment is dependent on the average sun look angle. Since the deployment process generates a large residual nutation (the actual angle varying between 14 and 30 deg), one and possibly both of the sensors will occasionally miss $\Psi_1 \Psi_1$ and $\Psi_1 \Psi_2$ measurements.

The sensor with the best field of view is selected during deployment and both sensors are exercised after deployment to cross check spin speed, average sun look angle and nutation. The spin speed is then increased to 15 rpm, reducing the average nutation angle to ≈7 deg and perturbing spacecraft attitude ≈0.5 degree. The nutation damper will then be more efficiently tuned and will allow reliable star measurements to be made within two hours.

The stars that are visible near the nominal attitudes with the boom now deployed (nominal sensor geometry) are illustrated in Figure 4.1.1.2-7. Note that Vega would be the only stellar reference for the locus of nominal attitudes with no nutation. It is possible that no star would be observed if the attitude were located in the region where all stars are outside the sensor field of view. A reor of the spin axis several degrees toward the north ecliptic pole based on sun information only would make Altair or Arcturus visible. The actual maneuver to the north ecliptic pole that follows would then nominally be based on sun and star information. An initial calibration of axial jet(s) pulsed performance can then be made after attitude measurements at the near-earth cruise attitude are obtained.

4.1.1.2.2 Interplanetary Cruise Phase. During the interplanetary cruise portion of the orbiter mission, the spacecraft positive spin axis (+Z axis) is nominally maintained within 2 deg of the north ecliptic pole (except as needed for trajectory correction maneuvers). Therefore, the celestial latitude of the positive spin axis is 88 to 90 degrees. The celestial longitude would be arbitrary (0 to 360 deg) if the attitude were not biased to take advantage of the entire attitude deadband and minimize the frequency of attitude touchups required to compensate for solar torque precession. Attitude biasing implies that the difference in celestial longitudes of the sunline and spin axis is ±90 deg, so that as solar torque precesses the spin

axis around the sunline, the entire attitude deadband is utilized.

The orbiter sun look angle history for the cruise phase is shown in Figure 4.1.1.2-8. The +Z axis is assumed oriented to the north ecliptic pole. The orbiter earth look angle history for the cruise phase is shown in Figure 4.1.1.2-9. Part of the attitude variation due to solar torque would be reflected in earth look angle through most of the cruise since the earth and sun lines of sight are near coincidence for short periods of time only. This is illustrated in Figure 4.1.1.2-10 by the sun-spacecraft-earth angle during cruise.

The azimuth and elevation of the HGA beam must be periodically updated to compensate for variations in the sun-to-earth spin angle and earth look angle, respectively. Figure 4.1.1.2-10 shows how the sun/spacecraft/earth angle varies during cruise because of the relative motion of the earth and spacecraft in the ecliptic plane. Since the spin axis is within 2 deg of the north ecliptic pole, this figure can be used directly to ascertain the required azimuth pointing command for antenna earth pointing with the sun as roll reference (allowing for bias explained in Section 3.3.2.4.3). If a star is selected as the roll reference, the relative longitude of the sun to the star must be added to the angle obtained from the figure plus a bias unique to star sensor usage. Figure 4.1.1.2-9 shows how the earth look angle changes due to spacecraft motion away from earth and out of the ecliptic plane. Any look angle change due to attitude tilt away from the north ecliptic pole in the direction of the earth-line must be subtracted from this nominal angle before computing the proper elevation angle for the HGA positioner. The HGA 3 dB beamwidth is 8 deg, so that azimuth and elevation angles may be adjusted to peak the gain after the initial open loop commands are executed.

The maximum angular rate of the earthline is about 2 deg/day and occurs toward the last third of the cruise phase. Antenna pointing

corrections (primarily in azimuth) would be required no more frequently than every four days in order to maintain the earthline within the 3 dB beamwidth; this implies that the earthline is allowed to drift from one edge of the beam to the other. A detailed discussion of HGA pointing is given in Section 4.3.4.

The earth look angle should nominally be greater than 85 deg for aft omni satisfactory transmit and receive capability. Figure 4.1.1.2-9 indicates that it requires 120 to 160 days of cruise for the nominal look angle to increase to 85 degrees. In the event of a despin anomaly and loss of HGA pointing, 36 hours may elapse before command capability is restored via the forward omni with a receiver reverse command. An attitude option that could circumvent this condition would require tilting of the positive spin axis 5 to 10 deg away from the north ecliptic pole and the earthline. Part of this tilt would reflect into a sun look angle change (tending to increase it during the first 100 days) and star availability (discussed below) would not be significantly compromised. Another option is to fly the cruise phase with the positive spin axis oriented at the south ecliptic pole. This strategy would always maintain the earth look angle greater than 89 degrees. However, this approach would require an additional pound of propellant for spin axis inversion prior to orbit insertion and would not permit study during cruise of northern hemisphere stars that are references at the orbit insertion attitude. In addition, the earth look angle during the near earth phase would have significant variations about 90 deg and resulting telmetry from the selected omni could be quite intermittent.

A third option is to fly the initial 120 to 160 days with the HGA antenna spinning and the forward omni in the transmit and receive modes. This configuration can support at least 8 bps without the 64 meter net for the first 160 days. Attitude corrections to compensate for solar torque precession would be more frequent. The

antenna would be despun periodically to take unbiased attitude data as well as to check out antenna pointing operations prior to the orbital phase.

The stars available for attitude determination and roll reference during cruise when the spin axis is oriented near the north ecliptic pole are illustrated in Figure 4.1.1.2-11. At least two stars are useful references at any time. Note that Arcturus and Altair are useful references for the first 85 and 170 days, respectively. The celestial longitude of the earth as observed at the spacecraft is illustrated in Figure 4.1.1.2-12. Since the longitude of the earthline varies from 25 to 68 deg during cruise, tilting the spin axis away from the pole and earthline in Figure 4.1.1.2-11 does not effect visibility of Arcturus and Altair. If the tilt is limited to ≈5 deg, Mirach would still be available, starting 36 days after launch. While Capella would not become nominally available at 99 days, the attitude could be returned to the north ecliptic pole prior to the loss of Altair, and two stellar references would still be available at all times. In summary, the earth look angle to the aft omni could be improved ≈5 deg without compromising spacecraft attitude, thermal or power configurations.

Trajectory correction maneuvers are nominally executed in the north ecliptic polar attitude by vector combination of individual axial and radial jet impulses. Forward and aft omnis are selected for command function, in the event of an anomalous attitude change. Two optional TCM methods exist which might avoid any propellant penalty associated with the vector mode, particularly during the first TCM. The axial jet mode requires precession of the spin axis to the desired ΔV direction, continuous aft or forward axial jet thrust, and reorientation of the spin axis back to the north ecliptic pole. Power and thermal considerations require sun look angles between approximately 40 and 110 deg for dwell at any potential maneuver attitude. With the earthline within ≈30 deg of the sunline at first

TCM, the earth look angle would range between 10 and 140 degrees. The available stars or planets are dependent upon the specific attitude required. The radial jet mode requires precession of the spin axis around the sunline until the desired ΔV direction lies in the radial jets thrust plane (i.e., the plane that contains the radial jets thrust vectors), pulsed radial jet pair thrust, and reorientation of the spin axis back to the north ecliptic pole. The sun look angle would be constant (≈90 deg) while the earth look angle would range between 60 and 180 degrees. The available stars again are dependent upon the specific attitude required. Both of these methods require precession at the cost of ≈1 lb/100 degrees. The maximum weight saving associated with these methods for the worst case first TCM ΔV magnitude and direction is ≈0.7 lbs. Therefore, the use of either method will depend on a trade-off of actual weight saving against operational complexity and star availability at the maneuver attitude.

4.1.1.2.3 Arrival Phase. The spin axis attitude at orbit insertion is nominally directed parallel and opposite to the spacecraft velocity at closest approach to Venus. The longitude and latitude of the attitude depend on the day of launch and time of arrival. The range of possible attitudes with corresponding sun and earth look angles are presented in Figures 4.1.1.2-13 and 4.1.1.2-14, respectively. The precession required to achieve the optimum attitude is 20 to 35 degrees. The sun look angles range from 85 to 96 deg so that no time limit due to power or thermal constraints exist. The earth look angles range from 78.5 to 84.5 deg so that the uplink and downlink can be maintained using the forward omni and the 64-meter DSN net.

The stars available for attitude determination at the nominal orbit insertion attitudes are illustrated in Figure 4.1.1.2-15. For launches prior to May 27, Mirach is the primary stellar reference; Capella is the primary stellar reference for the second half of the launch period. Several dimmer stars are available as

references at any potential attitude, as is evident in the figure. Other optional attitudes might be employed to raise periapsis latitude or bias the initial orbit period to limit initial orbit trim ΔV. Sun and earth look angles would not be materially different for any of these approaches and, in all cases, at least one stellar reference will be available.

4.1.1.2.4 Venus Orbit Phase. Following Venus orbit insertion, and spacecraft reacquisition, the spin axis is reoriented to the south ecliptic pole. The sun look angle during this ~160 deg reor can range between 75 and 100 deg while the earth look angle can decrease to a low of 30 deg before increasing to a nominal of 89 degrees. Communications will be maintained using the forward omni and the 64-meter DSN net.

During the Venus orbit phase of the orbiter mission, the spacecraft positive spin axis will nominally be maintained within 2 deg of the south ecliptic pole. Therefore, the celestial latitude of the positive spin axis is -88 to -90 degrees. The celestial longitude depends upon whether attitude biasing schemes are being used. During most of the orbital operations when the spacecraft periapsis altitude is above 160 km, the attitude can be biased so that as solar torque precesses the spin axis around the sunline the entire attitude deadband is utilized. However, whenever periapsis is lowered to ~150 km, the aerodynamic disturbance torque becomes dominant for several days. The attitude would then be nominally biased so that as atmospheric torque precesses the spin axis around the velocity vector at periapsis the entire attitude deadband is utilized.

Sun and earth look angle histories for the nominal 243-day on-orbit phase are presented on Figures 4.1.1.2-16 and 4.1.1.2-17, respectively. The sun/spacecraft/earth angle history is shown in Figure 4.1.1.2-18. The latter figure indicates that part of any attitude variation due to solar torque would be reflected in earth look angle change through most of the on-orbit phase;

the earth and sun lines of sight are near coincidence only toward the latter part of the orbital mission. The atmospheric torque can perturb both sun and earth look angles within the deadband limits.

The azimuth and elevation of the HGA beam must be periodically updated to compensate for variations in the sun-to-earth spin angle and earth look angle, respectively. Figure 4.1.1.2-18 shows how the sun/spacecraft/earth spin angle varies in orbit because of the relative motion of the earth and Venus in the ecliptic plane. Since the spin axis is within 2 deg of the south ecliptic pole, this figure can be used directly to ascertain the required azimuth pointing command for antenna earth pointing with the sun as roll reference (allowing for bias explained in Section 3.3.2.4.3). During occultation entry and exit, when the orbiter's despun antenna will be controlled to follow the virtual earth, a variable azimuth angle must be added to the true earth azimuth. If a star is selected as the roll reference, the relative longitude of the sun to the star must also be added plus a bias unique to star sensor usage. Figure 4.1.1.2-17 shows how the earth look angle changes due to Venus (spacecraft) motion away from earth and out of the ecliptic plane. Any look angle change due to solar or atmospheric torques must be added to this nominal angle before computing the nominal elevation angle for the HGA positioner. During occultation, a variable elevation angle must be added to true earth elevation angle in order to track the virtual earth.

The stars potentially available for attitude determination and roll reference during orbital operations when the spin axis is oriented near the south ecliptic pole are illustrated in Figure 4.1.1.2-19a and Figure 4.1.1.2-19b. At least two stars are not subject to sun interference at any time in orbit, while three stars are free of sun interference 90% of the time in orbit. Venus will interfere with star detection when the spacecraft comes over the pole on its way toward periapsis but will not interfere during the 12-

hour interval between periapsis exit and apoapsis approach. Rigel Kentaurus and Hadar are primary and backup stellar references, respectively, during periapsis passes in eclipse season.

The large number of stars visible in the vicinity of the south ecliptic polar attitude make several attitude options possible. The spin axis could be tilted several degrees away from the pole to increase the earth look angle during the first 100 days in orbit; this would improve the aft omni communications angle. The spin axis could be tilted several degrees toward the velocity vector at periapsis when periapsis lowering maneuvers are required; this would reduce the atmospheric torque and reduce the attitude disturbance. The spin axis could be tilted to change the angle of attack profile for the mass spectrometers or the view traces on the planet for many instruments.

Orbital correction maneuvers are nominally executed in the south ecliptic polar attitude by firing the two aft axials continuously. Periapsis corrections take place about an hour after apoapsis passage. Apoapsis corrections occur about 25 minutes after periapsis passage. Star data are available at both times to confirm minimal attitude disturbance or to plan attitude trims.

4.1.1.3 Spin Rate Profile. Immediately following separation from the Centaur, the orbiter is spun up to 6.5 ±0.3 rpm by a prestored command sequence. The nominal initial spin speed range has been selected to insure successful centrifugal deployment of the Z-fold magnetometer boom. A minimum spin speed of 5 rpm is required to provide sufficient latching energy at each hinge in the event hinge drag torques realize their estimated worst case values. A maximum spin speed of 7 rpm is required to limit latching energy at each hinge to be consistent with structural integrity of the boom in the event hinge drag torques are negligible. Because a tolerance of 5% on the initial spin speed (biased toward the high side of the acceptable spin speed

range) is allowable, most probably, there will be no need for a spin trim prior to deployment. Since angular momentum is conserved and the spin moment of inertia is increased during deployment, spin speed will nominally decrease to 5.4 ±0.2 rpm when the boom is fully deployed.

Following boom deployment, the orbiter is spun up to the nominal cruise spin speed range of 15 ±0.5 rpm. The cruise spin rate has been selected on the basis of many trade-offs. The initial spinup to 15 rpm reduces the rather large nutation (≈22 deg average) generated by boom deployment approximately by a factor of three. The smaller nutation together with the increase in body nutation frequency render the nutation damper (a small angle, broadly tuned device) nearly twice as effective. At 15 rpm, nutation will be damped to 0.1 deg within 2 hours after spinup. The cruise spin rate also provides sufficient spin "stiffness" to limit attitude errors and residual nutation resulting from subsequent midcourse maneuvers. Thrust pointing error and nutation induced by a continuous axial jet ΔV maneuver (both inversely proportional to spin speed squared) are bounded at 0.7 and 3.2 deg, respectively. Attitude change induced by a pulsed radial jet pair ΔV maneuver (inversely proportional to spin speed) is bounded at 2.4 degrees. The same gyroscopic stiffness limits cruise attitude precession due to solar torque (inversely proportional to spin speed) to ≈0.1 deg/day. The spin rate is limited to 15 rpm during cruise to minimize propellant required for precession maneuvers (proportional to spin speed) while being consistent with minimum gyroscopic stiffness requirements.

The orbiter is maneuvered to the orbit insertion attitude at 15 rpm and is then spun up to 30 rpm for Venus orbit insertion. The object of this maneuver is to increase gyroscopic stiffness and limit insertion errors due to the potentially larger attitude disturbance torques during motor burn. Thrust misalignment and HGA center of mass offset from the +Z axis generate an average thrust pointing error and a coning of the spin

axis about the target ΔV direction (the latter results in a cosine loss of impulse) that are both inversely proportional to spin speed squared. The 30 rpm spin rate "stiffens" the spacecraft by a factor of four relative to cruise and limits the worst case thrust pointing error and spin axis coning to 0.5 and 4 deg, respectively. The 30 rpm also provides the orbiter with an angular momentum comparable to that of many other spinning spacecraft that have utilized similar solid motors.

Despin control is nominally terminated prior to the burn and the HGA is allowed to spin up to 30 rpm. This is to spin average any disturbance torque due to HGA center of mass offset from the +Z axis (nominal thrust direction). If the antenna were despun during burn, this disturbance torque would slew the spin axis and angular momentum vector ≈14 deg, resulting in ≈7 deg of thrust pointing error. When the antenna is spun during burn, the "spin averaged" disturbance torque generates a thrust pointing error and spin axis coning of 0.13 and 0.1 deg, respectively; these errors are included in the worst case total errors discussed above. However, because the antenna is dynamically imbalanced relative to the spun section (rotor) principal axes, spinning the antenna will generate ≈1 deg coning of the +Z axis about the "all spun" spin axis prior to burn; this coning angle must be added to the coning angle generated by the disturbance torques during burn. The spinning antenna approach has been employed on many other spinning spacecraft whose despun platform mass properties were similar to the orbiter HGA.

After orbit insertion, the orbiter is despun to 15 rpm in preparation for precession of the spin axis to the nominal on-orbit attitude (south ecliptic polar direction) and for the initial axial ΔV correction that lowers periapsis altitude. These maneuvers are completed in the vicinity of the initial apoapsis pass and the spin speed can then nominally be reduced to ≈5 rpm for initial scientific observation during the ensuing periapsis pass. The actual spin rate

reduction may depend on the magnitude and direction of the second major axial ΔV correction required to establish the nominal orbit, i.e., a continuous burn at periapsis exit to adjust apoapsis altitude and orbital period. If the forward axial jet were required, a spin speed greater than 5 rpm would be desirable to avoid large attitude error and nutation. It is also possible that jet thrust and alignment biases of the required thruster(s) (which can be calibrated during the initial apoapsis pass) could reduce the spin speed excessively during this maneuver, this requiring a higher initial spin speed. The actual profile of how the orbiter achieves its nominal steady state, on-orbit spin speed at 5 rpm will depend then on the details of initial orbit corrections. It is probable that a near steady state condition will exist at third apoapsis and the spin speed will then be trimmed to 5 rpm as required.

Figure 4.1.1.3-1 illustrates the nominal orbit spin rate history. The phase lock loop (PLL) spin ranges, spin rate detector limits and spin speed range for HGA pointing are also shown in the figure. Since the acceptable spin rate range for magnetometer boom deployment is narrow and the on-orbit spin rate is well defined by scientific considerations, the primary spin rate options are available during cruise orbit insertion and orbit initialization. An increase in the cruise spin rate would make the orbiter more resistant to attitude disturbances but would require more propellant for precession maneuvers; the star sensor would also not operate at its peak performance. An increase in the orbit insertion spin rate would also "stiffen" the orbiter attitude but would require more propellant for the associated spinup and despin maneuvers. The inversion of the spin axis to the on-orbit attitude could be done at a lower spin rate to save propellant; however, any reduction in spin speed must take account of attitude error, nutation and possible spin speed decrease when the potentially large initial orbit corrections are executed.

Spin speed can vary as a result of changes in the spin moment of inertia, cross coupling during thrusting maneuvers, and external disturbance torques. Spin change resulting from magnetometer boom deployment is accounted for in the nominal mission profile. Spin change accruing from the first TCM or initial orbit corrections is expected to be on the order of 2 rpm; the spin change resulting from orbit insertion should be less than 1 rpm. Spin trims during cruise or orbit initialization should be consistent with the nominal profile or an available option. During the steady state orbit phase, spin trims will be made as required to compensate for spin speed variation caused by periodic orbital maintenance maneuvers (<0.5 rpm). Spin speed variation due to atmospheric disturbance torques is expected to be two orders of magnitude smaller.

4.1.1.4 Maneuver Mission Profile. Table 4.1.1.4-1 delineates the chronological sequence of nominal spin speed, attitude and velocity maneuvers during the orbiter mission. Each maneuver has an associated row of tabulated values defining maneuver magnitude, required propellant, maneuver mission time, nominal thruster(s) employed, pulse width selected, number of pulses and execution time. Spin speed maneuvers are nominally executed by continuously firing a pair of radial jet spinup or despin thrusters. Attitude maneuvers are nominally executed by pulsing the axial jet couple at spin frequency. However, at 5 rpm, a single axial jet is pulsed at spin frequency in order to maintain a precession granularity of ≈0.15 deg/pulse in the lowest momentum state; the ΔV associated with this type of maneuver in orbit can be used to trim periapsis altitude. The midcourse velocity maneuvers are assumed to be executed in the vector mode, and the magnitudes of both axial and radial ΔV corrections are assumed maximum and equal, corresponding to the worst case required ΔV magnitude and direction. The maximum available pulse width has been selected for the first and second radial ΔV midcourse corrections

to limit execution time; this results in negligible change in overall impulse efficiency.

The initial orbit trim maneuvers are nominally executed by continuously firing axial jets. The initial periapsis adjustment employs the two aft axials 12 and possibly 36 hours after insertion; two maneuvers may be required depending upon how much periapsis altitude is actually lowered during the first opportunity. The initial orbital period adjustment employs either the forward or aft axial jets 24 and possibly 48 hours after insertion; two maneuvers may be required if the maximum required correction cannot be completed during the first periapsis pass. After orbit initialization, typical orbital maintenance maneuvers employ the aft axial jets in the continuous mode. Optional methods for execution of any one of the nominal set of maneuvers are discussed in Section 4.3.1.

The orbiter is initially loaded with 70 lbs of propellant. Figure 4.1.1.4-1 illustrates remaining propellant versus mission time. Note that prior to orbit insertion, approximately 9 lbs of propellant are consumed. After the orbit has been initially trimmed, a total of ≈25 lbs of propellant will have been consumed. Orbit maintenance maneuvers over 243 days require an additional 26 lbs of propellant so that approximately 18 lbs of propellant are available for calibration maneuvers and contingency planning.

All maneuvers should be executed within a sun look (aspect) angle range of 40 to 110 degrees in order to avoid any limitation in attitude dwell time due to thermal and power considerations. If, in addition, the sun aspect angle is maintained greater than 55 deg, the mid-range sun sensor is always visible and no sensor switching is required. Maneuvers are nominally executed in sunlight so that the sun can always be employed as the spin reference pulse for the PLL and spin rate detector. This is particularly important during long continuous axial jet burns, since a single bit error in the jet select magnitude

TABLE 4.1.1.4-1

ORBITER MANEUVER PROFILE

Maneuver Description	Mission Phase (Cross-Coupling Errors Treated in 4.3.1) Nominal Maneuver: ΔV M/Sec	ΔP Deg.	Δ RPM	Prop. Req. lbs.	Mission Time (Days)	Thruster(s)	Pulse Width (Msec)	No. of Pulses	Execution Time (Δ Hours)
Initial Spinup		9.8	-6.5	0.37	S	R1 · R3			0.008
Spinup II			9.6	0.58	S + 0.2	R1 · R3			0.011
Reor to Cruise Attitude		19		0.18	S + 0.4	A5 · A6	128	124	0.138
First TCM Axial ΔV	5			3.0	I + 5	A5+A6+A7			0.132
First TCM Radial ΔV	5			3.1	L + 5	R3 · R4	512	490	0.544
Second TCM Radial ΔV	0.42			0.26	L + 20	R3 · R4	512	41	0.046
Second TCM Axial ΔV	0.42			0.25	L + 20	A5+A6+A7			0.012
Third TCM Radial ΔV	0.07			0.04	[E] - 20	R3 · R4	128	28	0.031
Third TCM Axial ΔV	0.07			0.04	[E] - 20	A5+A6+A7			0.002
Preinsertion Reorientation		34		0.32	[E] - 2	A5 · A6	128	222	0.247
Spinup for Insertion			15	0.85	[E] - 2	R1 · R3			0.023
Post Insertion Spindown			-15	0.82	[E] - 0.1	R2 · R4			0.022
Reor to On-Orbit Attitude		153		1.25	[E] + 0.5, +1.5	A5 · A6	128	965	1.072
Intial Periapsis Adjustment	24.4			8.77	[E] + 0.5	A6 · A7			0.279
Initial Period Adjustment	13.5			4.88	[E] + 1, [E] - 2	A5			0.309
Despin to 5 rpm			-10	0.54	[E] + 1.5	R2 · R4			0.021
Typical Periapsis Correction	8			2.84	As Required	A6 · A7			0.112
Typical Apoapsis Correction	1			0.36		A6 · A7			0.014
Typical Attitude Correction	0.05	2		0.01		A5+A6+A7	128	12	0.040

NOTE: S – Time of Separation; A – Axial Jet

I – Time of Launch; R – Radial Jet

[E] – Time of Orbit Insertion; – Spinup

command or a logic failure could cause one of the spin thrusters to inadverently fire. Earth look angle is unlimited when both forward and aft omnis are in receive mode. However, the forward omni should be in transmit mode for look angles less than 95 deg while the aft omni should be in transmit mode for look angles greater than 85 degrees. If an attitude maneuver requires crossing either barrier, provision should be made to switch the appropriate omni to the transmit mode in order to avoid loss of telemetry. When the HGA is in both transmit and receive modes, the aft omni receive mode should be religiously protected in order to avoid any loss of command and telemetry for 36 hours due to despin anomaly or anomalous earth look angle change. An insurance command to switch transmit and receive function to the forward omni at some appropriate time after nominal maneuver termination should always be entered prior to a maneuver; this command can always be cleared immediately after the maneuver is completed. Long, continuous burns such as orbit corrections at apoapsis are preferably executed with both omnis in receive mode and with the omni having the best earth lock angle in transmit mode; this practice would insure telemetry coverage of any anomalous behavior and provide command capability for any immediate actions that might be required.

During any maneuver, spin period and sun angle should be the selected attitude measurement words. After the initial spinup maneuver at separation, the spin rate detector should also be enabled for the remainder of the mission. For continuous axial jet burns, the sun gate and PLL loss of lock are enabled so that the maneuver will terminate automatically when the time count reaches zero, the JCE output is switched off, the PLL loses lock or the spin rate range is exceeded. One of the last two conditions will quickly terminate the maneuver in the event a spin thruster is inadvertently fired. For continuous radial jet burns (spin maneuvers), the PLL loss of lock and sun gate are disabled to permit the maneuver to continue as the PLL loses lock and to always provide a spin reference pulse

to the spin rate detector, respectively. Therefore, this type of maneuver will terminate when the time count reaches zero, the JCE output is switched off or the spin rate range is exceeded. For pulsed maneuvers (precession, velocity or spin), the PLL loss of lock and sun gate are enabled and the same shutdown conditions as those for the continuous axial burns apply. These conditions insure accurate thrust direction and no erroneous spin thruster firing. However, a duty cycle detector enabled only in pulse mode can also terminate the maneuver. If the duty cycle exceeds limits dependent on spin speed and thruster pulse width selection, automatic shutdown occurs. This feature basically limits any anomalous pulse widths to less than 60 degrees of the spin cycle and therefore limits the error in the direction of applied jet impulse. The duty cycle detector, which cannot be disabled by ground command, precludes the use of the 512 ms pulse width above 20 rpm.

4.1.2 Disturbances. External forces will perturb spacecraft attitude, spin rate and velocity during the mission. Attitude drift due to solar pressure is discussed in Section 4.1.2.1. Attitude, spin rate and velocity perturbations due to aerodynamic forces arising during periapsis passes at low altitudes are presented in Section 4.1.2.2. Nutation can be induced by external forces (e.g., jet maneuvers) as well as internal forces (e.g., changing the HGA spin rate); nutation is discussed in Section 4.1.2.3. Wobble refers to the resultant up/down motion of the despun HGA beam when the +Z axis is tilted relative to the axis of maximum moment of inertia; the sources and effects of wobble are summarized in Section 4.1.2.4.

4.1.2.1 Solar Disturbance Torque. An external torque due to solar pressure arises when the spacecraft center of pressure (c.p.) and center of mass (c.m.) are not coincident. The magnitude of the torque and therefore the attitude drift rate are proportional to the solar pressure flux and the c.p./c.m. offset. Since solar pressure is inversely proportional to the square of the

distance from the sun, the precession rate of the orbiter at Venus (0.7 Au) would be about twice that at earth (1 Au) for a given c.p./c.m. offset. The spacecraft center of pressure can also change as a consequence of the variable solar pressure surface presented by the high gain antenna as its beam tracks the earth. During cruise, the center of pressure nearly coincides with the center of mass when the sun has an edge view of the HGA; the c.p. is about 15 inches higher (toward the +Z axis) than the c.m. when the sun has a front or rear view of the HGA. After orbit insertion the center of mass is shifted approximately 5 inches toward the +Z axis. Consequently, during on-orbit operations, the c.p. is approximately 5 inches below the c.m. for solar edge views of the HGA, and about 10 inches above the c.m. for solar front or rear views of the HGA. Note that if the antenna were spinning in orbit (as, for example, after the completion of the primary orbit mission), the average c.p./c.m. offset over a spin cycle, and therefore the daily attitude precession rate would be small. A one and a half inch solar skirt has been added to the aft end of the orbiter solar panel substrate to limit attitude drift in this configuration.

Figure 4.1.2.1-1 illustrates timelines of the orbiter solar precession rate during cruise for the HGA despun (earth pointing mode) and all-spun (omni mode) configurations. Positive precession rate indicates that the c.p. is forward of the c.m. and that the direction of precession is given by the right-hand rule with the right thumb pointing from the spacecraft to the sun. The precession rate is near zero for the despun configuration during the cruise interval of 90 ±40 days after launch. During this interval, the sun and earthlines are nearly orthogonal, so that the resultant solar edge view generates a small attitude drift when the variation in the earth look angle is most sensitive to solar disturbance torque. On the other hand, a relatively large attitude drift occurs toward the latter part of cruise near interior conjunction, because of the solar rear view of the HGA and the increased

solar flux. However, the near colinearity of the sun and earthlines results in a fraction of the attitude drift being propagated into earth look angle variation. The precession rate timeline for the all-spun configuration reflects an increase in solar disturbance torque due exclusively to decreasing spacecraft distance from the sun. The average spacecraft c.p./c.m. offset associated with a spinning HGA during cruise is about 8 inches which results in an attitude drift higher than that corresponding to the earth pointing mode, except during the last 30 days of cruise.

The cumulative solar torque precession for the orbiter during cruise in the earth pointing mode is about 6 degrees. Appropriate biasing of the attitude relative to the sunline to take advantage of the entire attitude deadband would limit the number of attitude touchups to two. The cumulative solar torque precession during cruise with the HGA spinning would be about 12 degrees. Attitude biasing would limit the number of corrections to four. Since the earth pointing mode is primary, no more than three attitude touchups should be required during cruise.

Figure 4.1.2.1-2 illustrates timelines of the orbiter solar precession rate on orbit for the HGA despun (earth pointing mode) and all-spun (sleeve dipole backup mode) configurations. Positive precession rate again indicates that the c.p. is forward of the c.m. Note that the forward shift of the c.m. resulting from orbit insertion burn generates a significant negative c.p./c.m. offset for solar edge views of the HGA (30-110 days in orbit). The resulting attitude drift of ~0.1 deg/day would reflect directly into earth look angle change. During the initial forty days in orbit (primary occultation season), however, the c.p./c.m. offset will change sign and the average drift rate over this time interval will be near zero. Therefore, it would be possible to allow the attitude to drift forward and then backward within the attitude deadband without requiring a touchup. After some 130 days in orbit, the sun starts to illuminate

the front of the HGA, and the precession rate increases to a maximum of 0.25 deg/day at the end of the primary mission. The precession rate for the all-spun configuration is shown to be 0.021 deg/day with the addition of the 1.5m solar skirt.

The cumulative solar torque precession for the orbiter during the 243-day period in earth pointing mode is about 25 degrees. If solar torques were the only disturbance, attitude biasing could limit the number of attitude touchups to about seven. However, aerodynamic disturbance torques will come into play when periapsis is lowered during the on-orbit phase. This type of disturbance, whose effect on attitude is discussed in conjunction with solar disturbances in Section 4.1.2.2, can increase the cumulative precession an additional 15 degrees so that several additional corrections will be necessary.

In the all-spun mode, a maximum cumulative attitude precession of 1.5 degrees occurs, if no attitude touchups are made. Therefore, the sleeve dipole is nominally a viable antenna for communications during the extended mission without the need for any attitude maneuvers.

4.1.2.2 Aerodynamic Disturbances. An external (drag) force on the orbiter is generated by spacecraft motion through the sensible atmosphere in the vicinity of periapsis. The magnitude of this force and the resultant velocity loss are proportional to the dynamic pressure. An external torque also arises near periapsis when the spacecraft aerodynamic center of pressure (a.c.p.) and center of mass (c.m.) are not coincident. The magnitude of the torque and therefore the attitude disturbance are proportional to the dynamic pressure and the a.c.p./c.m. offset.

Atmospheric density in the vicinity of periapsis can be approximated by an exponential law, i.e., ρ decreases exponentially with scale height H from a maximum ρ_ρ at periapsis. This exponential

behavior of density implies that almost all of the aerodynamic disturbance occurs in the first three scale heights above periapsis. It can be shown that the height above periapsis h can be approximated by a parabolic function of time from periapsis, t; for the nominal set of orbital parameters, the height above periapsis is observed to increase at the rate of 12.5 km/(minute)^2. With the scale height near periapsis approximately 10 km, the orbiter spends about 1.8 minutes in the first scale height and 3.1 minutes in the first three scale heights. This calculation illustrates why atmospheric disturbances are described as the impulsive type. The relations defining ρ and h can be used to evaluate the average atmospheric density in the first three scale heights to be $1/2\ \rho_\rho$. Since spacecraft velocity is essentially constant in magnitude near periapsis, the average dynamic pressure during a periapsis pass is effectively half the dynamic pressure at periapsis. For the Model I density of 10^{-12} gm/cc and velocity of 10.75 km/sec at a periapsis altitude of 150 km, the average dynamic pressure is calculated to be 0.0005 lbs/ft^2.

The instantaneous drag force is the product of the dynamic pressure, drag coefficient (C_D) and projected area (A) perpendicular to the direction of motion. The latter two variables are functions of spacecraft shape and angle of attack (i.e., the angle between the spacecraft +Z axis and the spacecraft velocity vector). The angle of attack at periapsis is a function of orbit inclination and periapsis latitude. For the nominal inclination of 105 degrees and periapsis latitude of 15 deg (early launch), the angle of attack at periapsis is 21 degrees. The periapsis latitude of 32 deg associated with a late launch results in a 35-deg angle of attack. The angle of attack will vary ±2.5 deg from the periapsis value for either launch date as the spacecraft moves through the first three scale heights above periapsis. Aerodynamic studies indicate that the spacecraft drag coefficient and projected area are approximately constant (i.e., 1.75 and 85 ft^2, respectively) for this range of angles of

attack in the region of interest. Therefore, integration of the instantaneous drag impulse can be approximated by multiplication of average dynamic pressure, $C_D A$ and the time spent in the first three scale heights. For the nominal parameters cited above, the cumulative drag impulse is about 13.8 lb-sec and the resultant velocity loss for a 800-lb. spacecraft is 0.55 ft/second.

Since velocity loss per pass is effectively proportional to periapsis density, it will indicate (decrease) exponentially with decrease (increase) in periapsis altitude. Therefore, it is a useful indicator in the initial lowering of periapsis altitude to a level consistent with spacecraft safety. Moreover, it is a direct measure of periapsis density, whose accuracy is limited by uncertainty in drag coefficient (less than a factor of two).

The aerodynamic center of pressure also depends on the angle of attack. Aerodynamic studies indicate that the a.c.p. is offset aft of the center of mass (toward the -Z direction) for angles of attack greater than 22 degrees (i.e., periapsis latitude greater than 17 degrees). In effect, the exposed surface of the solar panel drum generates a tipping moment greater than that caused by the forward thermal barrier, high gain antenna, and "hatbrim" flange added at the periphery of the forward thermal barrier. The maximum effect a.c.p./c.m. offset is about 2.3 inches for the largest angle of attack (35 deg) at periapsis.

The instantaneous drag moment is the product of the dynamic pressure, drag coefficient, projected area, and a.c.p./c.m. offset (r). Integration of the instantaneous drag angular impulse can be approximated by multiplication of the cumulative drag impulse and the effective a.c.p./c.m. offset. For the worst case offset cited above, the cumulative drag angular impulse is about 2.9 ft-lb-sec; the resultant attitude precession for a spacecraft spinning at 5 rpm is 1.2 degree. The variation of this attitude disturbance with

periapsis latitude (i.e., angle of attack) is illustrated in Figure 4.1.2.2-1. The precession per pass for a periapsis altitude of 150 km is shown as a function of the angle between the antenna boresight axis and the orbit plane with the two periapsis latitude extremes (15 and 32 deg) as parameters. The variation in each curve as a function of boresight angle is due to the variation in frontal area of the antenna dish as seen by the oncoming gas flow, the size and location of the "shadow" cast by the antenna onto the forward thermal barrier, and the resulting movement of the aerodynamic center of pressure. The large difference between curves is caused by the tendency (cited above) of the aerodynamic center of pressure to move aft of the center of mass as the angle of attack increases. Positive precession rate corresponds to an aerodynamic torque which would tend to tip the high gain antenna (positive spin axis) toward the oncoming gas flow if the vehicle were not spinning. Spacecraft spin actually causes the spin axis to precess around the velocity vector.

Since attitude precession per pass is effectively proportional to periapsis density, it too will increase (decrease) exponentially with decrease (increase) in periapsis altitude. Therefore, the observed velocity loss and attitude precession complement one another during the initial periapsis lowering process. Either observation provides a direct measure of periapsis density limited only by uncertainty in the drag coefficient or a.c.p./c.m. offset. The two measurements can be used in conjunction with one another to estimate one of these parameters as well as density.

Figure 4.1.2.2-2 shows the initial and final velocity and sun vectors for the 243 day on orbit phase. Aerodynamic torque tends to precess the spin axis around the velocity vector at periapsis while solar torque tends to precess the spin axis around the sunline. Since the sunline rotates relative to periapsis velocity as Venus transports the spacecraft around the sun, the total solar + aerodynamic precession is a vector

combination of the individual contribution. Solar torque precession has been shown to be about 0.15 deg/day, substantially less than the maximum attitude disturbance. Because of the orbital parameters and sun/Venus/earth geometry during the on-orbit mission, the solar disturbance will generally reduce the aerodynamic disturbance for orbits having periapsis latitudes greater than 17 degrees. Therefore, the maximum attitude disturbance at a 150 km periapsis on a daily basis is about 1 degree.

Aerodynamic torque will also have a small component along the spin axis direction. Similar to the precession torque, the spin torque is a function of angle of attack, tending to be largest for the high periapsis latitude orbits. Aerodynamic studies indicate a maximum spin speed change of 0.003 rpm per pass for the 32-degree periapsis latitude orbit.

4.1.2.3 Nutation. A spinning body will always tend toward a spin about its axis of maximum moment of inertia, because this is the lowest rotational energy state. A "pure spin" equilibrium is characterized by coincidence of the body spin (maximum inertia) axis, angular velocity vector and angular momentum vector. External moments such as those arising during jet maneuvers will usually perturb the "pure spin" state by accumulating angular velocity and angular momentum orthogonal to the body spin axis. Nutation refers to the angular motion of the spinning body under such a perturbation. The resulting nutational motion for a body whose spin axis is an axis of symmetry (i.e., moments of inertia about all axes perpendicular to the spin axis are equal) is illustrated in Figure 4.1.2.3-1. In inertial space, the body spin axis is observed to cone counterclockwise around the angular momentum vector at the inertial nutation rate, ω_n. The latter frequency is the product of the spin-to-transverse moment of inertia ratio, σ (greater than one), and the spin speed, ω_z. Two important observations can be made from study of the figure. The first is that with respect to a body-fixed observer, the angular momentum vector

appears to cone counterclockwise around the body spin axis at inertial nutation frequency minus spin speed (i.e., body nutation frequency, $\Omega = (\sigma-1)\omega_Z$). This is the frequency at which any dampers on the spinning body are excited. The second observation is that any sensor (e.g., sun sensor) which samples the angle between the spin axis and an inertial target direction once a spin period, will detect an amplitude modulation on the measured angle at the difference (body nutation) frequency. Thus, for example, nutation would generate a sinusoidal variation about the average value of the Ψ_2 data type at 20% of spin frequency, if σ were 1.2. An attitude sensor/spin rate detector such as the sun sensor, is therefore, a useful nutation/mass properties sensor on a spinning body.

The Pioneer Venus Orbiter is an asymmetrical body, i.e., moments of inertia about axes perpendicular to the spin axis are unequal. The resulting nutational motion for such a body is somewhat more complicated than the symmetrical case, as illustrated in Figure 4.1.2.3-2. The body spin axis is observed to precess counterclockwise around the angular momentum vector at a nutation angle, θ, which varies between θ_{min} and θ_{max}. The precession rate can still be identified as the inertial nutation rate, ω_n, where the latter frequency is the product of the effective spin-to-transverse moment of inertia σ_e (greater than one) and the spin speed ω_Z. σ_e is defined by

$$\sigma_e = 1 + \sqrt{(\sigma_1 - 1)(\sigma_2 - 1)}$$

where σ_1 is the ratio of the spin to maximum transverse moment of inertia (Iz/Ix), and σ_2 is the ratio of the spin to minimum transverse moment of inertia (Iz/Iy). The nutation angle varies because of a tendency for the nutation angle to increase when the angular momentum vector approaches the plane defined by the axes of spin and maximum transverse moments of inertia. Since the angular momentum vector appears (to a body-fixed observer) to cone about the body spin axis at body nutation frequency (Ω_e

$= (\sigma_e - 1)\omega_Z$), it will approach the plane of interest at twice this frequency. In inertial space, then, the spin axis spirals from θ_{min} to θ_{max} and back to θ_{min} at twice body nutation frequency, while precessing about the angular momentum vector at inertial nutation frequency. The ratio of the maximum to minimum nutation angles is defined by

$$\frac{\theta_{max}}{\theta_{min}} \sqrt{\frac{\sigma - 1}{\frac{2}{\sigma - 1}}}$$

This ratio is approximately 1.8 for the Pioneer Venus Orbiter. The variable nutation angle will tend to distort the amplitude of the sinusoidal variation in sun angle at body nutation frequency, when the sun sensor is sampling at spin frequency; however, the average nutation angle $(\theta_{min} + \theta_{max})/2$ will be readily apparent.

Nutation will be induced by any thrusting maneuver which generates a torque about the spacecraft center of mass that is orthogonal to the spin axis for a noninteger number of body nutation periods. After spinup, the continuous firing of a single axial jet is the major source of nutation. A thruster firing for an integer plus a half nutation period will develop a peak average nutation proportional to the transverse torque and inversely proportional to the spin rotational energy ($1/2\ I_Z \omega_Z^2$) and $(\sigma_e - 1)$. Maximum nutation perturbations during cruise at 15 rpm and on-orbit at 5 rpm are 3° and 18°, respectively. The continuous firing of two axial jets will induce a small nutation due to thrust mismatch (less than 2%). A small nutation (<0.25 deg) will also be induced by continuous firings of radial thrusters during spin trim maneuvers.

Axial and radial thrusters will always be pulsed for a small fraction of the body nutation period. As a result, each pulse will induce a change in the nutation angle equivalent to the precession (angular movement of the angular momentum vector) per pulse. However, since the pulsing

frequencies (spin speed and twice spin speed) are asynchronous with body nutation frequency Ω_e, nutation will alternately increase and decrease. The maximum nutation, θ_p, resulting from a pulsing maneuver at frequency Wp which generates a precession per pulse, α_p, is given by

$$\theta_p = \frac{\alpha_p}{\sin\left[\pi\left(\frac{\Omega e}{Wp}\right)\right]}$$

Precession maneuvers which pulse axial jets at spin frequency will generate the maximum nutation disturbance associated with pulsed maneuvers. θ_p is approximately 70% greater than the precession per pulse.

Nutation will also be induced when the high gain antenna is despun (despin acquisition) or allowed to spin up with the spacecraft (friction spinup). The HGA is dynamically imbalanced with respect to spacecraft principal axes. When the HGA is despun, the imbalance has no effect on spacecraft spin and the +Z axis is the nominal spin axis (axis of maximum moment of inertia). When the HGA is spinning with the spacecraft, this imbalance causes the nominal spin axis to tilt approximately 1 degree from the +Z axis. During any transition (despin acquisition or friction spinup) between these two states (despun or spinning HGA), the new spin axis is not perfectly aligned to the constant angular momentum vector. Some momentum orthogonal to the spin axis remains resulting in residual nutation (less than 1 deg).

Nutation is damped by employing onboard movable devices that dissipate energy associated with the transverse angular rates so that the principal axis of maximum moment of inertia can become coincident with the momentum (constant in the absence of external torque). The spin speed will always increase slightly during the passive damping process, consistent with both conservation of angular momentum and the lowest

possible energy state for "pure spin". The analytical method, corroborated by many in-orbit observations, for describing the converging character of the nutation angle is the time constant approximation. The basic assumption is that the rate of nutation angle change is proportional to nutation angle; the resulting exponential decay of nutation angle is described via the time constant (the reciprocal of the proportionality constant). Nutation dampers will generate different rates of decay over different regions of nutation angle; time constant, therefore, are identified with specific ranges of nutation.

The primary method for onboard nutation damping is provided by the freon tube damper described in 3.3.2.6. The damping performance of this device is a function of spin speed, mass properties, temperature, and nutation angle level. Figure 3.3.2.6-1 illustrates its performance over spin speed and temperature for small nutation angles (<1°). Table 3.3.2.6-1 defines damper performance for key mission events and their associated spin speed and nutation levels.

There is an auxiliary method for nutation damping provided by the despin control system. Figure 4.1.2.3-3 illustrates how despin control is coupled to nutation dynamics. Despin control will always apply a torque to the despun section in response to position and rate errors. In the presence of nutation, the angular acceleration orthogonal to the spin axis acts on the dynamically imbalanced despun section to generate a sinusoidal azimuth torque disturbance at nutation frequency. Despin control will respond by commanding a sinusoidal torque whose amplitude and phase depend upon its frequency response. The total azimuth torque, then, will tend to rock the despun section at nutation frequency. The resulting azimuth angular acceleration acts on the dynamically imbalanced despun section to generate a sinusoidal transverse torque disturbance at nutation frequency. This torque will tend to reduce the angular velocity component orthogonal to the spin axis when the

despin control transfer function is properly synthesized. Table 4.1.2.3-1 defines the nutation damping time constants associated with despin control for 5 rpm (on-orbit), 15 rpm (cruise), 30 rpm (orbit insertion) and two nutation angle levels, 1° and 5°. Note that this auxiliary damping is (1) significant at spin speeds greater than 15 rpm and (2) effective at nutation angels as large as 5°. These properties can be exploited after magnetometer boom deployment/spinup to provide effective damping at large nutation angles, a condition where the primary damper is somewhat weak. Table 4.1.2.3-1 also defines nutation damping time constant when despin control is deactivated (despin section in spinning). Note from Figure 4.1.2.3-3 that even if the despin command torque is not present, the presence of friction torque will introduce some phase change in the dynamic reaction torques, resulting in some nutation damping. This effect is most significant at high spin rates and small nutation angles, a condition corresponding to post-orbit insertion motor burn.

4.1.2.4 Wobble. The spacecraft Z axis and the antenna despin axis are nominally aligned along the axis of maximum moment of inertia. In the absence of nutation, then, every point on the spun section rotates about the Z axis and the despun HGA beam is stationary in inertial space. Uncertainties in the spacecraft alignment and balancing processes cause the true spin axis (axis of maximum moment of inertia) to be tilted with respect to the spacecraft Z axis. Consequently, every point on the spun section rotates about the tilted spin axis. This implies that the Z axis and the antenna despin axis cone about the true spin axis at the tilt angle. Since the HGA beam is despun, it appears to pitch up and down from the earthline at spin frequency as shown in Figure 4.1.2.4-1. This motion of the antenna beam is referred to as wobble.

The wobble or spin axis tilt angle is proportional to the spacecraft residual dynamic imbalance and inversely proportional to the difference between the spin and transverse

TABLE 4.1.2.3-1

DESPIN CONTROL/FRICTION NUTATION DAMPING PERFORMANCE

Mode	Spin Speed (rpm)	Nutation Damping Time Constant (Min.)	
		1^{o} Nutation Angle	5^{o} Nutation Angle
DCE ON, Position Hold	5	840	900
	15	13.7	16.4
	30	3.9	6.0
DCE OFF, Friction Only	5	363	287
	15	162	199
	30	18.3	37.2

moments of inertia. Since the Pioneer Venus Orbiter is an asymmetrical body, the wobble angle would be largest if the dynamic imbalance were located in the plane defined by the spin and maximum transverse moments of inertia. The dynamic imbalance component due to each error source has been assumed to lie in the worst plane in order to conservatively bound the total wobble angle error. Table 4.1.2.4-1 lists the various uncertainty errors and their contribution to the wobble angle before and after the Venus orbit insertion maneuver (VOIM). The residuals associated with the deployed magnetometer boom are the dominant contributor to orbiter wobble angle error. The weight and center of gravity of the boom have been carefully measured in order to limit the wobble angle due to these error sources.

There are systematic error sources for the orbiter wobble angle that are also listed in Table 4.1.2.4-1. The cap on the neutral mass spectrometer is nominally ejected after VOI. The spacecraft will be nominally balanced for the on-orbit configuration, so that a systematic 0.34 deg tilt of the true spin axis from the +Z axis (toward $\theta \approx 150$ deg) will be present during cruise. The electron temperature probe antenna is deployed after VOIM, causing an additional small tilt ($\approx$0.01 deg) during cruise. Two movable instruments on the orbiter, the radar mapper and cloud photopolarimeter, will cause mass relocation and thus wobble during their operation in orbit; the magnitude and direction of spin axis tilt during their respective opeations is known and can therefore be eliminated from, for example, occultation data, if necessary.

The total wobble angle due to systematic and non-systematic errors in the dynamic balance of a rigid spacecraft is indicated in Table 4.1.2.4-1. There are two non-rigid parts on the spacecraft, namely, hydrazine propellant and nutation damper fluid, that can increase the wobble. Any spin axis tilt toward one hydrazine tank would cause fluid to migrate to the other tank in order to

TABLE 4.1.2.4-1

ORBITER WOBBLE ANGLE SUMMARY
(DESPUN ANTENNA)

Contributor		Wobble Angle (Deg)	
		Pre-VOIM	Post-VOIM
Uncertainty Errors			
1.	Alignment to Spin Balance Machine	±0.014	±0.008
2.	Spin Balance Machine Residual	±0.003	±0.002
3.	Orbit Insertion Motor Residuals	±0.054	±0.038
4.	Propellant Tank/Lines Residuals	±0.006	±0.004
5.	Magnetometer Boom Residuals	±0.136	±0.078
6.	OETP Boom Residual	±0.005	±0.002
7.	Thermal Distortion	±0.03	±0.03
	RSS Uncertainty	±0.15	±0.09
Measurable Errors			
1.	ONMS Cap (Ejected After VOIM)	0.34	-
2.	OETP Radial Antenna (Deployed After VOIM)	0.01	-
3.	Radar Mapper (Movable)	-	±0.06
4.	OCPP Telescope (Movable)	-	TBD
	Total Rigid Body Wobble	0.34 ±0.15	±0.06 ±0.09
	Worst Case Amplification Factor	1.12	1.12
	Maximum Expected Wobble Angle*	0.55	0.17

*Whenever the antenna is allowed to spinup with the spun section, antenna imbalance causes an additional wobble angle contribution of ≈0.5 degree.

keep the free surface in each tank equidistant from the spin axis. The result of this migration is to amplify the wobble component in the plane containing the tanks and the Z axis. Studies indicate that a worst case amplification of 1.083 occurs at a propellant fill fraction of 33%. Similarly, any spin axis tilt toward the nutation damper would cause damper fluid to move such that the free surface is always parallel to the spin axis. The readjustment of damper fluid amplifies the wobble component in the plane containing the damper and the Z axis; the worst case amplification is 1.024. When both hydrazine and damper fluid are present on-board, the dispalcement of one fluid is coupled to the dispalcement of the other. Studies indicate that the worst case resultant amplification factor due to both fluids is 1.12. Table 4.1.2.4-1 indicates that the maximum expected wobble angles before and after VOI are 0.55 and 0.17 deg, respectively. Whenever the antenna is allowed to spinup with the spun section, antenna imbalance would increase these maximum wobble angles by ≈0.5 deg.

As discussed previously, the primary effect of spacecraft wobble is to nod the HGA antenna beam up and down from the earline. This effect has been minimized for the on-orbit configuration by intentionally balancing the spacecraft without the ONMS cap. As a consequence, the HGA beam will nod approximately ±0.5 deg during cruise. Wobble will also affect sun and star sensor alignment parameters; calibration of biases in these parameters due to spin axis tilt will be necessary in order to accurately define attitude. Wobble will also affect axial jet alignment, causing equal and opposite spin biases on the two axial jets when using the ΔV mode.

There are several times in the mission where wobble has a transitory nature. Wobble increase after antenna spinup has already been discussed above. The orbiter is initially launched with the magnetometer boom stowed; consequently a large wobble angle, ≈9 deg, is present after initial spinup and prior to boom deployment. ΔV

maneuvers may temporarily draw propellant asymmetrically from one tank causing momentary wobble changes of at most 1 deg; after redistribution of propellant (less than 1 hour), only the fuel amplication effect discussed above will be present.

4.1.3 Mission Dynamics. The dynamics aspects of several key points in the Mission is discussed below.

4.1.3.1 Launch Vehicle Separation/Spacecraft Spinup Dynamics. Spacecraft separation occurs when the Centaur sequencer fires the two explosive bolts releasing the V-band clamp which holds the spacecraft to the attach fitting. When the attach fitting and spacecraft physically separate, two redundant separation switches are activated, each independently initiating one of the two command memories that contains the spin-up sequence of commands. The spacecraft clears the attach fitting 1.80 seconds after separation (T_S + 1.80 sec), spinup jet ignition follows at T_S + 3.38 seconds, and spinup to 6.5 rpm is completed at T_S +28.88 seconds.

The center of mass (c.m.) of the Orbiter spacecraft is laterally offset by about one inch from the Z axis with the magnetometer boom stowed. As a consequence, the axial force exerted by the springs during separation gives rise to a torque tending to rotate the tip of the aft omni antenna, which points parallel to the -Z axis of the spacecraft and is located in the Z-Y plane, toward the adapter. However, the lateral force of each spring (arising from the axial compression) is selectively aligned to provide a torque counteracting the introduced by the axial forces. With the separation springs rotationally aligned, the deterministic portion of the spacecraft tip-off rate is reduced in magnitude to 0.29 deg/second. The tip-off rate indicated in the chart is exclusive of the Centaur rate prior to separation, estimated at 0.76 deg/sec (RSS for pitch and yaw).

With the orbiter separated, the initial spinup maneuver begins at T_S + 3.38 seconds. During

spinup, the spacecraft momentum vector will be under the influence of deterministic torque perpendicular to the nominal spin direction. These torques occur primarily because the orbiter axis of maximum inertia (the principal axis of inertia or true spin axis) is tilted 9° from the geometric spin axis (spacecraft +Z axis) with the magnetometer boom stowed. Since the spinup torque is applied essentially along the geometric axis, angular momentum will be added to the vehicle in a direction perpendicular to the principal axis. This results in a deterministic attitude displacement of the spacecraft momentum vector, and a residual spacecraft rate transverse to the principal axis (nutation condition) subsequent to the maneuver.

The deterministic transverse torques present during spinup also result from the independent canting of the two pairs of radial jets to fire through system c.m. locations at different times in the mission. As shown in Figure 3.4.1-3, jets R3 and R4 are aimed at the spacecraft c.m. location during cruise mode to minimize precession during the nominal ΔV maneuvers. The other pair of jets (R1 and R2) is aimed at the post orbit-insertion center of mass to minimize precession should any ΔV maneuvers be called for in that phase of the mission. Since one of each pair of jets (R1 and R3) is used during initial spin and since the system c.m. does not shift axially during boom deployment, it is primarily the alignment of the post-OIM c.m. oriented jet (R1) which gives rise to a torque perpendicular to the geometric axis. This torque adds to the transverse torque already present in principal coordinates due to the boom in stowed position. In assessing the trade-off of pointing this second pair of radial jets at the post-orbit insertion c.m. rather than the cruise c.m., the following should be noted. The deterministic attitude tilt of spacecraft momentum vector which can be biased out prior to spacecraft separation would be present due to the principal axis tilt even if the jets were co-aligned. Furthermore, the radial jet pair which is not aimed at the current c.m. location could be used for precision

attitude control or as a backup attitude precession mode. The deterministic transverse torque introduced by the jet geometry and principal axis tilt is shown in Figure 4.1.3.1-1. The deterministic tip-off rate, which contributes an additional tilt of the momentum vector during spinup is also shown.

The transverse torque causes a deterministic attitude displacement of the spacecraft momentum vector toward the initial direction of the total torque. This dynamic phenomena can be understood by noting that for a body whose ratio of roll inertia to pitch inertia is near one (1), the principal axis tends to cone about the instantaneous angular momentum vector in inertial space at nearly the same rate at which the body spins about its principal axis. Therefore, the total torque exerted on such a body tends to be fixed in inertial space and momentum tends to accumulate along the initial direction of the total torque. In addition, the momentum vector will tend to tilt due to the presence of the tip-off rate in an inertial direction roughly perpendicular to and trailing the direction of the rate itself. Superposition of these two effects leads to a deterministic estimate of the momentum vector attitude tilt during initial spinup.

The results of this analysis are seen to yield a net momentum vector tilt of 9.8 degrees as shown in the review of spinup performance given in Table 4.1.3.1-1. A priori knowledge of the magnitude and direction of this motion enables the Centaur vehicle to be used to bias the spacecraft attitude prior to separation so that the deterministic effects experienced during separation and initial spinup will position the average attitude vector to a desired atttitude. In this way, with deterministic errors combined with Centaur reorientation, the final attitude uncertainty of 8.9 degrees is seen to result purely from random sources.

The desired average attitude vector pointing for the orbiter spacecraft following spinup is in the

general direction of the north ecliptic pole, but with a sun angle less than 80 degrees for magnetometer boom deployment thermal control. Taking into account a possible 8.9 degree random tilt of the average attitude vector, the spacecraft Z axis must be targeted for a tilt angle of 18.9 degrees (10 degrees bias plus 8.9 degrees random error) to satisfy the thermal constraint. Looking down at the spacecraft average atttitude vector from the north ecliptic pole (Figure 4.1.3.1-2) it may be seen that the spacecraft +Z axis (which initially points up at the viewer) must be tilted down toward the plane of the drawing to achieve an 80 degree sun angle. To account for all possible random errors, the magnitude of the tilt must be 18.9 degrees. However, advantage can be taken of the deterministic nature of the 9.8 degree shift during spinup to reorient the spacecraft as close as 9.1 degrees to the north ecliptic pole before separation. As long as the reorientation maneuver both tilts the spacecraft Z axis 9.1 degrees toward the sunline and orients the X axis of the spacecraft so that the deterministic separation and spinup disturbances will move the momentum vector an additional 9.8 degrees toward the sun, the post spinup attitude will still meet the thermal requirement even if the tilt due to random effects is directly away from the sun.

The reorientation maneuver is related to the inertial location of the sun at the time of separation. For a given launch day, the Centaur must place the spacecraft coordinates as shown in Table 4.1.3.1-2. The X axis longitude which is 39.1 degrees on May 16, increases 1 deg/day during the launch opportunity as does the Z axis longitude.

4.1.3.2 Magnetometer Boom Deployment Dynamics. The magnetometer boom is deployed by ground command. Upon command, pin pullers release the caged boom assembly, and centrifugal force developed by the orbiter's spin drives the three hinged boom segments (S-fold) outward. Figure 4.1.3.2-1 illustrates how the boom deploys relative to spacecraft coordinates. Detent pins at the outer

TABLE 4.1.3.1-1

ORBITER SPINUP PERFORMANCE SUMMARY

Error Source	Attitude Error (Deg)	Residual Nutation At End of Maneuver (Deg)
1. Transverse Torque (Deterministic, 14% of Spin Torque)	9.8	10.3
2. Tip-Off Rate (Deterministic, 0.29°/Sec)		
(Random, 0.76°/Sec)	7.4	0.2
Resultant*	7.4	10.3
3. Attitude Uncertainty of Centaur	5.0	-
TOTAL	8.9	10.3

*Deterministic Attitude Error (9.82°) combined with Centaur Reorientation and roll control to satisfy sun angle constraint ($\leq 80^{\circ}$) for Boom Deployment.

TABLE 4.1.3.1-2

CENTAUR REORIENTATION REQUIREMENT

	Celestial Latitude (Degrees)	Celestial Longitude (Degrees)
Spacecraft Z-Axis Direction	80.9	θ_S
Spacecraft X-Axis Direction	-8.7	θ_S -16.9

θ_S - Celestial longitude of the spacecraft sunline at the time of the maneuver.

two hinges lock the boom in a linear configuration about 3 seconds after command initiation. The boom continues to rotate relative to the spacecraft for another 7 seconds until the root hinge detent pin locks the boom along a spacecraft radial.

Since the vehicle spin moment of inertia is larger after deployment, the resulting excess rotational energy in the deployed configuration must be either dissipated during deployment or transferred to boom vibration and spacecraft nutation. During boom deployment, a fraction of this excess energy is dissipated by friction torque at each hinge, as boom segments move relative to one another. At the time of any hinge latching, another fraction of the excess energy is available to be stored and dissipated by boom flexing and structural damping. The latching energy at each hinge is a measure of the peak bending moment developed to inhibit any further relative motion between adjacent boom segments. After all hinges have latched and boom vibrations have terminated, any residual excess energy manifests itself in spacecraft nutation.

The spin speed limits prior to deployment have been selected to (1) limit latching energy at each hinge consistent with structural integrity of the boom in the event hinge drag torques are negligible, and (2) to provide sufficient latching energy at each hinge in the event hinge drag torques realize their worst case values. Figure 4.1.3.2-2A illustrates the effect of high initial spin speed (6.8 rpm) and no drag torque upon boom link angle history and total system energy during deployment. The middle hinge latches at about 2.5 seconds with the outer hinge following suit one-half second later. The boom continues to swing about the root hinge until latch occurs some 10 seconds after deployment initiation. The total system energy is seen to step down at each hinge latch with hinge damping ($\approx$2%) rapidly dissipating the vibrational energy. Since no friction torque is assumed to act, system energy is constant between any two latches. The peak bending moments experienced at

the root, middle and outer hinges are 165, 230 and 45 ft-lbs, respectively. These moments are at least 30% less than the ultimate stress levels of 233, 554 and 106 ft-lbs, respectively. Since bending moments experienced at any hinge scale approximately with spin speed squared, the ultimate stress level at the root hinge would be exceeded for initial spin speeds greater than 8 rpm. A spin speed upper bound of 6.8 rpm has been selected to insure conservative stress margin at the root hinge.

Figure 4.1.3.2-2B illustrates the effect of low initial spin speed (5 rpm) and the maximum drag torque upon boom link angle history and total system energy during deployment. The middle hinge latches at about 5 seconds. The outer hinge remains quasi-stationary about 20 degrees from latchup for another 13 seconds until the root hinge latch action triggers the outer hinge latchup. The total system energy is seen to decrease significantly between latches as a result of the worst case hinge friction torques. Latch energies and therefore peak bending moments are substantially smaller than those associated with deployment at the high initial spin speed. Although a spin speed lower bound of 5 rpm appears acceptable, a nominal spin speed range of 6.5 ±0.3 rpm has been selected to (1) insure positive latching action consistent with adequate stress margins and (2) to nominally deploy the boom after confirmation of the spin rate at acquisiton, without requiring any spin trim. The variations in the total deployment time and peak bending moments are one second and 20%, respectively, over this recommended initial spin speed range.

Prior to boom deployment, the spacecraft +Z axis is tilted a maximum of 8.6° from the axis of maximum moment of inertia (the nominal direction of the angular momentum vector). Consequently, the +Z axis appears to cone about the angular momentum vector (initial attitude) at spin frequency. The deployment process establishes the +Z axis as the new axis of maximum moment of inertia while conserving the magnitude and

direction of angular momentum. Alignment of the new spin axis (+Z) to the constant angular momentum vector does not occur, however, and a large residual nutation results (the energy associated with nutation is the excess rotational energy that has not been dissipated, as discussed previously). Figure 4.1.3.2-3A illustrates the inertial trace of the +Z axis on a plane normal to the angular momentum direction (spacecraft attitude prior to and after deployment). The dotted circle describes the coning path of the +Z axis prior to deployment. T_1 and T_2 define the start and end points of the +Z axis path during deployment. The +Z path between T_3 and T_4 describes nutational coning of the new spin axis about the angular momentum vector at an average angle of 22 degrees. The actual nutation angle varies between 14 and 30 degrees due to asymmetrical moments of inertia (see Section 4.1.2.3); in the absence of nutation damping, the +Z axis would continue to cone about the angular momentum vector at inertial nutation frequency, while spiralling in and out between the nutation limits of 14 and 30 degrees.

The spin rate will decrease to about 5 rpm because of the larger spin moment of inertia when the boom is deployed. The low spin rate together with the large nutation significantly degrade nutation damper performance. The spinup to 15 rpm following boom deployment reduces the average nutation angle from 22 to 7 degrees. Figure 4.1.3.2-3B illustrates the reduction in nutation resulting from a spinup process initiated at time T_3. The higher spin speed also generates a nutation excitation frequency closer to the damper natural frequency. This tuning results in a nutation damping time constant <30 minutes for nutation less than about 12 degrees. The residual nutation will be <0.1 degree within two hours of the secondary spinup.

4.1.3.3 Orbit Insertion Dynamics. During the Venus Orbit Insertion (VOI) burn of approximately 25 seconds, the angular motion of the spinning spacecraft will be perturbed due to thrust misalignment and variable mass properties of the vehicle.

Therefore, the spacecraft thrust (+Z) axis will precess about the intended direction for the ΔV addition as shown in Figure 4.1.3.3-1. As a consequence, the vehicle will experience a cosine loss in thrust magnitude and an offset in the direction of the ΔV delivered.

Prior to orbit insertion, the Orbiter is spun up to 30 rpm and the despun platform (HGA) is allowed to spin up. A 1.22 degree wobble is present as a result of the dynamic imbalance of the spinning platform and a maximum 0.67 inch offset of the platform center of mass (c.m.) from the geometric spin axis (spacecraft +Z axis). In this condition, the spacecraft geometric axis, or nominal thrust axis, cones about the angular momentum vector which is (in the absence of attitude determination error) the nominal direction for the insertion. Consequently, some small portion (0.02%) of the ΔV delivered during the burn would be lost due to the inertial coning motion of the thrust axis.

If the platform were not spinning, the torque arising from the platform center of mass offset would tend to cause a direct slewing of the angular momentum vector. The magnitude of this potential slewing is approximately 14° or, equivalent, a 7° ΔV directional error. It is to average the effect of this disturbance that the platform is inertially spun prior to burn. The spinning of the platform reduces the ΔV directional error to approximately 0.15°.

During the burn, the angular motion of the spacecraft will also be disturbed by a combination of thrust misalignment and variable mass properties, the latter resulting from depletion of fuel, relative motion of the imbalanced platform, and flexing of the 15.7-foot magnetometer boom. The qualitative and quantitative action of these effects, acting separately and in concert, have been determined both individually and in combination where spacecraft mass geometry was arranged to yield worst case disturbance effects. The results of this worst case simulation are given in Figures

4.1.3.3-2 and 4.1.3.3-3 and Table 4.1.3.3-1 summarizes the individual effects.

Considering the perturbation effects separately, the 0.14° (RSS) thrust misalignment gives rise to a disturbance torque perpendicular to the angular momentum vector which rotates in inertial space at rotor spin frequency. The platform center of mass offset results in a system center of mass offset which also leads to a transverse torque, only this time rotating at platform inertial rate. (at the start of the burn, the vehicle is all-spinning but relative rate may develop from the variation of coulomb and viscous friction experienced at the rotary joint as thrusting proceeds).

Torques perpendicular to the angular momentum vector of the spacecraft tend to precess the vector toward the disturbance torque. Since the torque is rotating in inertial space, the angular momentum vector undergoes a conical precession. This phenomena can be seen in viewing the inertial tilt of the momentum vector (trace (b) of Figure 4.1.3.3-2) where the spin frequency action of the transverse disturbance torque is evident. The momentum vector represents the average direction of the thrust axis so the average tilt of this vector is similar to the velocity added angle. The disturbance torques also introduce nutation and, therefore, a cosine loss due to the coning of the +Z axis about the momentum vector.

The motion of the magnetometer boom may be seen in trace (a) of Figure 4.1.3.3-3. Once the initial vibrations of the boom have settled out, the boom assumes an offset position of approximately 4° axial displacement in response to the 5g acceleration experienced by the system center of mass. This causes the +Z axis to tilt further from the principal axis of inertia, i.e., momentum vector at start of burn. In effect, the wobble angle of the spacecraft increases and the coning motion of the thrust axis about the initial attitude is amplified. The trace of the wobble angle in Figure 4.1.3.3-2 shows that the

wobble angle, which starts at 1.25°, increases to approximately 2° during the burn.

The key parameter pictured on Figure 4.1.3.3-2 is the coning of the nominal thrust axis (spacecraft +Z) about the desired ΔV direction. The coning of this vector leads directly to the ΔV cosine loss. This trace is seen to exhibit all of the attitude perturbations discussed previously, namely, the spin frequency wobble due to the platform center of mass offset; the wobble amplification due to the boom bending; the conical precession, at spin frequency, of the momentum vector due to the disturbance torques; and the sinusoidal variation, at body nutation frequency, of the amplitude of the tilt between the principal axis and the momentum vector. This last effect, which is actually nutation, is also a result of the action of the disturbance torque.

In reviewing the qualitative results of the simualtion study presented in Table 4.1.3.3-1, it can be seen that the figures recorded for the worst case simulation do indeed reflect the cumulative sum of the separate effects listed at the left of the table. The figures of merit used to assess the disturbances are the velocity added angle, or tilt of the cumulative velocity increment in inertial coordinates, and the cosine loss experienced in the magnitude of the delivered ΔV. This cosine loss may be interpreted as a reflection of the average geometric spin axis tilt during the burn since it is this tilt which causes part of the ΔV to be dissipated in the transverse direction during the burn. The cosine loss is arrived at by taking the cosine of the quotient of the actual ΔV over the nominal ΔV.

The tabulated values indicate that the primary effect of the boom on insertion performance is a cosine loss of 0.97° with virtually no velocity added angle. The thrust misalignment and platform center of mass offset introduce both thrust pointing error and cosine loss, the latter to a lesser degree. The thrust misalignment

contributes more than half of the total values recorded in the worst case run.

Table 4.1.3.3-2 reviews the overall orbit insertion accuracy. The total pointing error (RSS) amounts to approximately 0.71° with the preburn attitude uncertainty and spacecraft burn dynamics contributing equally. The cosine loss in impulse due to spacecraft dynamics is TBD with a 0.6% uncertainty in the total impulse delivered by the motor.

TABLE 4.1.3.3-1

VENUS ORBIT INSERTION BURN PERFORMANCE

		Cosine Loss (-)	Velocity Added Angle (Degrees)
Parameter Runs	2.75 Hz, 2% Damping Boom Flexing	0.97	0.01
	0.14° Thrust Misalignment	3.75	0.42
	1.2° Wobble resulting from platform static plus dynamic imbalance	TBD	0.13
Worst Case	All three effects combined to give worst response.	TBD	0.51

TABLE 4.1.3.3-2

ORBIT INSERTION ACCURACY

Error Source	Pointing Error Degree	Impulse Percent
Attitude Determination	0.5	-
VOIM Impulse	-	±0.6
Spacecraft Dynamics*	0.51	TBD
	0.714	TBD ±0.6

*Includes:

- Thrust misalignment ~0.14 Deg.
- Platform C.M. offset ~0.671 Inch.
- Boom Bending at 2.75 Hz, 2% Damping.

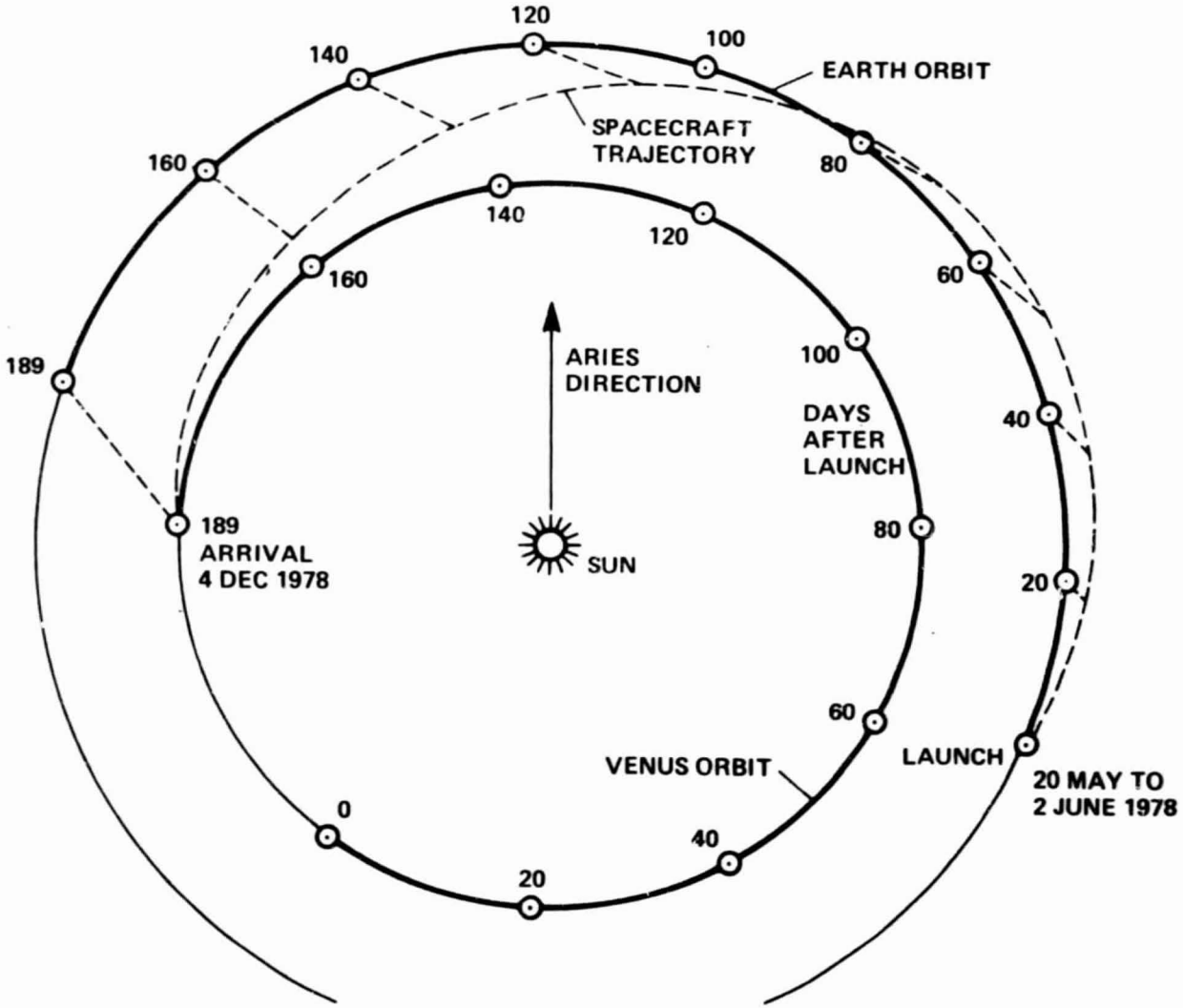


Figure 4.1.1.1-1. North Ecliptic Polar View of 1978 Type II Orbiter Earth-Venus Trajectory

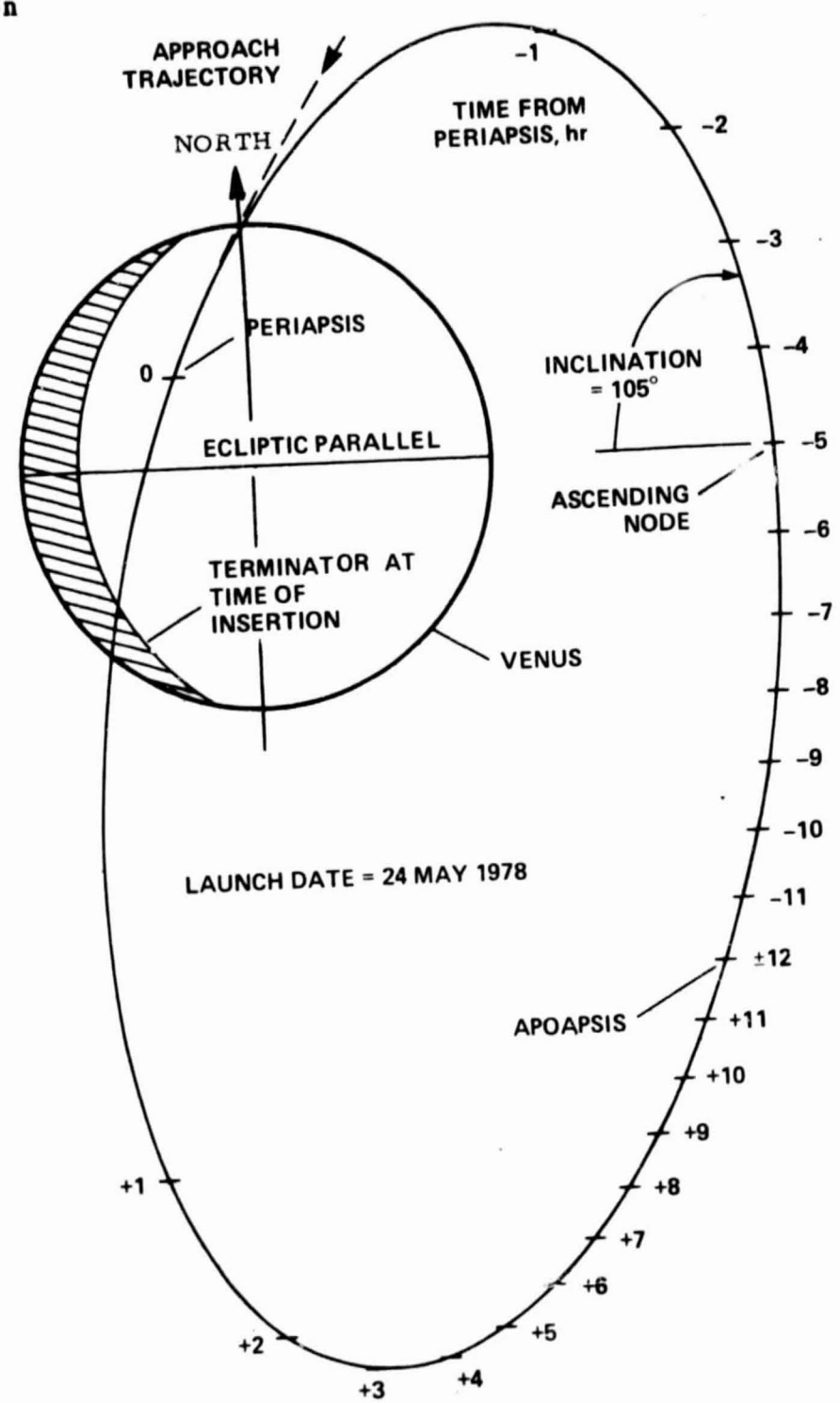


Figure 4.1.1.1-2. Perspective View of Baseline Orbit For Pioneer Venus Orbiter.

Revision

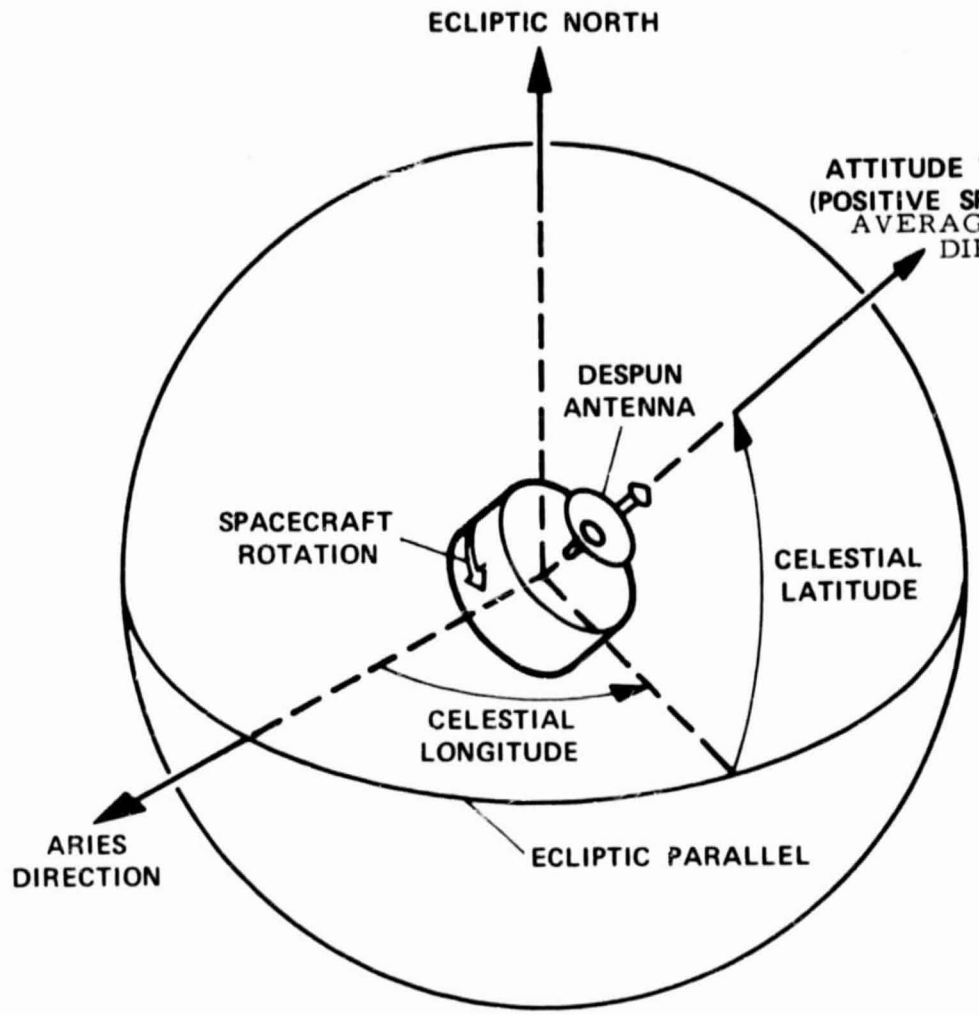
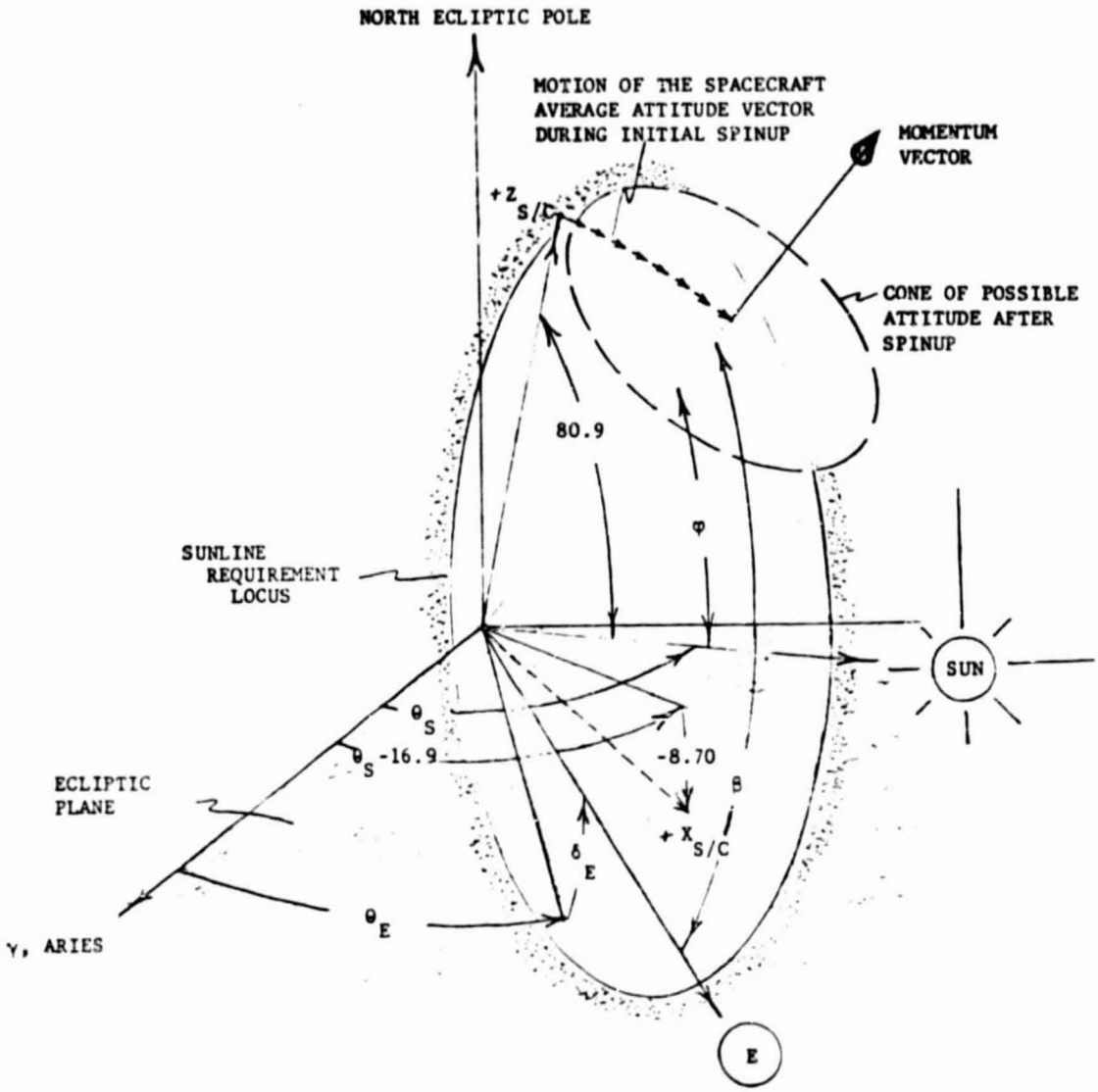


Figure 4.1.1.2-1. Representation of Spacecraft Attitude.



β - AVERAGE EARTH LOOK ANGLE

φ - AVERAGE SUN LOOK ANGLE

Figure 4.1.1.2-2. Separation, Spinup and Initial Acquisition Geometry

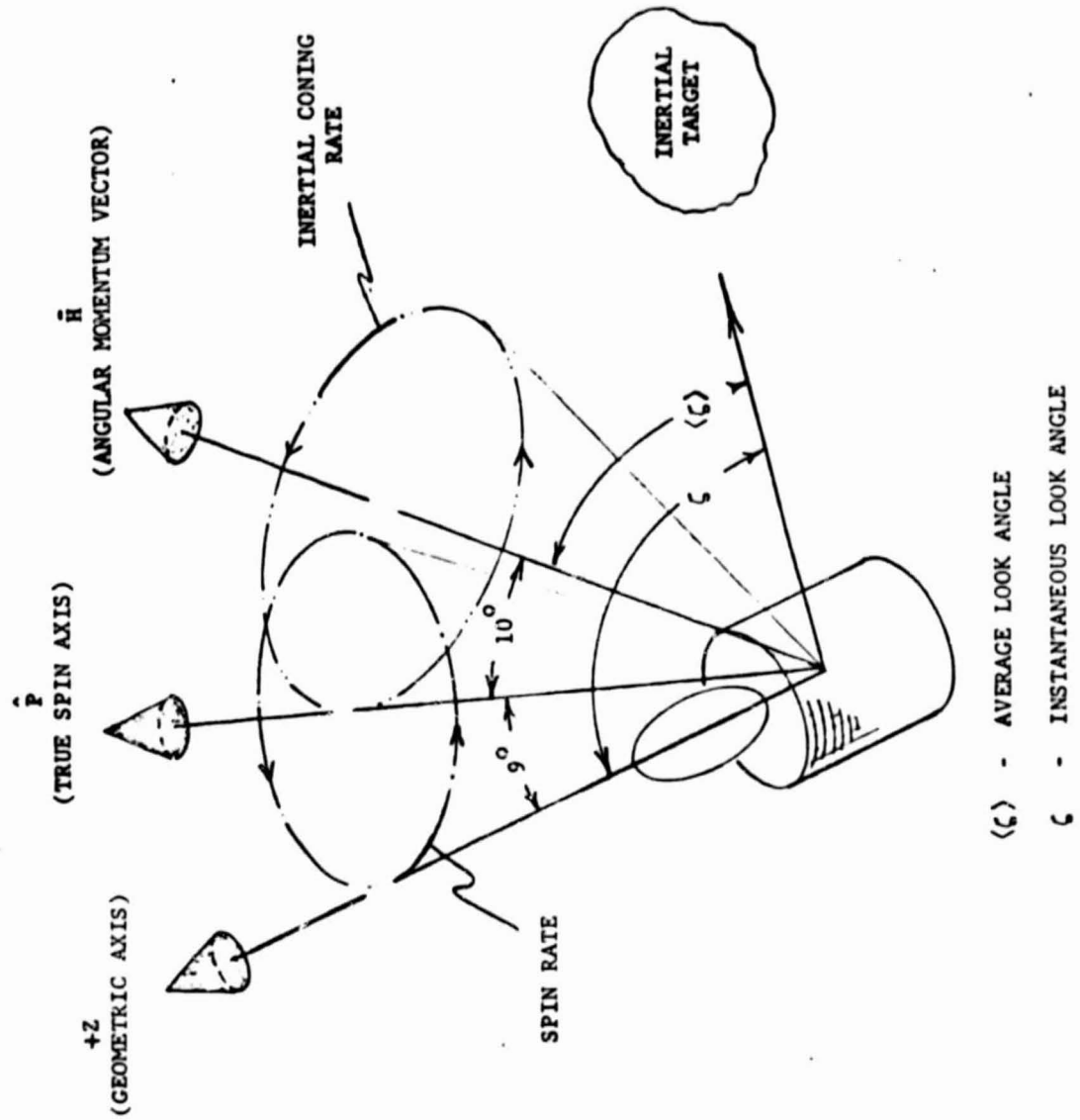


Figure 4.1.1.2-3. Motion of S/C +Z Axis and Spin Axis in the Presence of Nutation, Including Effect of Spin Axis Tilt from the S/C +Z Axis

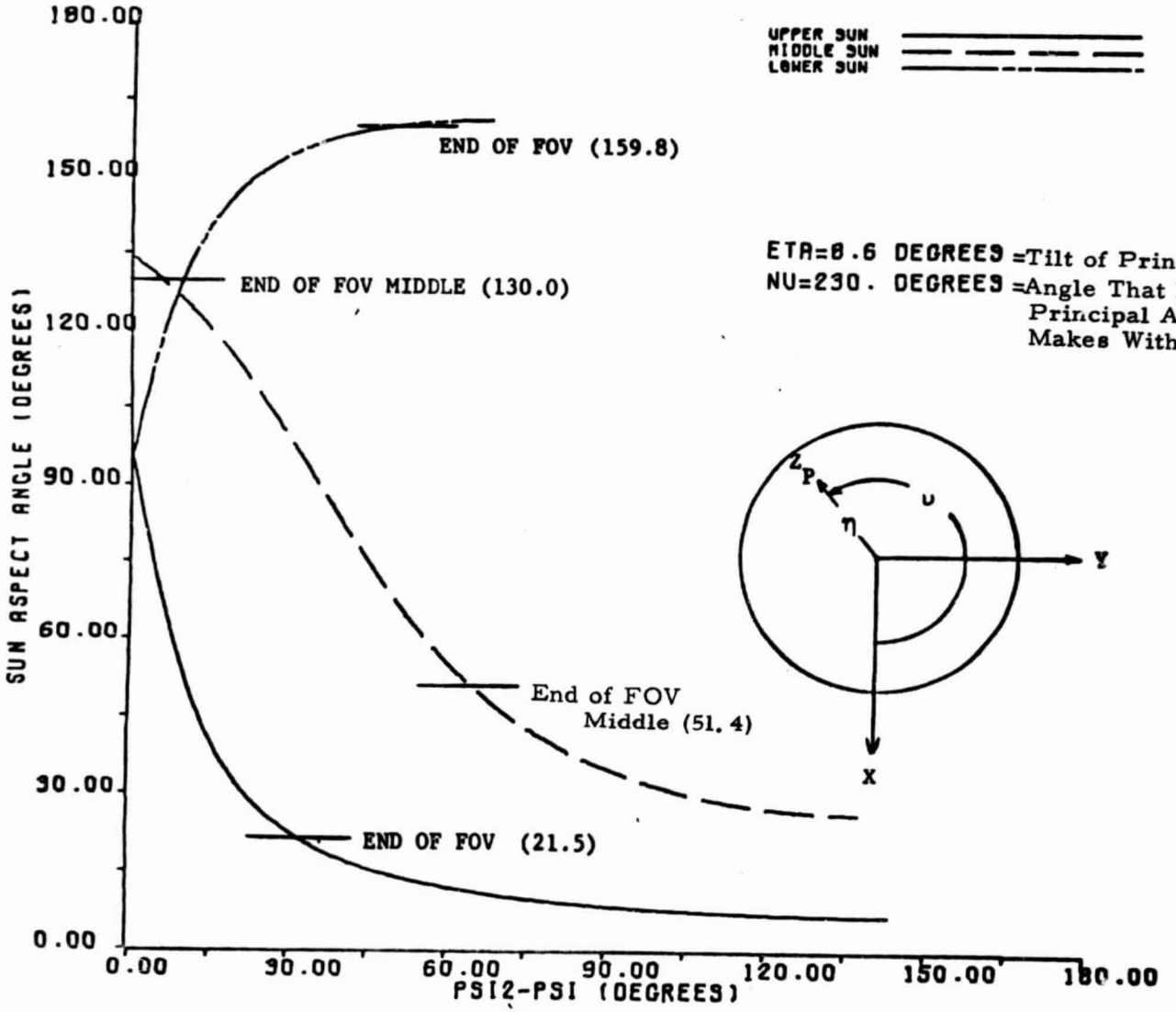


Figure 4.1.1.2-4. Sun Sensor Transfer Function Prior to Boom Deployment

C - 8

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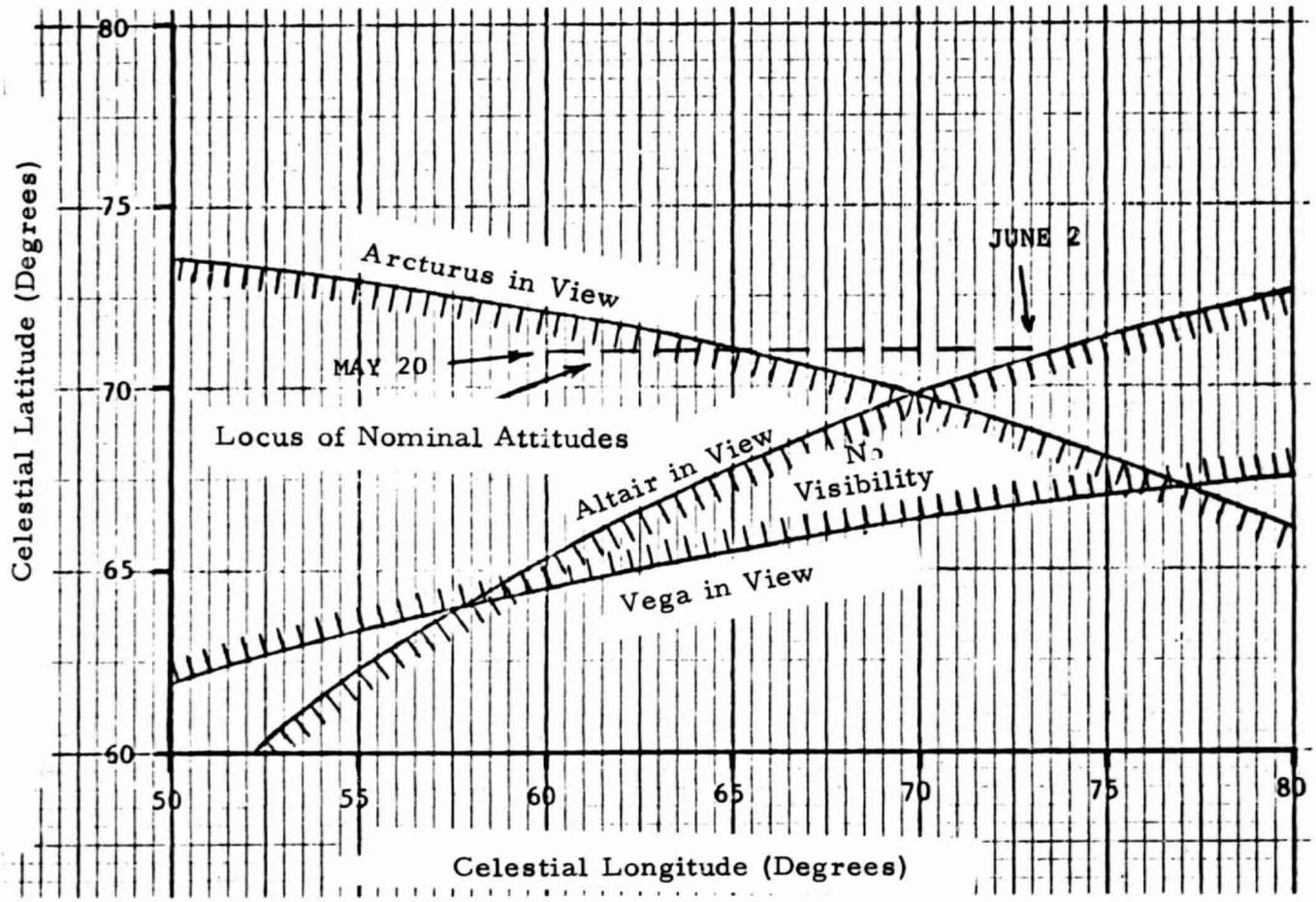


Figure 4.1.1.2-5. Star Visibility Prior to Magnetometer Boom Deployment

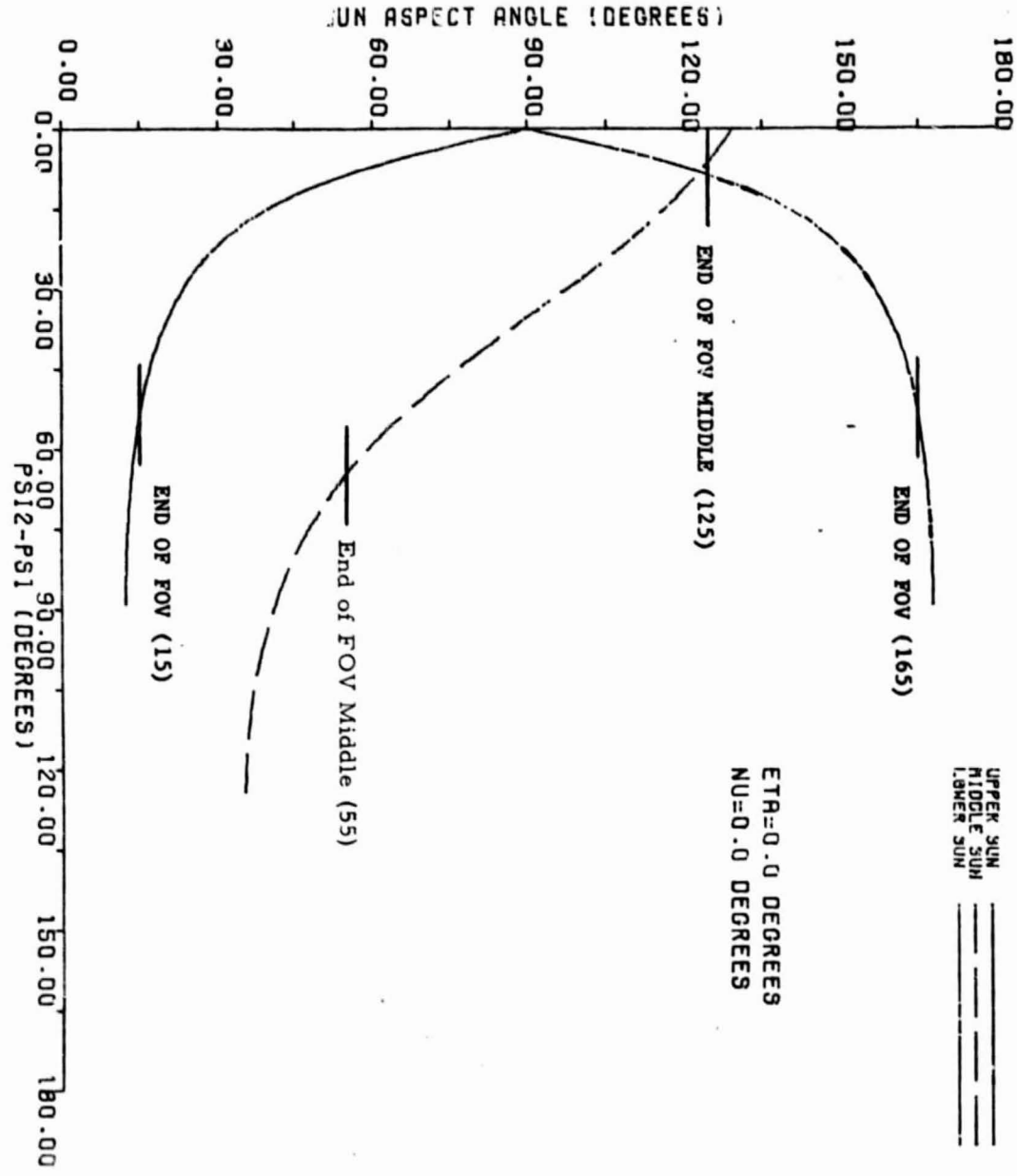


Figure 4.1.1.2-6. Sun Sensor Transfer Function After Boom Deployment

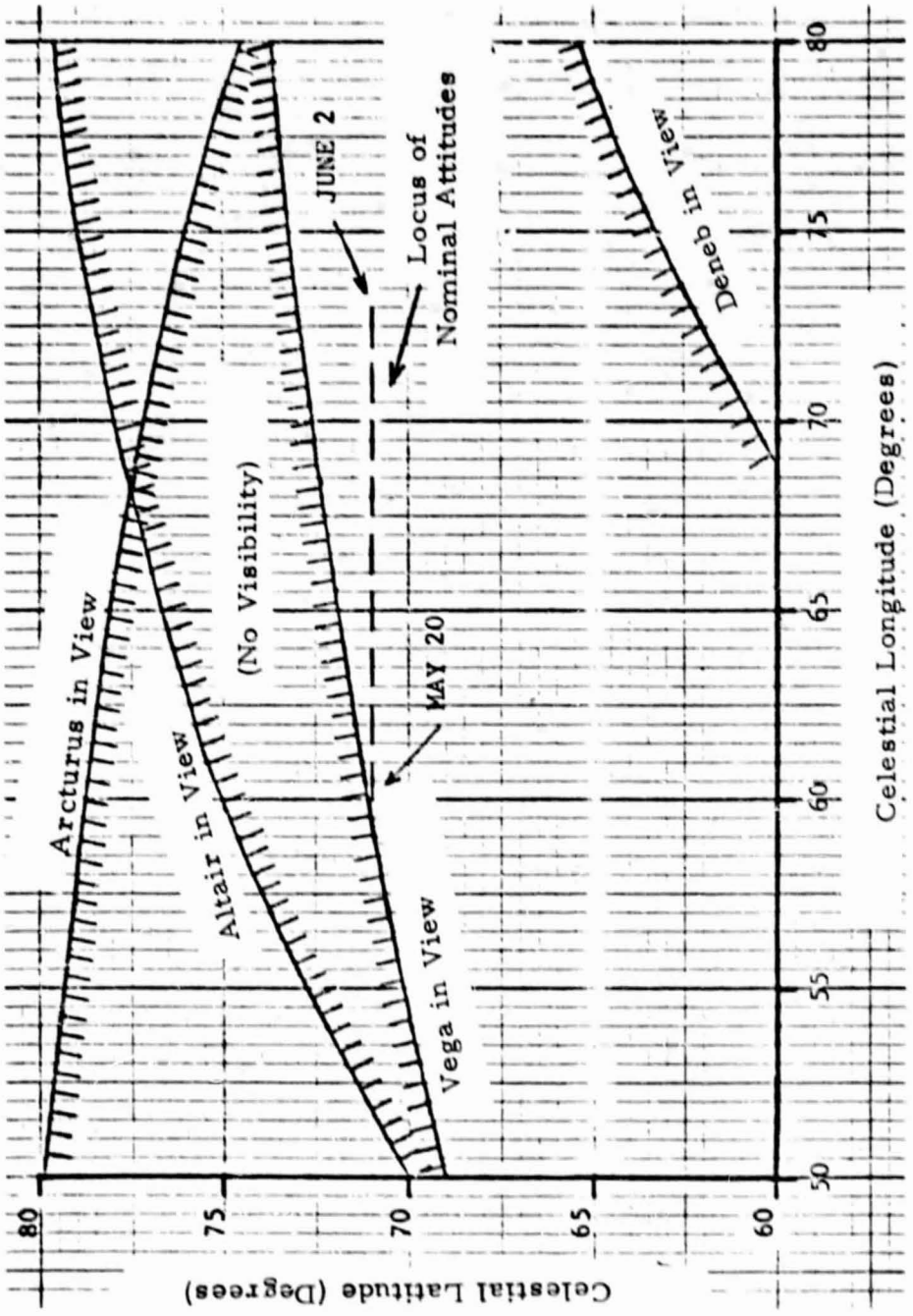


Figure 4.1.1.2-7. Star Visibility After Magnetometer Boom Deployment

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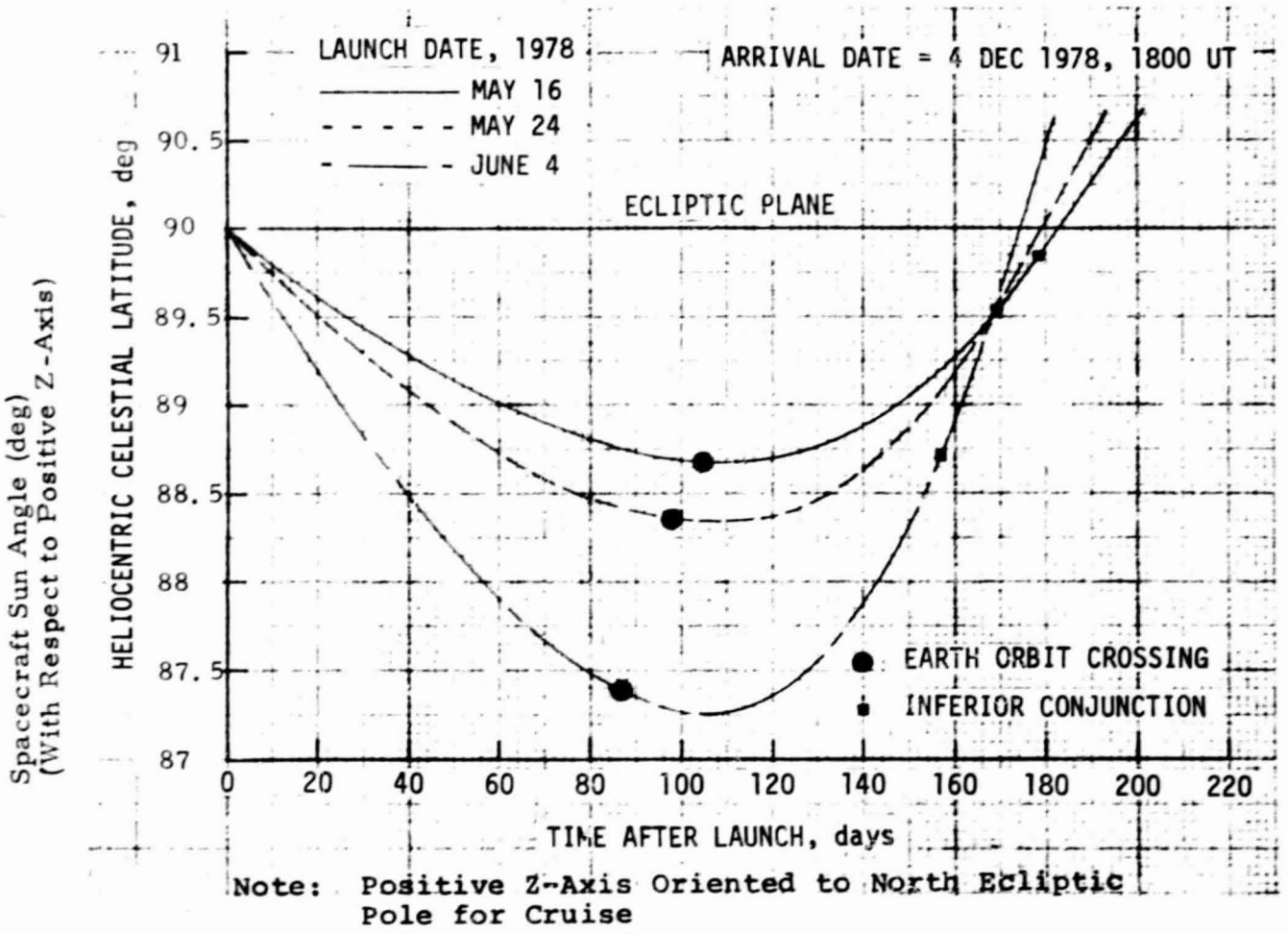
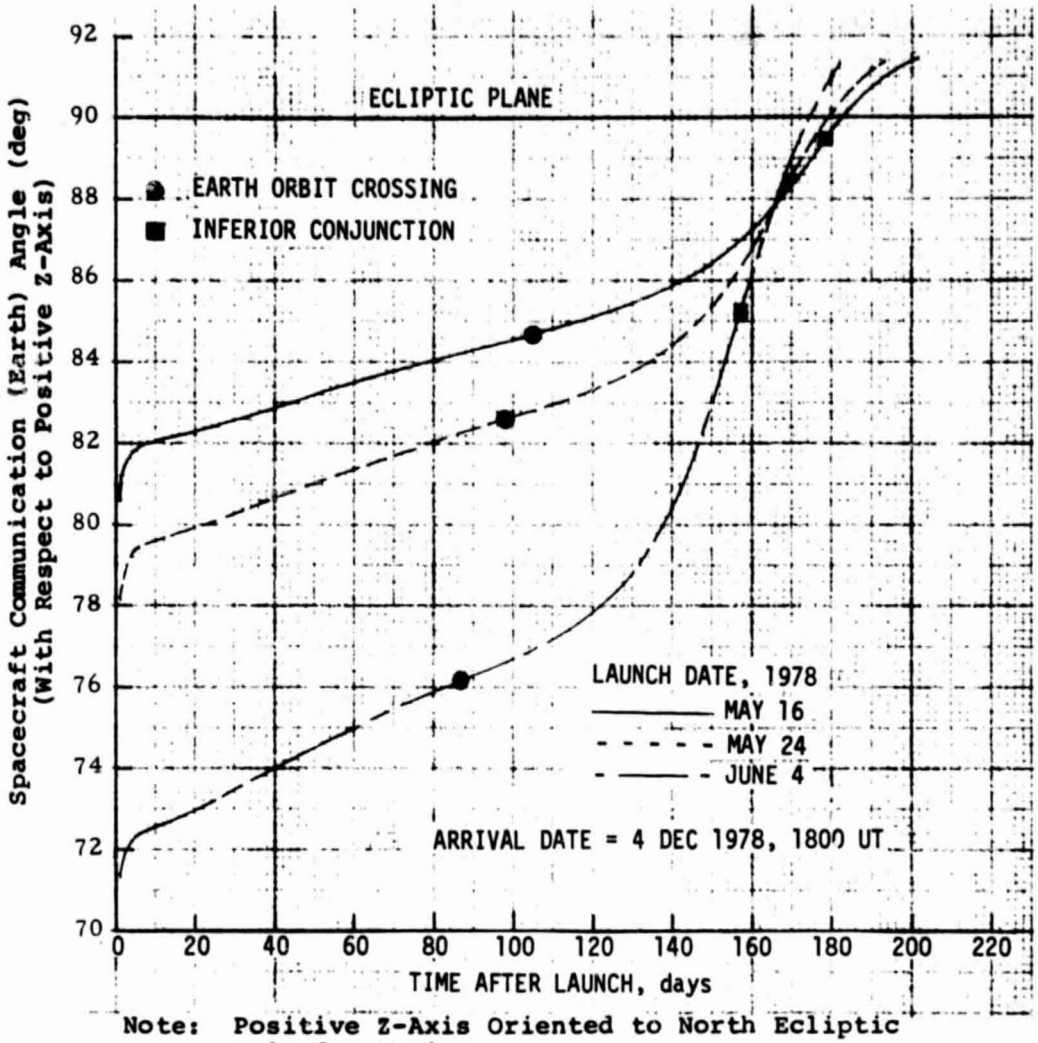


Figure 4.1.1.2-8. Orbiter Cruise Sun Angle History

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Note: Positive Z-Axis Oriented to North Ecliptic Pole for Cruise.

Figure 4.1.1.2-9. Orbiter Cruise Communication Angle History

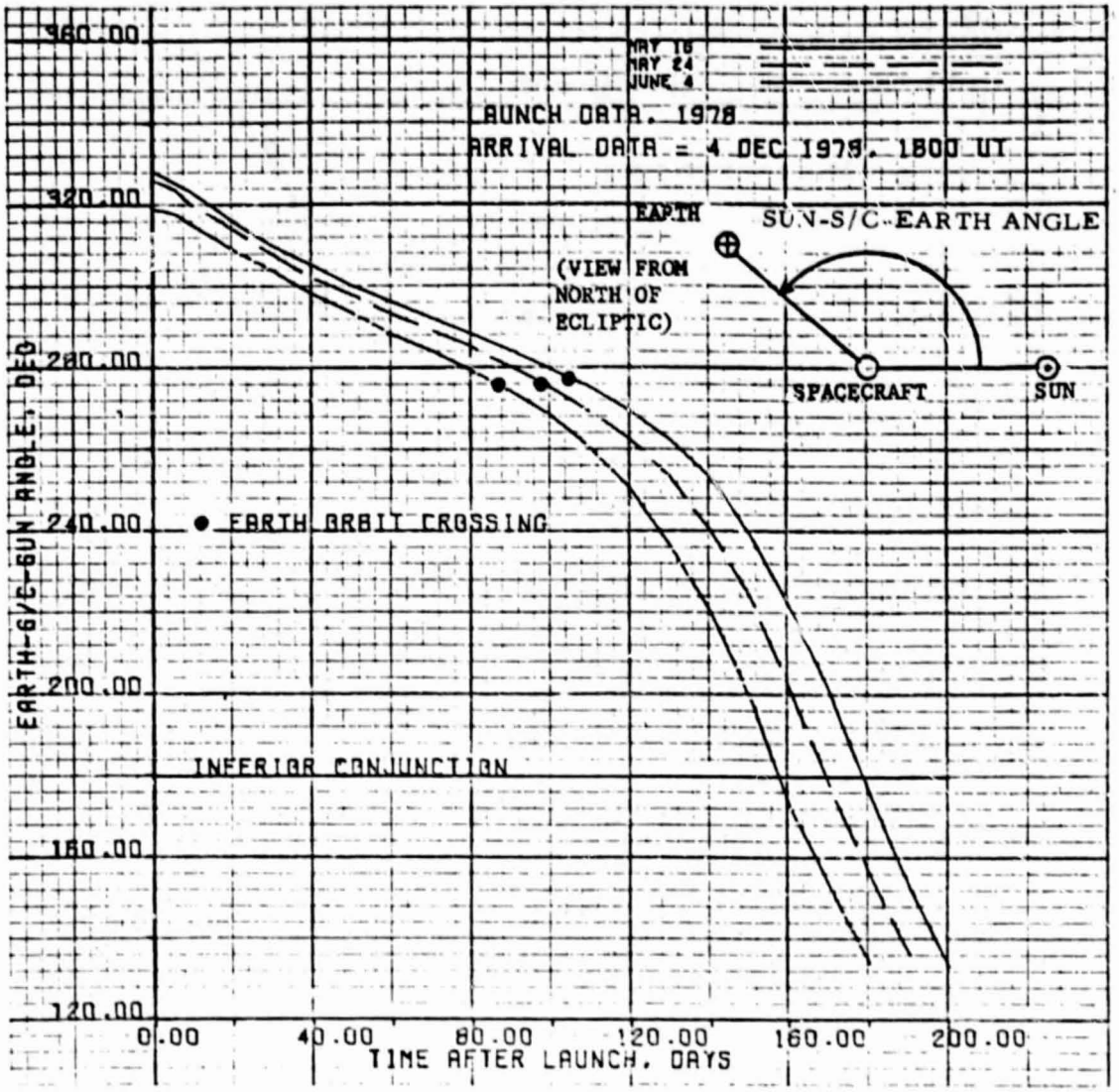
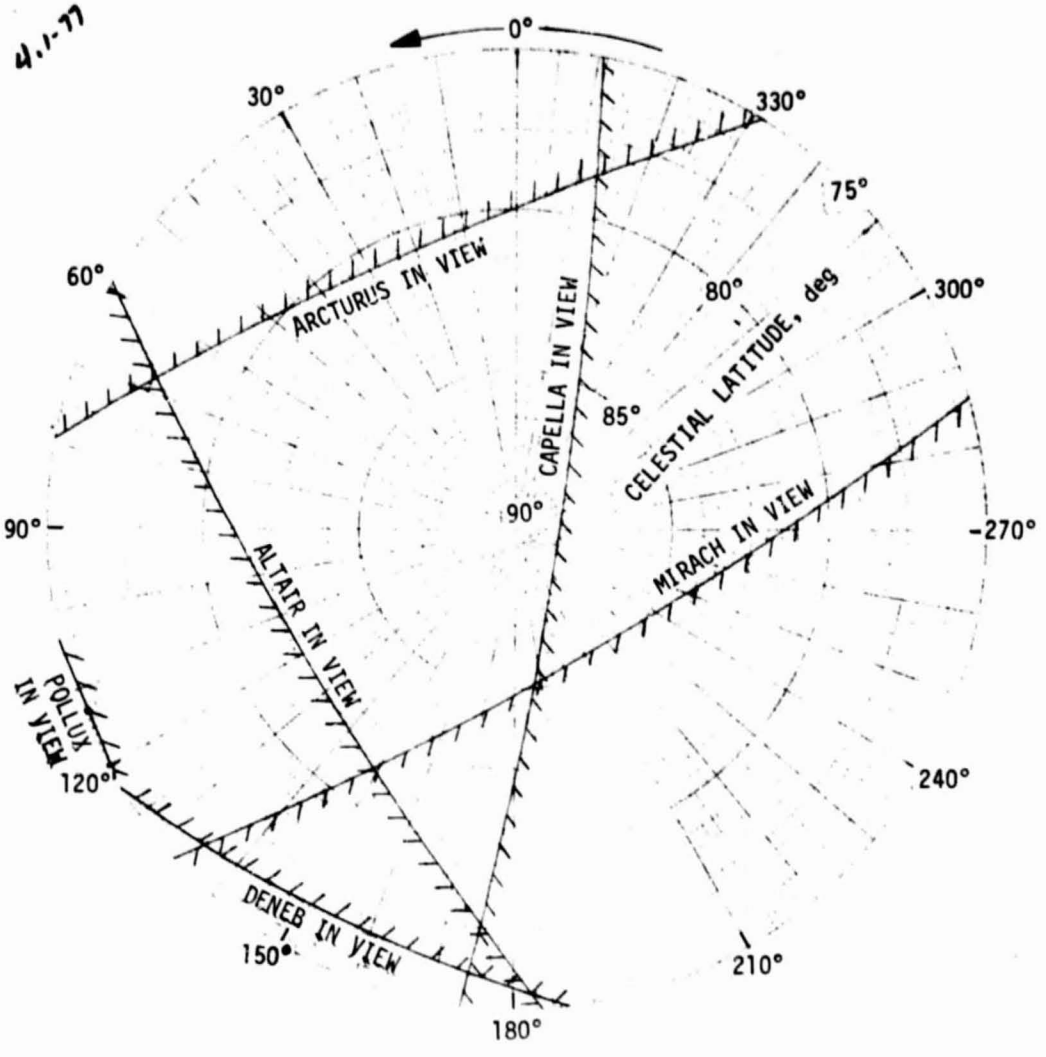


Figure 4.1.1.2-10. Sun-S/C-Earth Angle During Interplanetary Cruise for Pioneer Venus Orbiter

NOMINAL SPIN AXIS ATTITUDE: ECLIPTIC NORTH POLE

CELESTIAL LONGITUDE, deg



LOCATE SPACECRAFT ATTITUDE BY ITS CELESTIAL LATITUDE AND LONGITUDE TO DETERMINE VISIBLE STARS

ABOVE STARS SUBJECT TO SUN INTERFERENCE IN CRUISE AS FOLLOWS:

ARCTURUS - Day 85 to Venus	POLLUX - Launch to day 133
CAPELLA - Launch to day 99	DENEB - No interference
ALTAIR - Day 170 to Venus	MIRACH - Launch to day 36

Figure 4.1.1.2-11. Reference Star Visibility for Spacecraft Attitudes During Interplanetary Cruise for Pioneer Venus Orbiter

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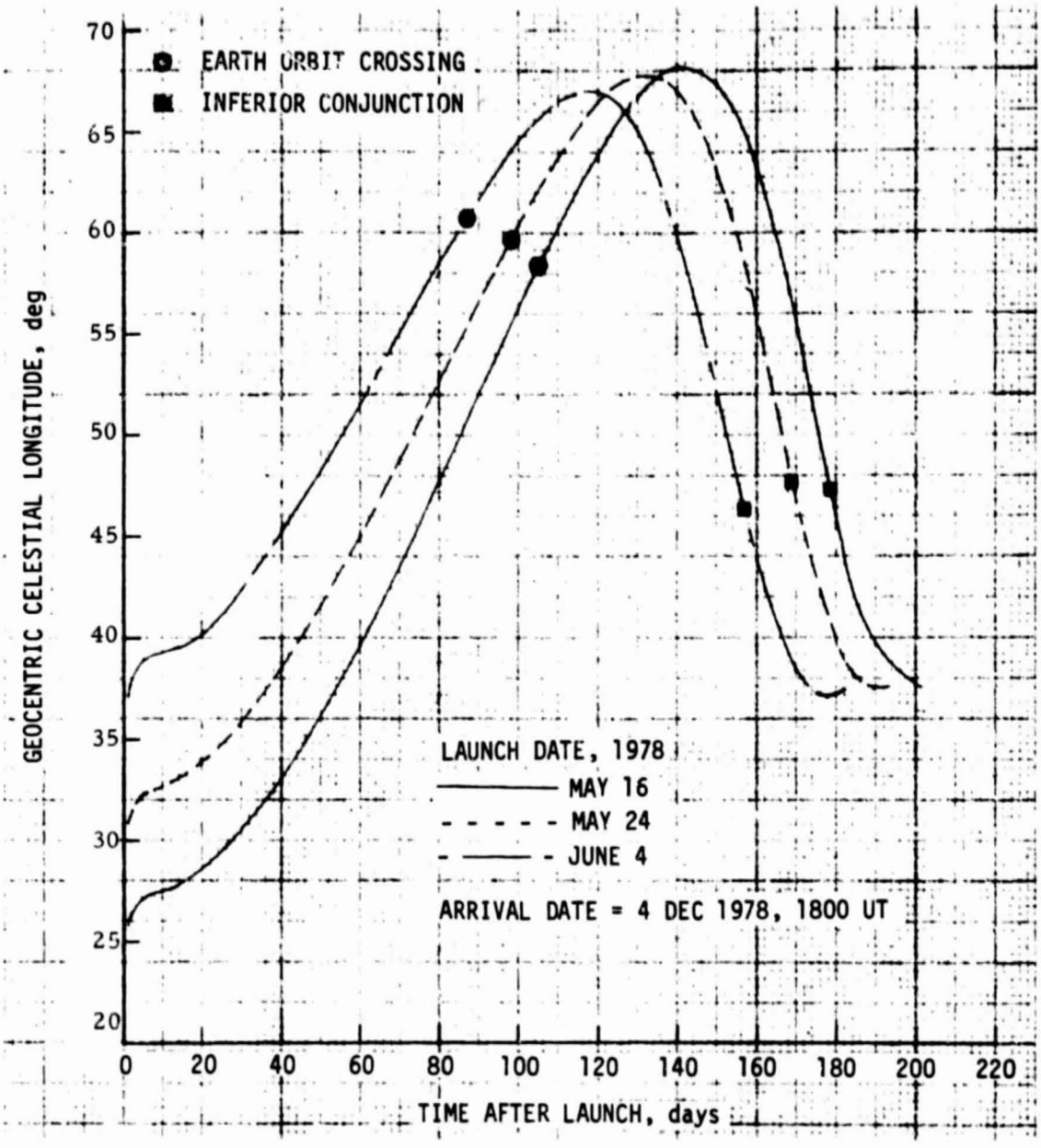
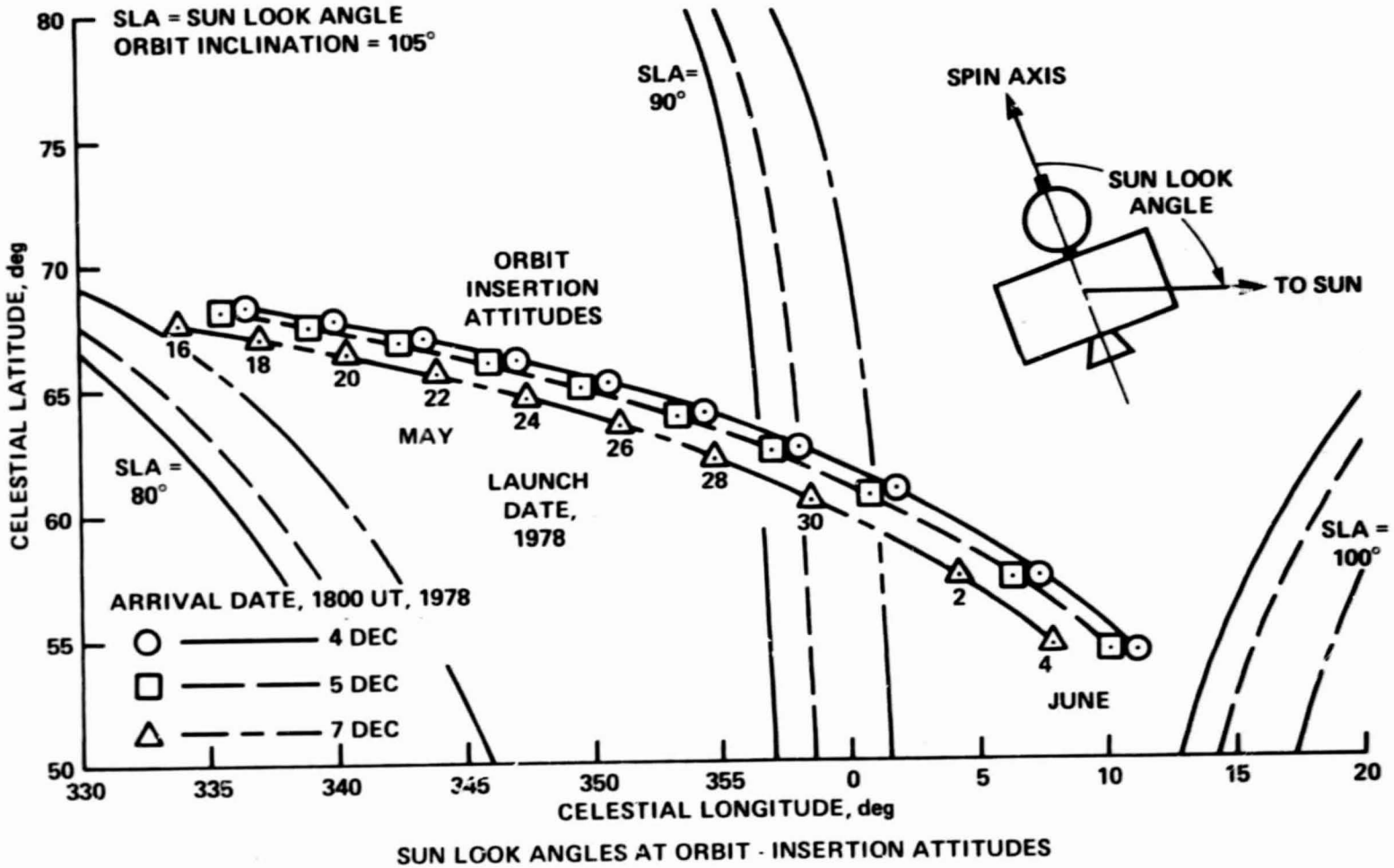
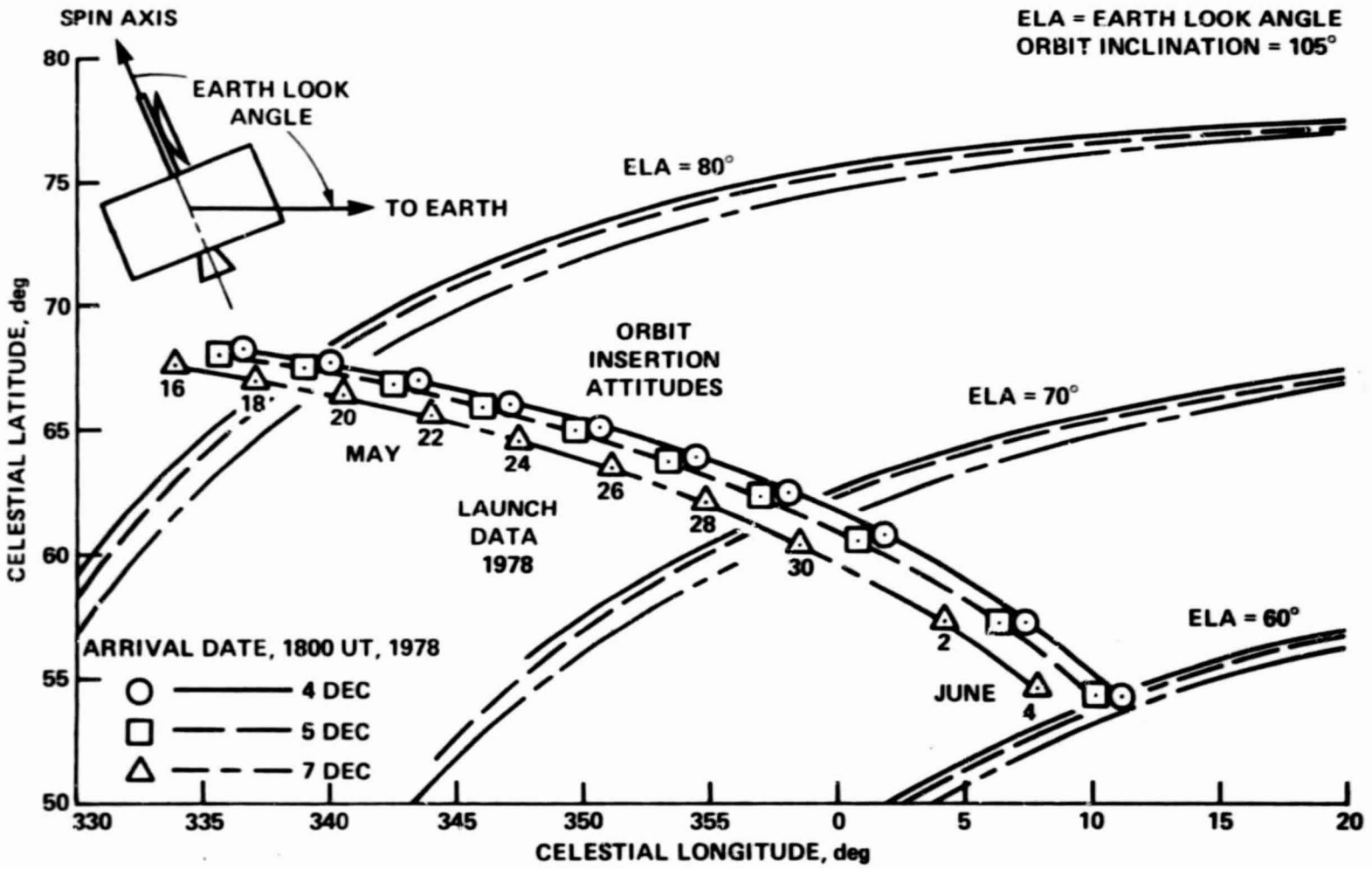


Figure 4.1.1.2-12. Geocentric Celestial Longitude During Interplanetary Cruise for Pioneer Venus Orbiter



SUN LOOK ANGLES AT ORBIT - INSERTION ATTITUDES

Figure 4.1.1.2-13. Sun Look Angles at Orbit-Insertion Attitudes for Pioneer Venus Orbiter



EARTH LOOK ANGLES AT ORBIT-INSERTION ATTITUDES

Figure 4.1.1.2-14. Earth Look Angles at Orbit-Insertion Attitudes for Pioneer Venus Orbiter

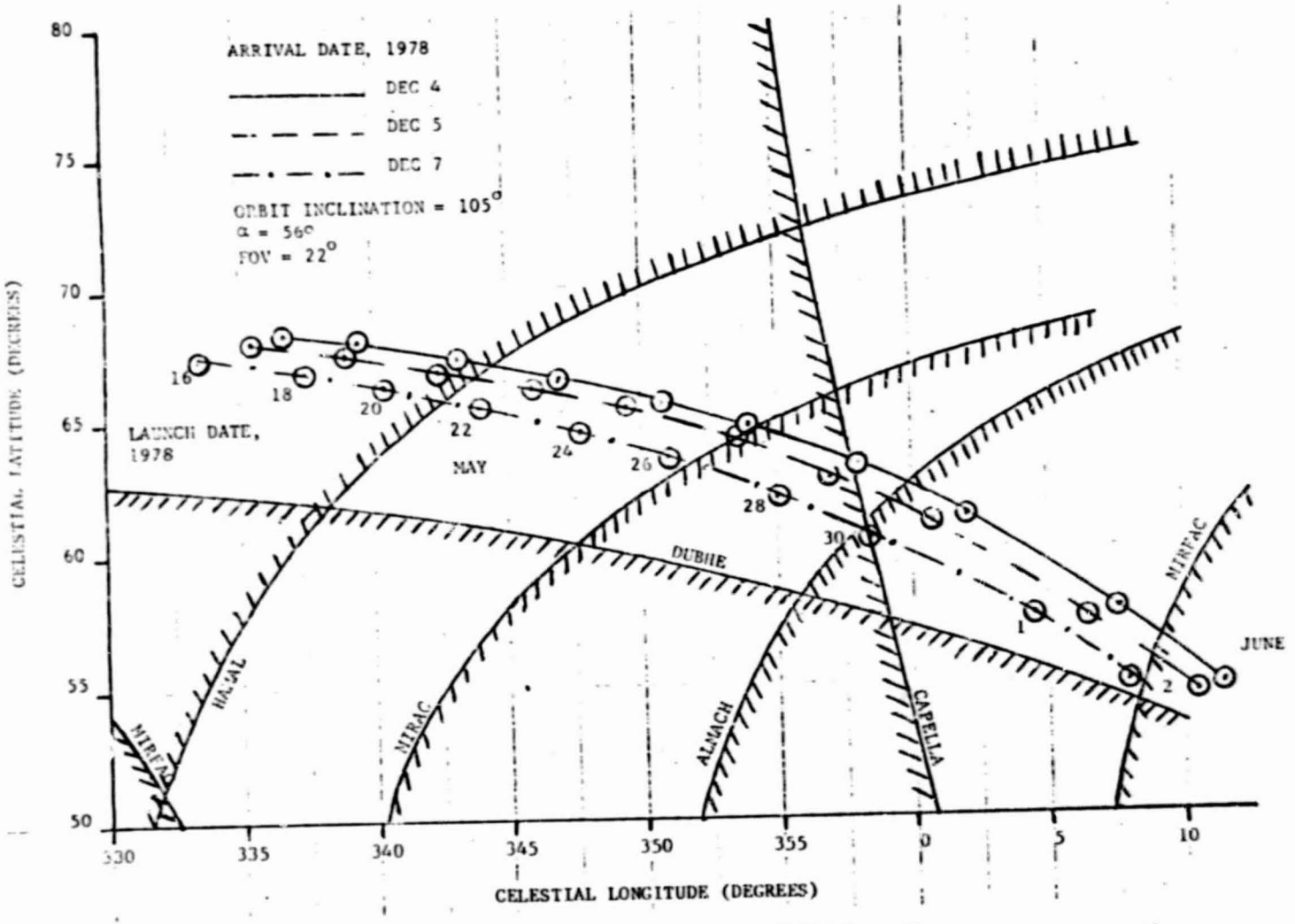
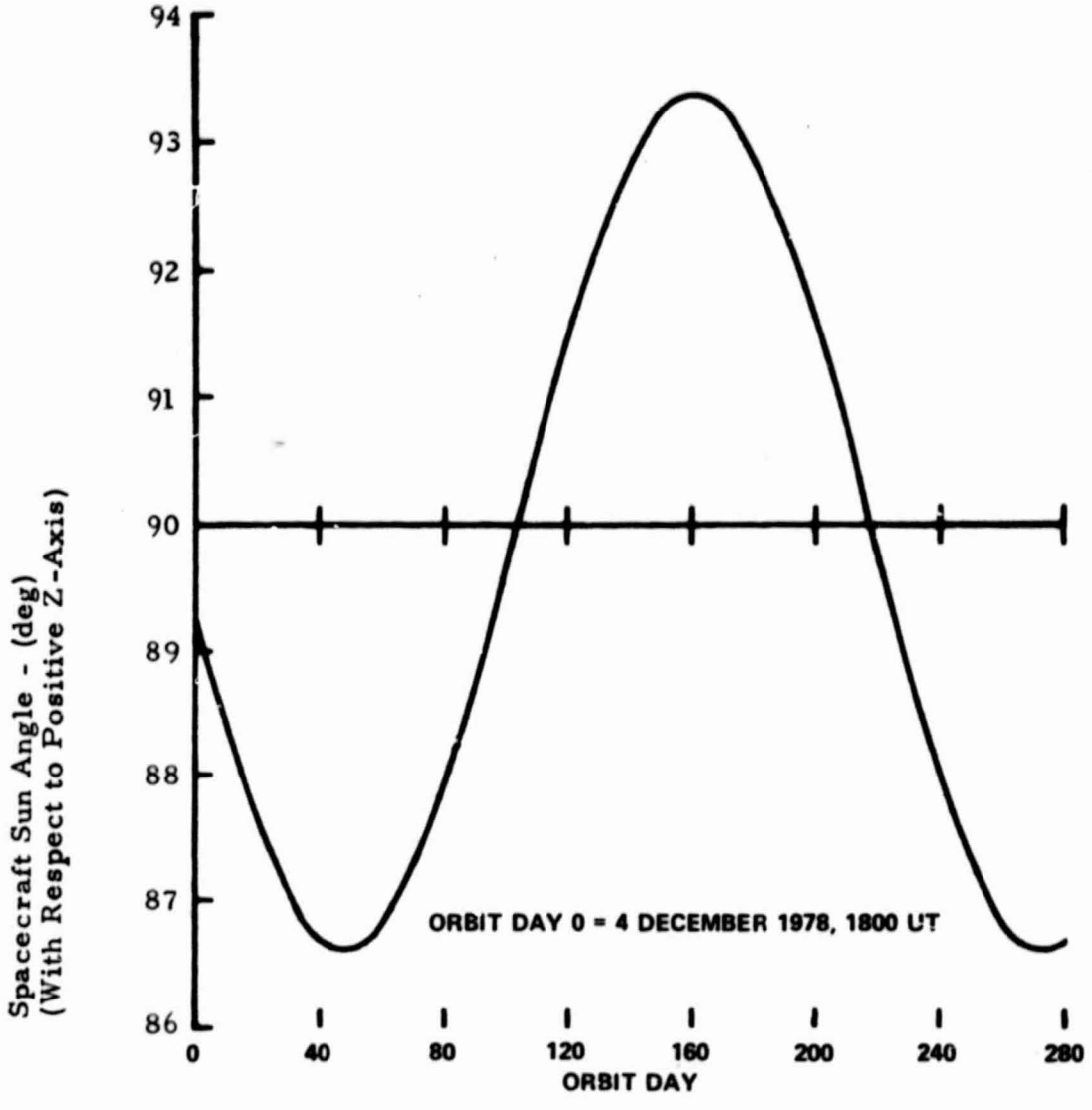
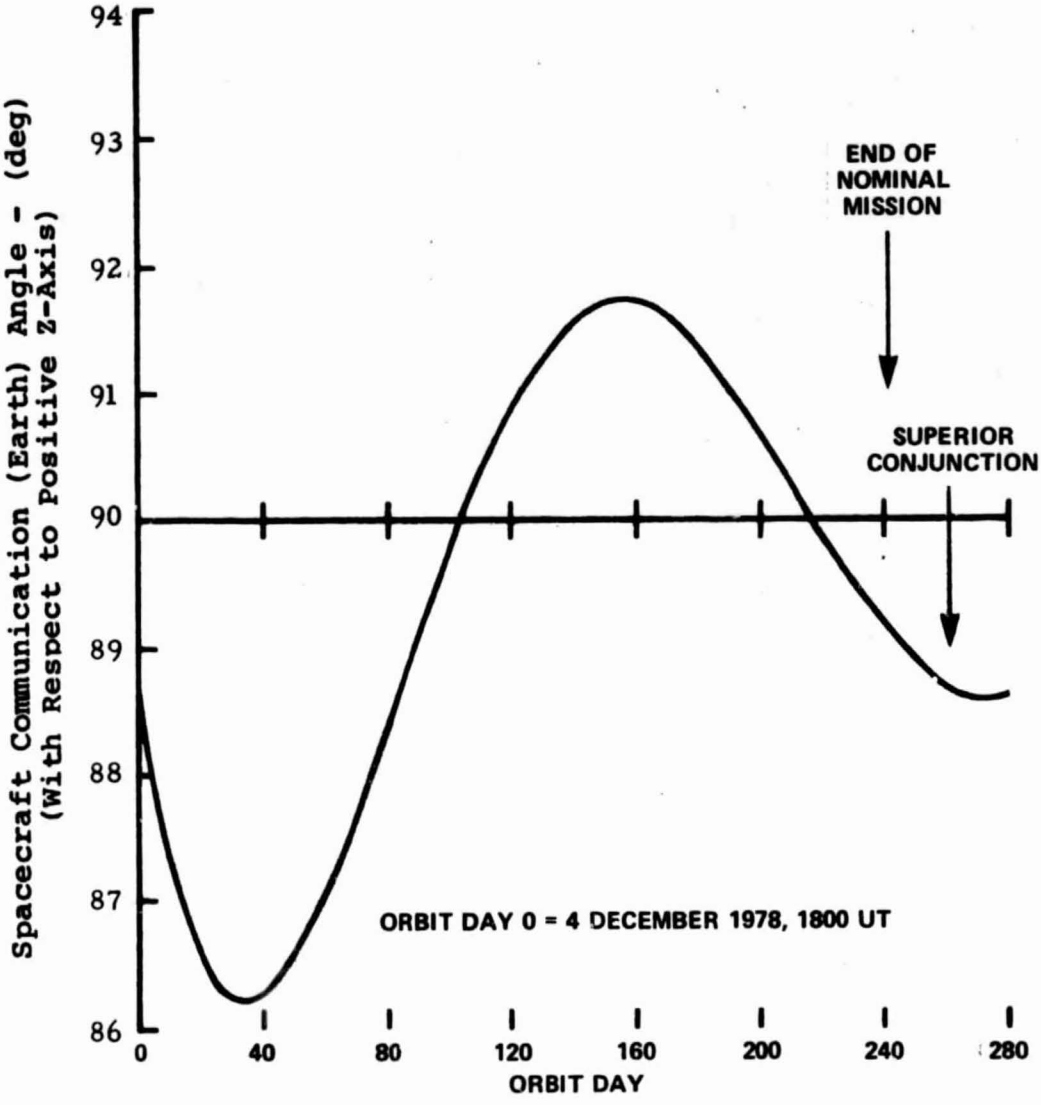


Figure 4.1.1.2-15. Reference-Star Visibility at Orbit-Insertion Attitudes for Pioneer Venus Orbiter



Note: Positive Z-Axis Oriented to South Ecliptic Pole On-Orbit.

Figure 4.1.1.2-16. On-Orbit Sun Angle History



Note: Positive Z-Axis Oriented to South
Ecliptic Pole On-Orbit

Figure 4.1.1.2-17. On-Orbit Communication Angle History

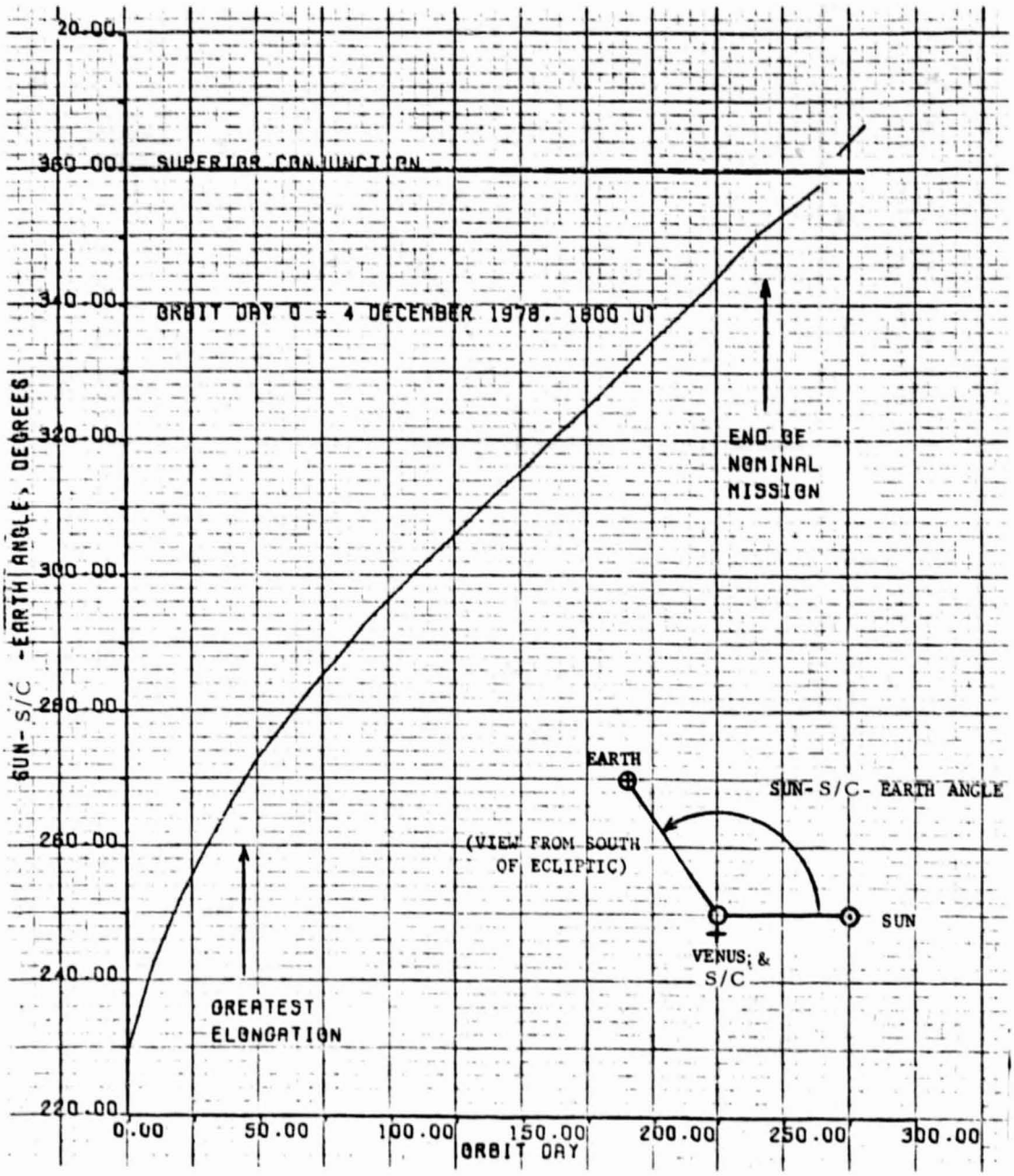
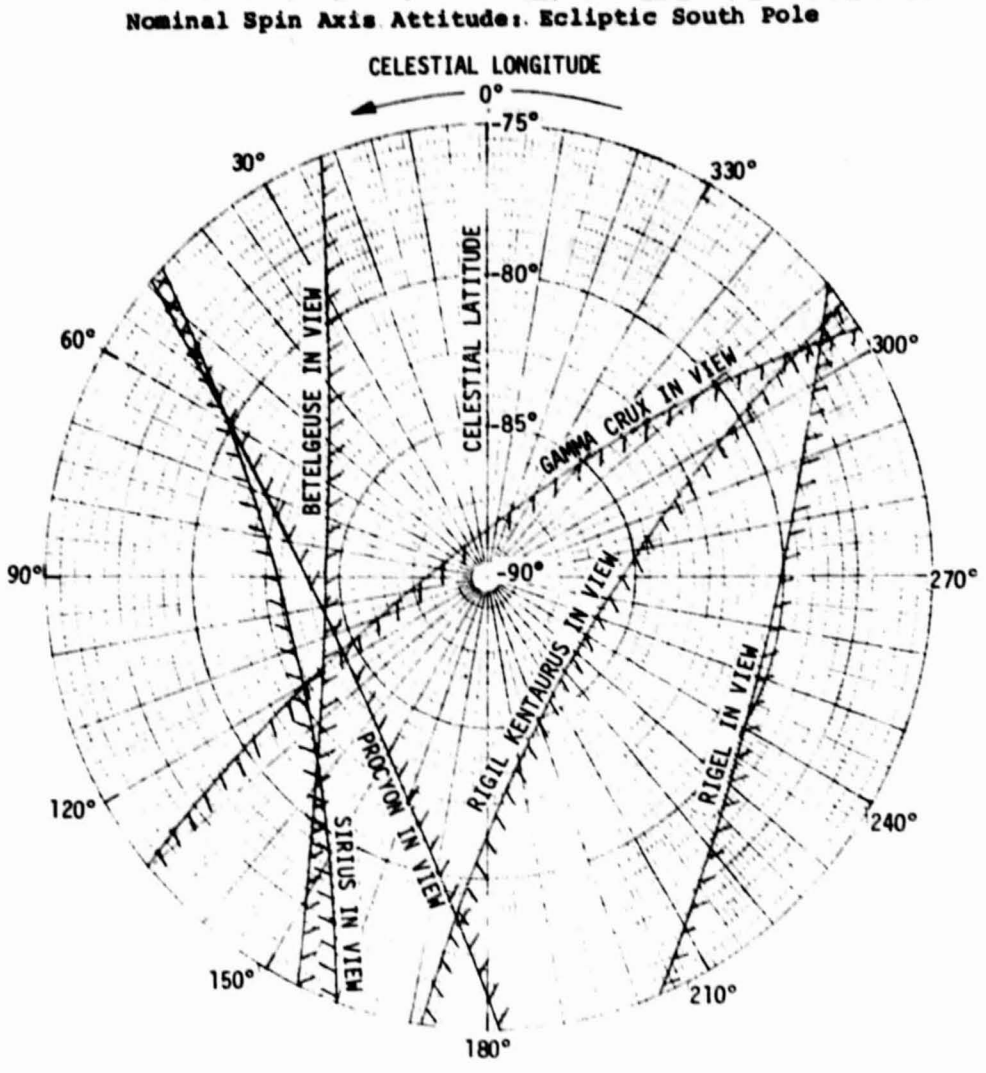


Figure 4.1.1.2-18. Sun-S/C-Earth Angle During Pioneer Venus In-Orbit Mission

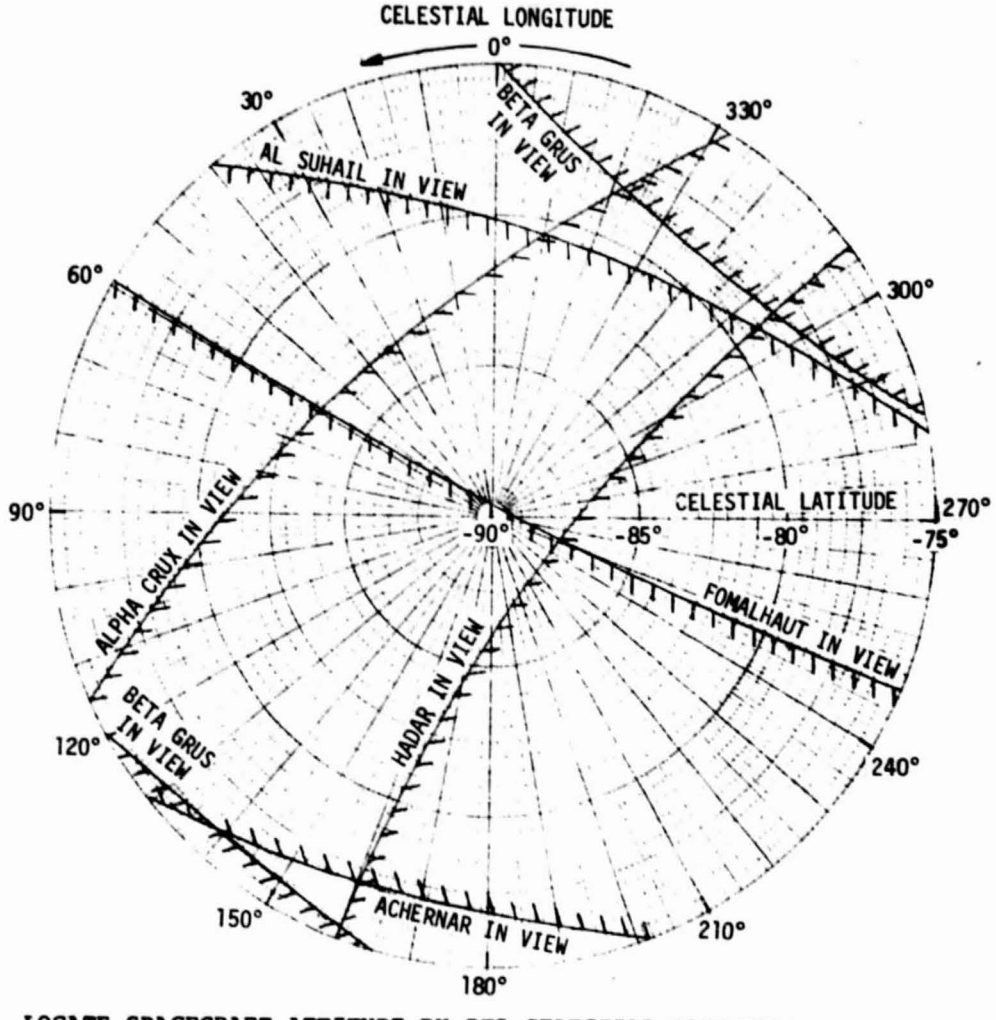


LOCATE SPACECRAFT ATTITUDE BY ITS CELESTIAL LATITUDE AND LONGITUDE TO DETERMINE STARS IN SENSOR FIELD OF VIEW

VISIBILITY SUBJECT TO INTERFERENCE FROM VENUS AND THE SUN

Figure 4.1.1.2-19A. Reference Stars in Field of View for Spacecraft Attitudes in Orbit for Pioneer Venus Orbiter

Nominal Spin Axis Attitude: Ecliptic South Pole



LOCATE SPACECRAFT ATTITUDE BY ITS CELESTIAL LATITUDE AND LONGITUDE TO DETERMINE STARS IN SENSOR FIELD OF VIEW

VISIBILITY SUBJECT TO INTERFERENCE FROM VENUS AND THE SUN

Figure 4.1.1.2-19B. Reference Stars in Field of View for Spacecraft Attitudes in Orbit for Pioneer Venus Orbiter

Figure 4.1.1.3-1. Orbiter Spin Rate History

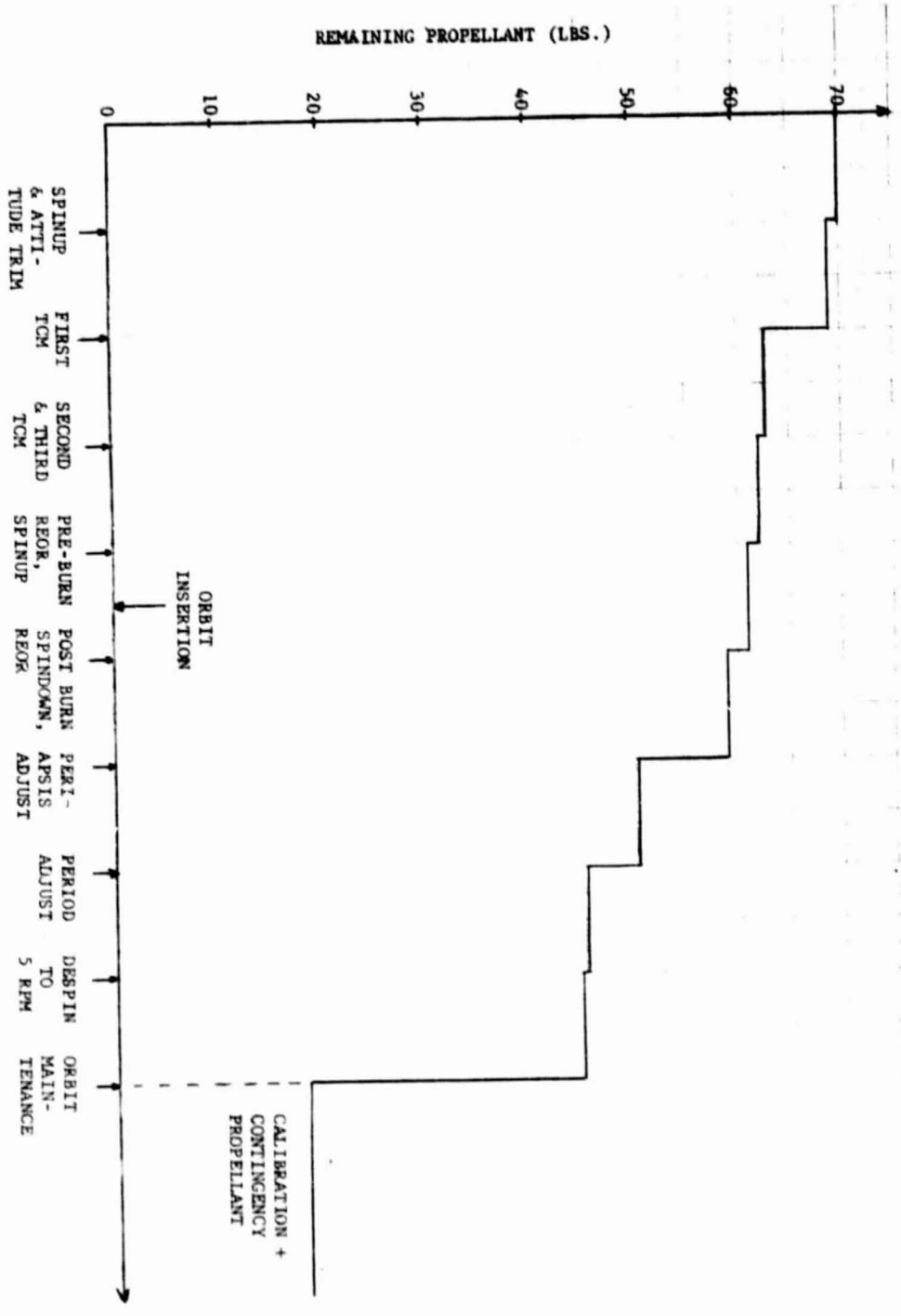


Figure 4.1.1.4-1. Orbiter Propellant Utilization Profile

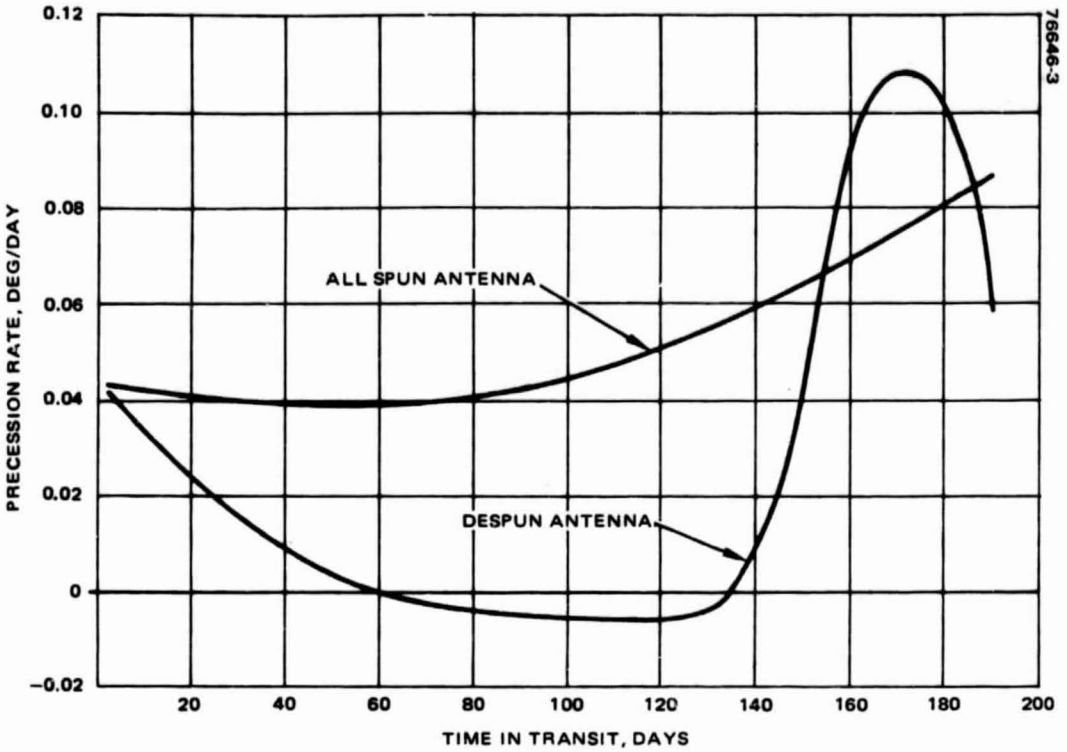


FIGURE 4.1.2.1-1. ORBITER SOLAR TORQUE PRECESSION RATE DURING CRUISE

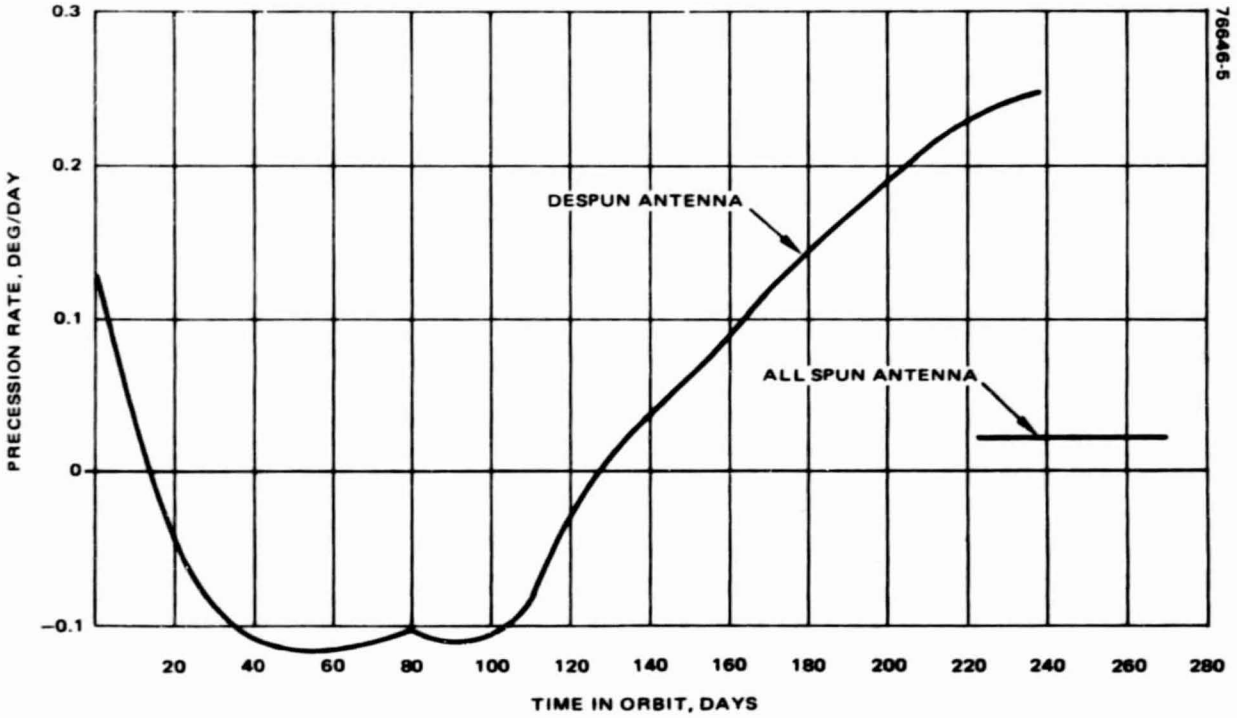


FIGURE 4.1.2.1-2. ORBITER SOLAR TORQUE PRECESSION RATE DURING ON-ORBIT PHASE

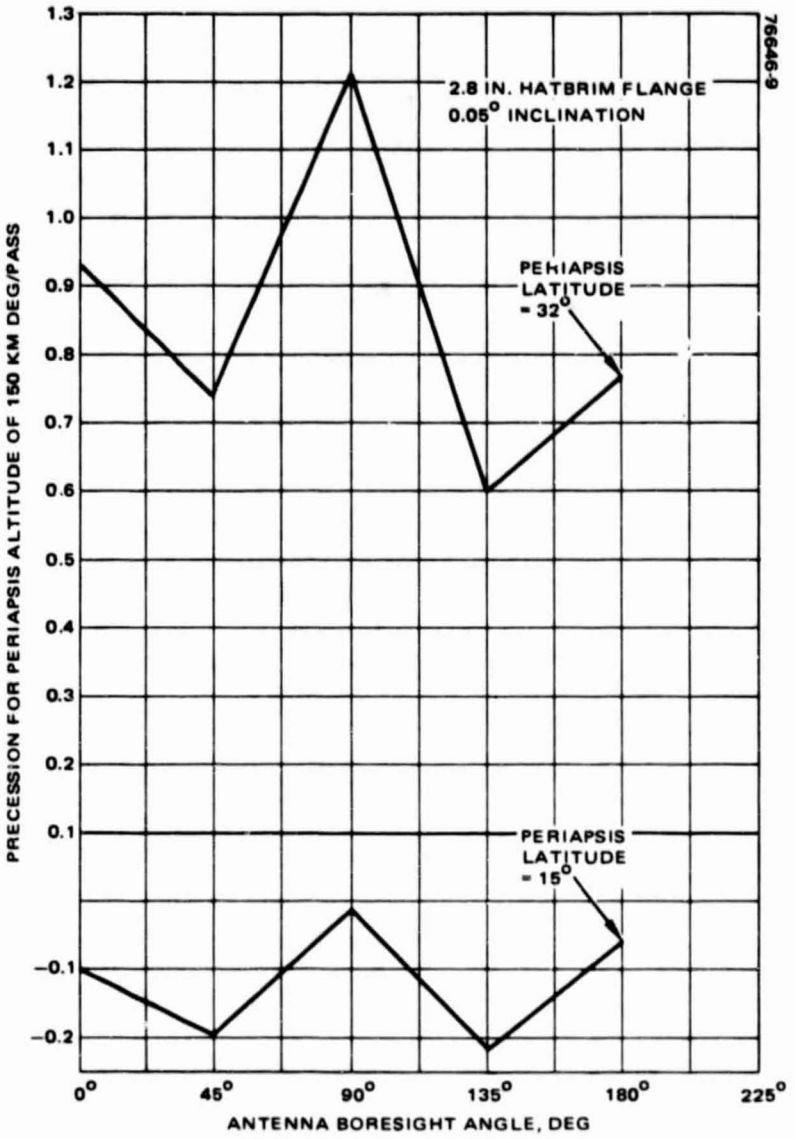
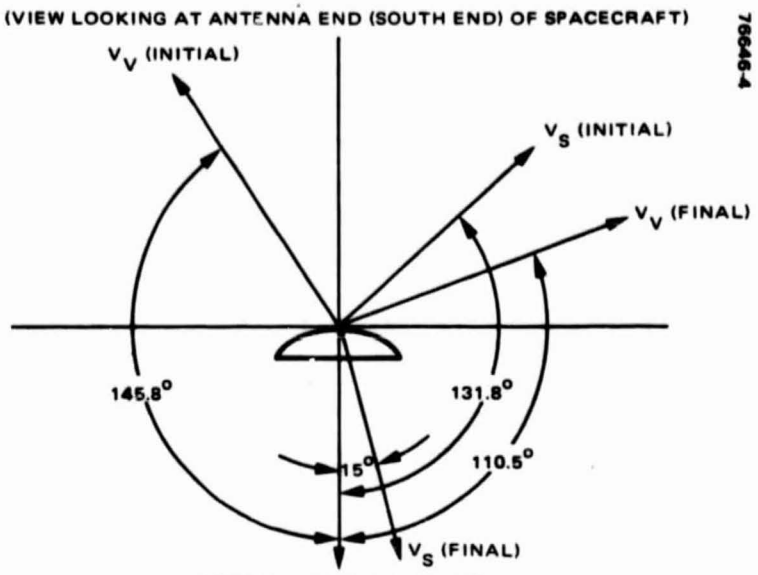
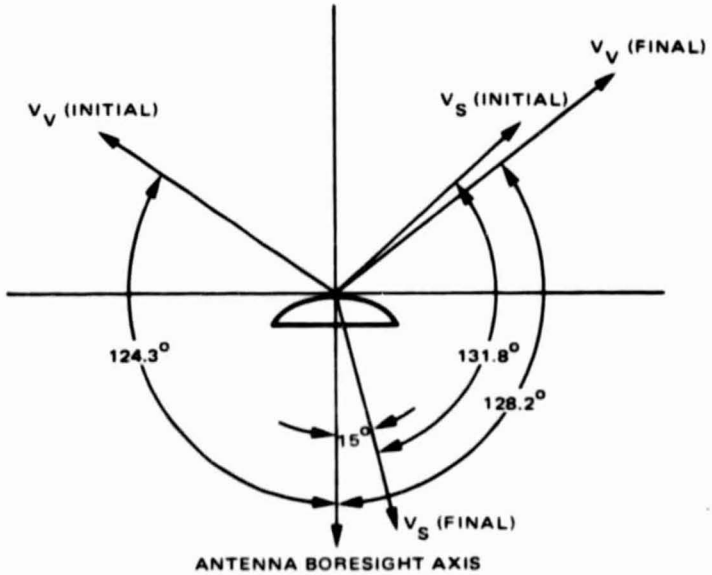


FIGURE 4.1.2.2-1. ORBITER AERODYNAMIC PRECESSION PER PERIAPSIS PASSAGE VS ANGLE BETWEEN ANTENNA BORESIGHT AND ORBIT PLANE



a) PERIAPSIS LATITUDE = 32°
(JUNE 4 LAUNCH)



b) PERIAPSIS LATITUDE = 15°
(MAY 16 LAUNCH)

FIGURE 4.1.2.2-2. RELATIVE ANGLES BETWEEN PROJECTION OF VELOCITY (V_V) AND SUN (V_S) VECTORS AND ANTENNA BORESIGHT AXIS IN TRANSVERSE PLANE DURING 243 DAYS ON-ORBIT PHASE

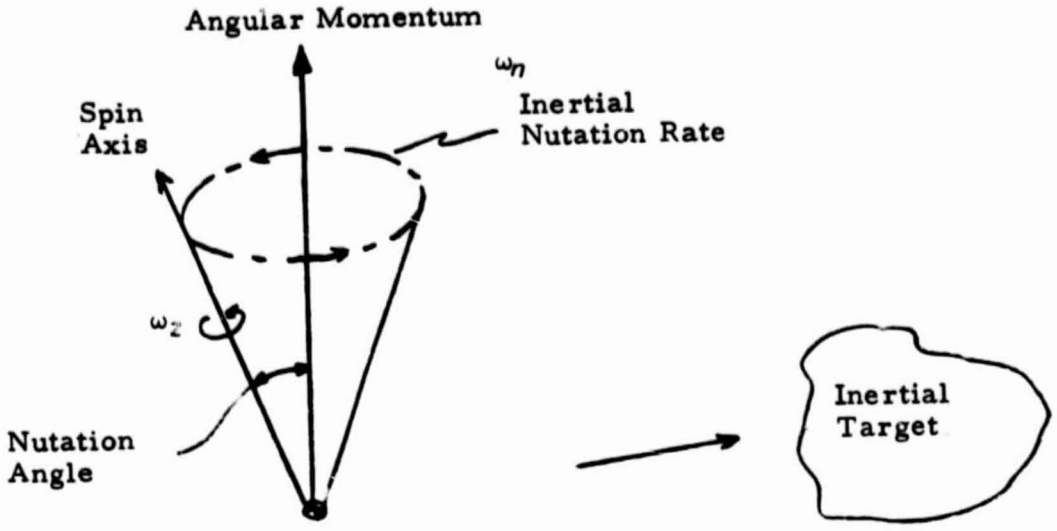


Figure 4.1.2.3-1. Nutation for Symmetric Body

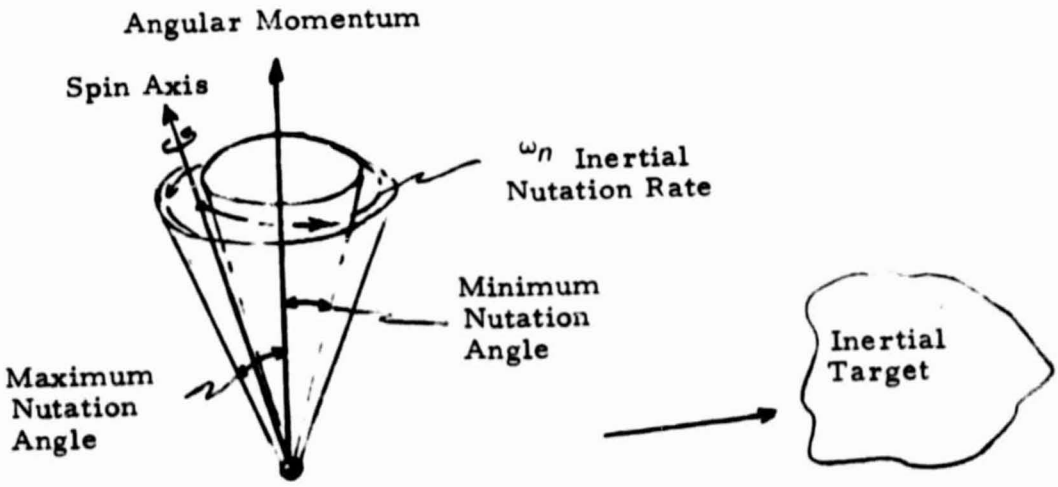


Figure 4.1.2.3-2. Nutation for Asymmetric Body

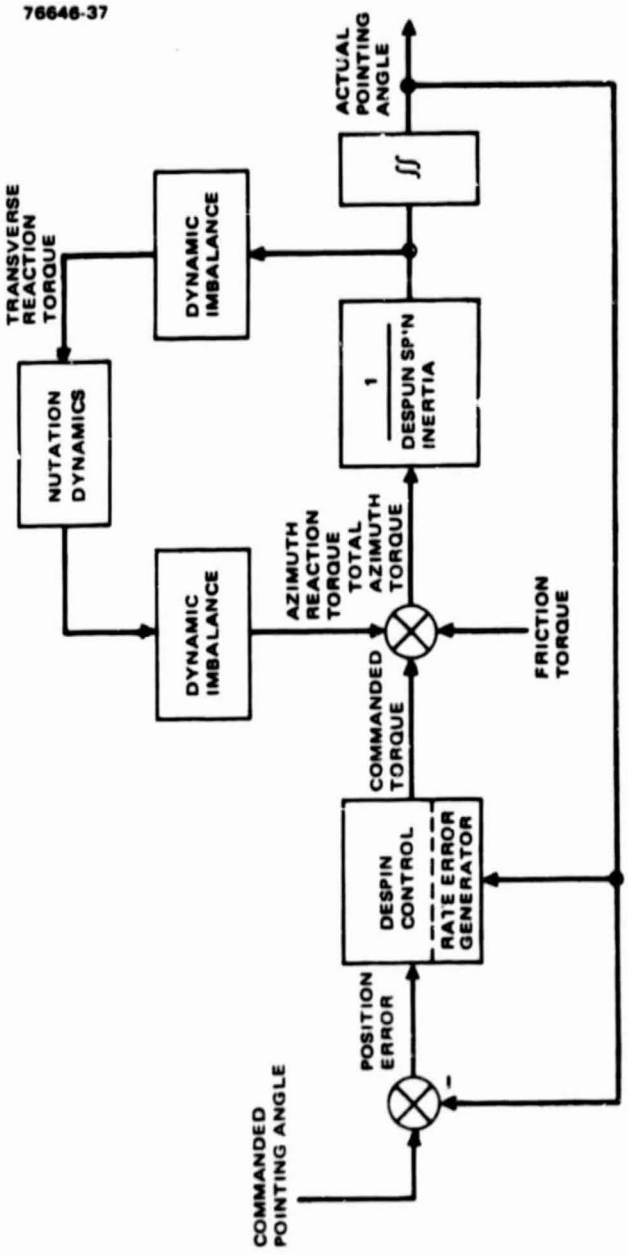


FIGURE 4.1.2.3-3. DESPIN CONTROL/NUTATION COUPLING

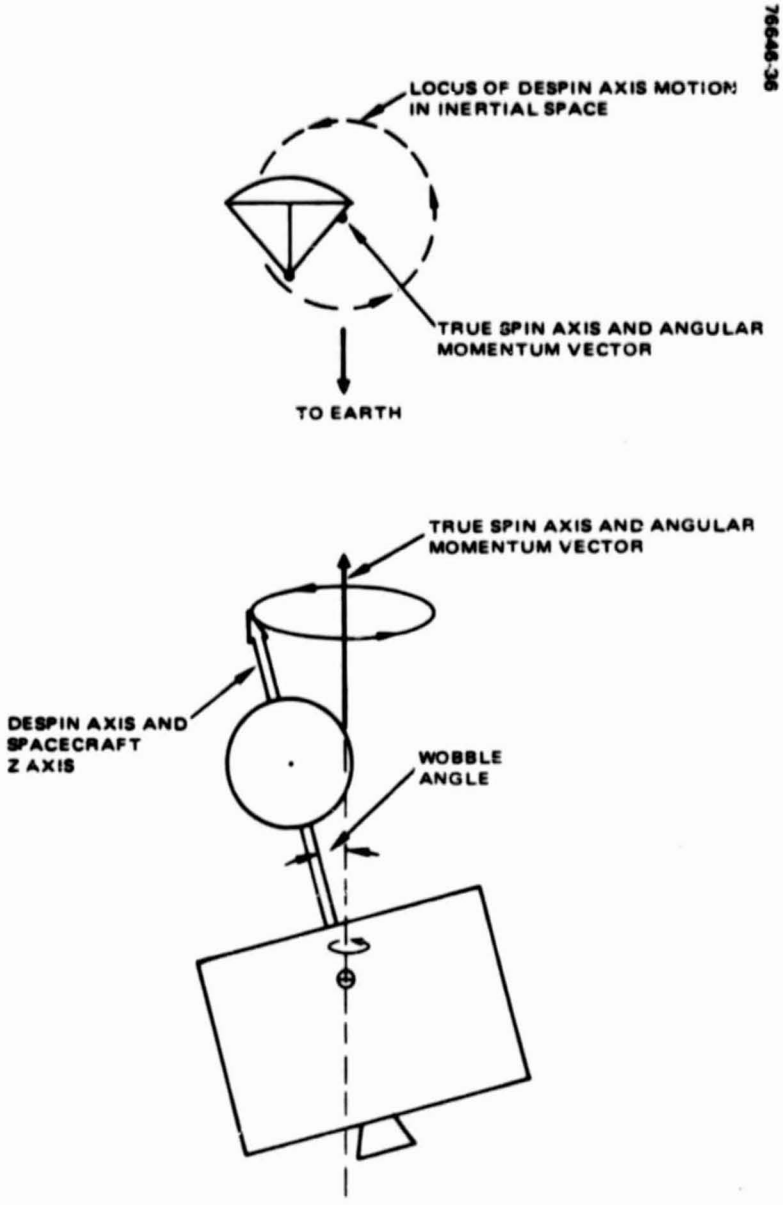


FIGURE 4.1.2.4-1. WOBBLE OF HGA BEAM

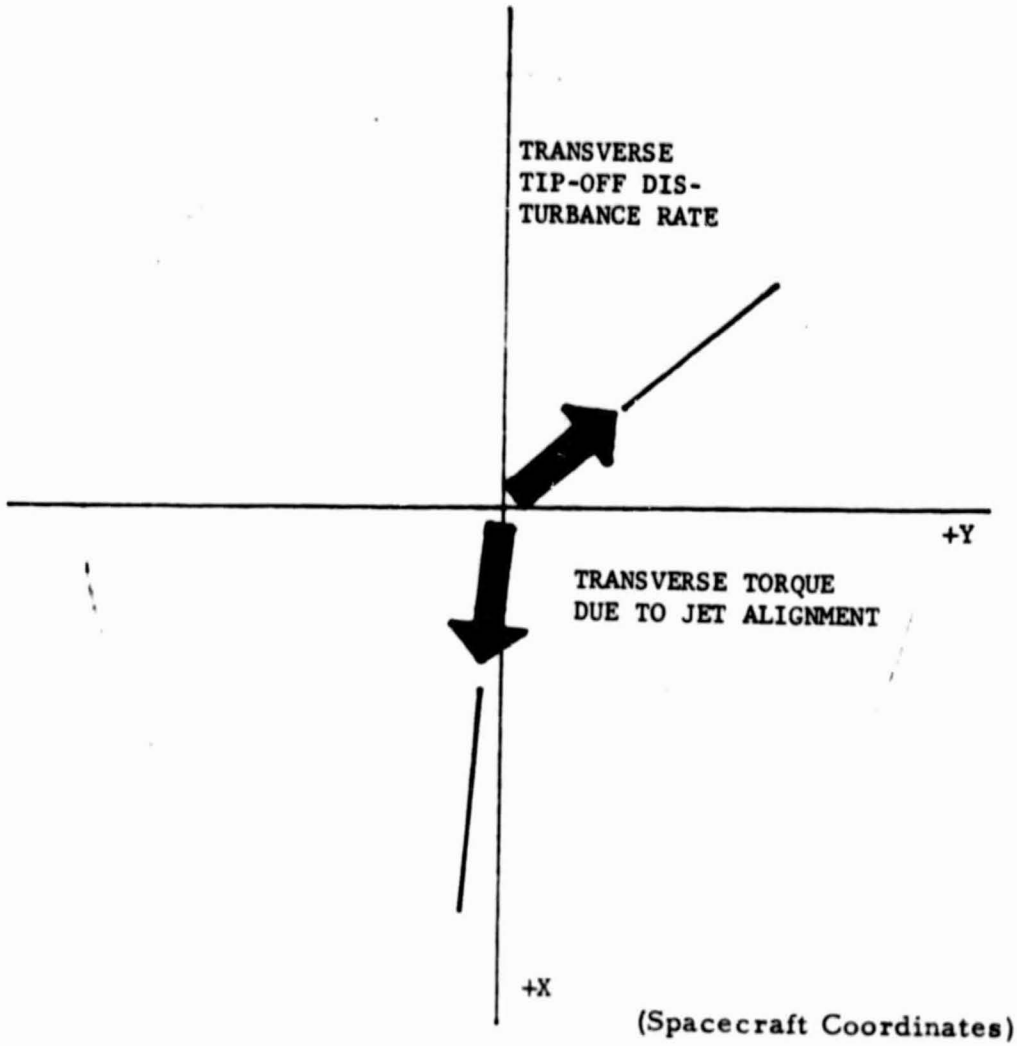


Figure 4.1.3.1-1. Orbiter Disturbance Effects Following Separation

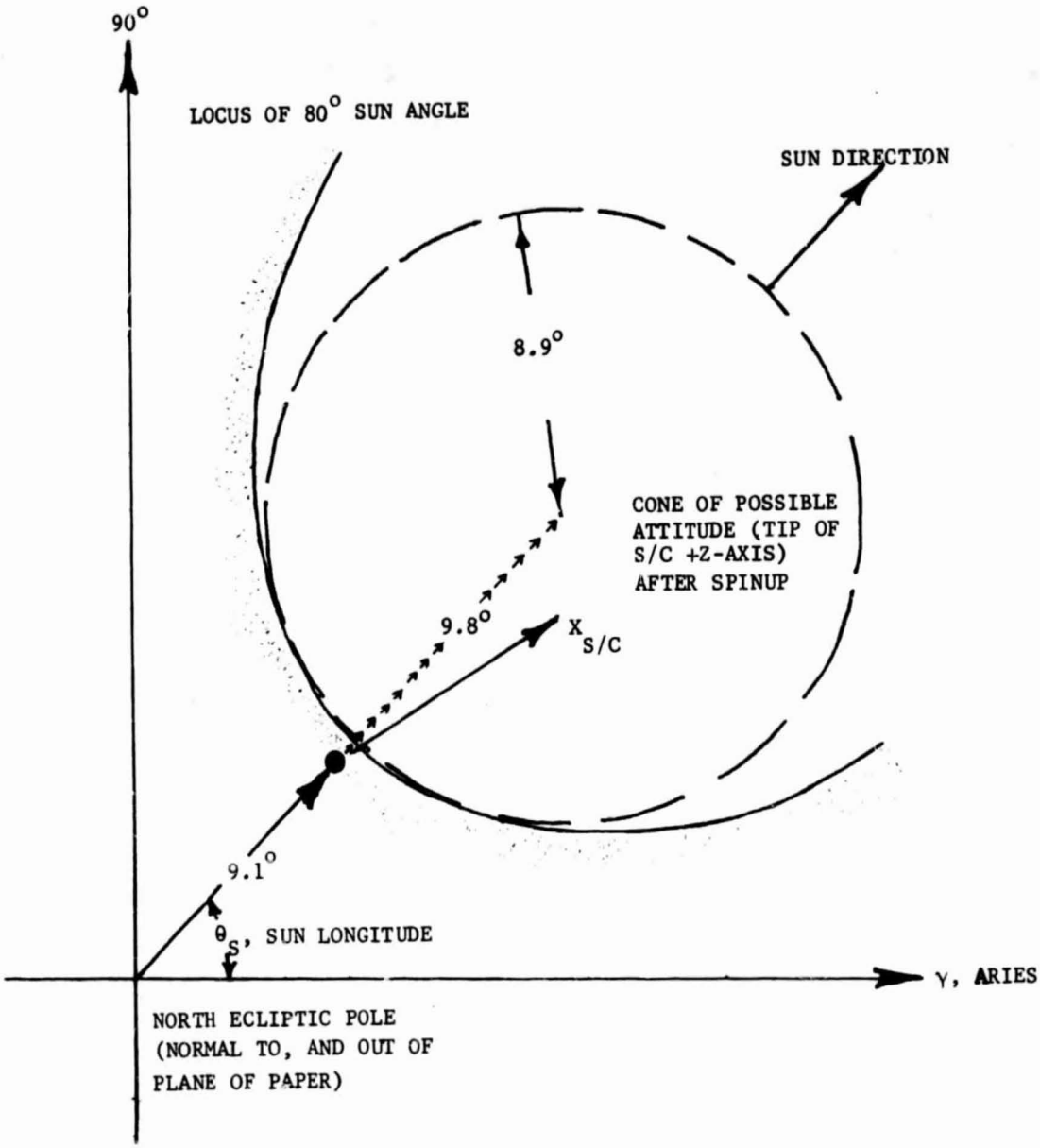
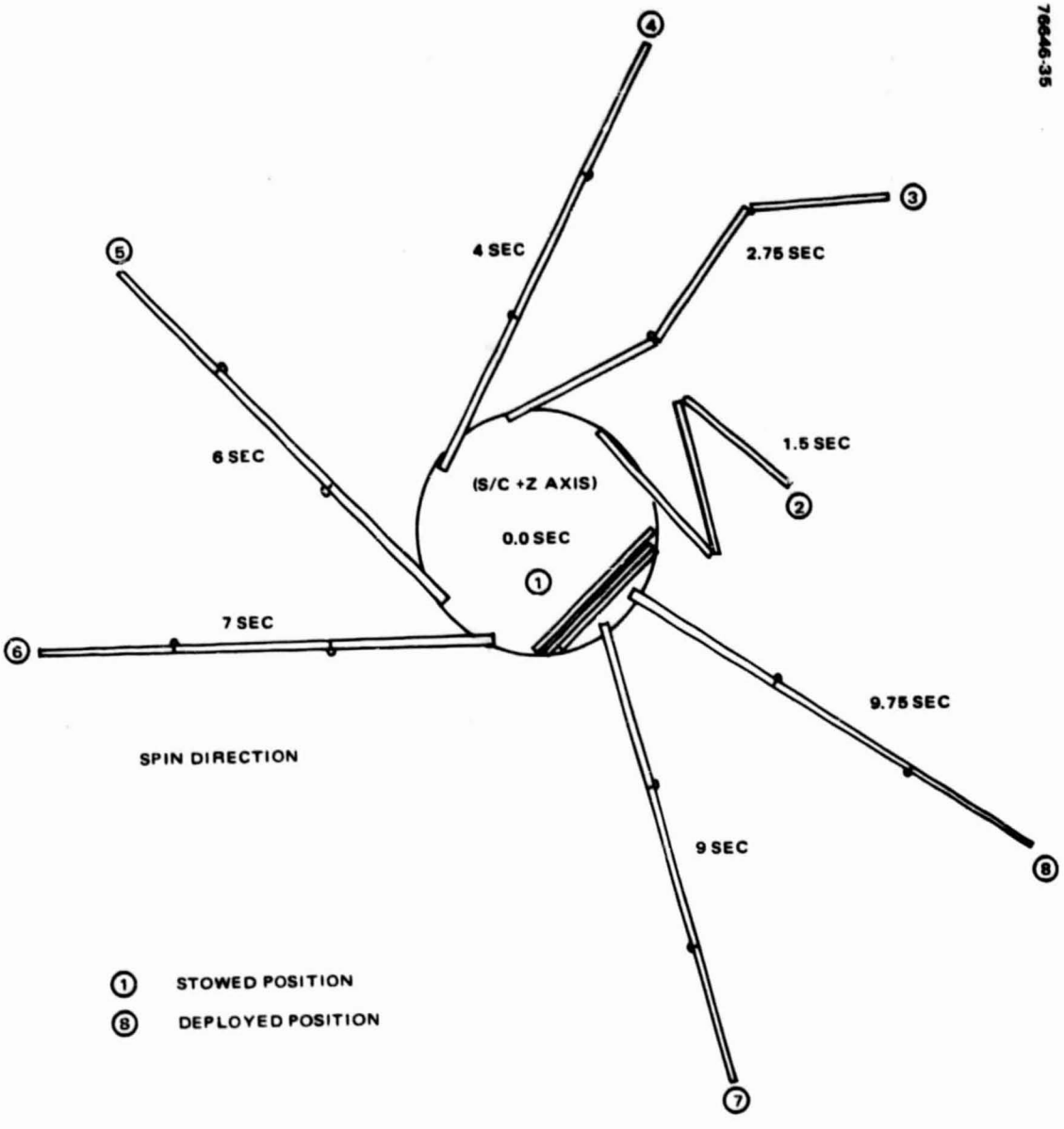
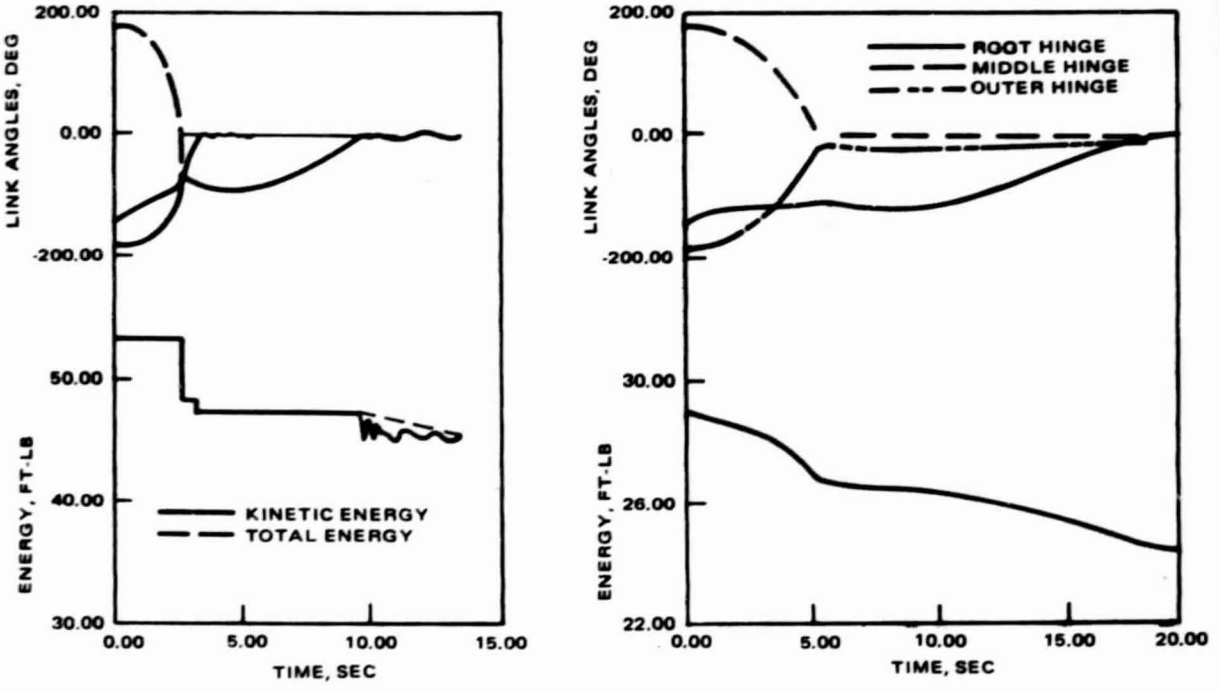


Figure 4.1.3.1-2. Simplified View of Orbiter Reorientation Maneuver



TYPICAL MAGNETOMETER BOOM DEPLOYMENT SEQUENCE

Figure 4.1.3.2-1. Boom Motion Relative to Spacecraft Coordinates



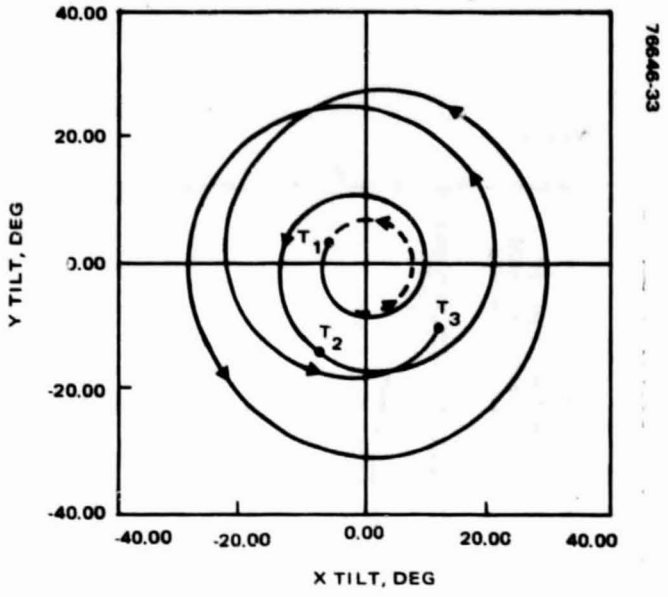
a) HIGH SPIN SPEED (6.8 RPM), NO DRAG TORQUE

b) LOW SPIN SPEED (5.0 RPM), MAXIMUM DRAG TORQUE

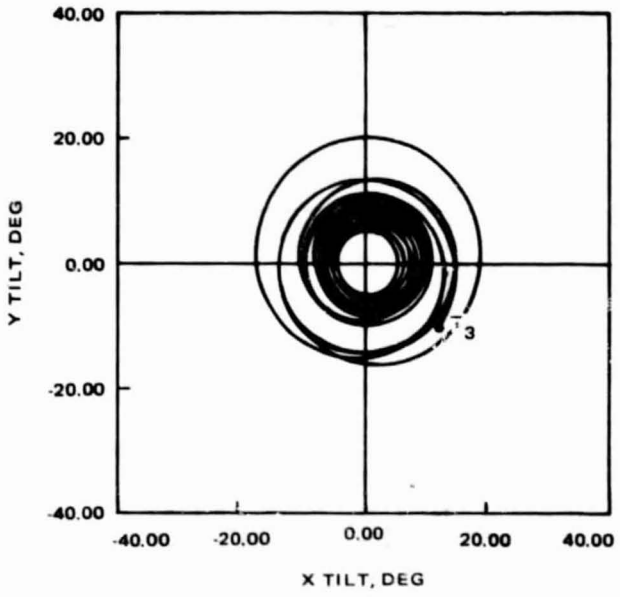
FIGURE 4.1.3.2-2. EFFECT OF SPIN SPEED AND DRAG TORQUE VARIATION ON DEPLOYMENT

Section No. 4.1.3.3
Doc. No. PC-402
Orig. Issue Date 2/28/78
Revision No. _____

Revision



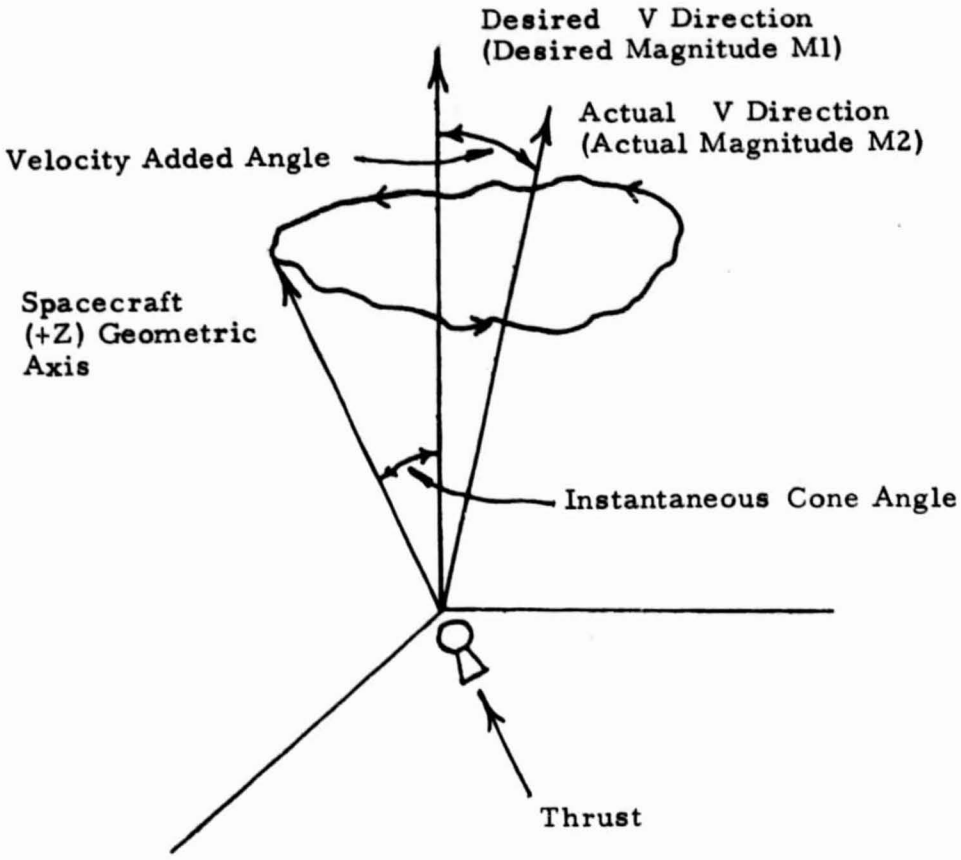
a) SPIN AXIS INERTIAL MOTION



b) SPIN AXIS INERTIAL MOTION

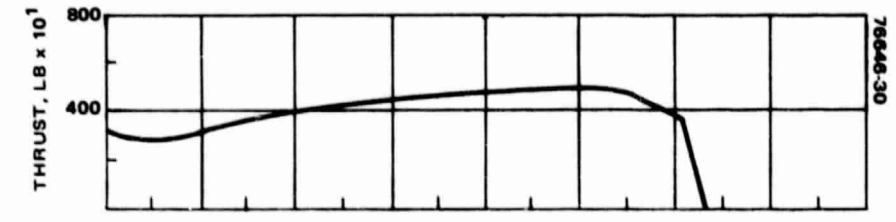
- INITIAL WOBBLE ~8.6°, SPIN SPEED 6 RPM
- T_1 = START OF DEPLOYMENT
- T_2 = END OF DEPLOYMENT
- T_2 TO T_3 = NUTATION ~22°
- T_3 = START SECOND SPINUP, 5 → 15 RPM
- NUTATION DAMPER REDUCES RESIDUAL NUTATION

FIGURE 4.1.3.2-3. INERTIAL TRACE OF SPIN (Z) AXIS FOLLOWING DEPLOYMENT

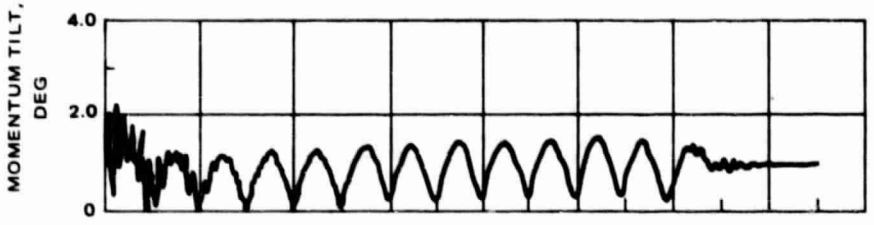


Effective Cosine Loss Angle=COS^{-1} (M2/M1)

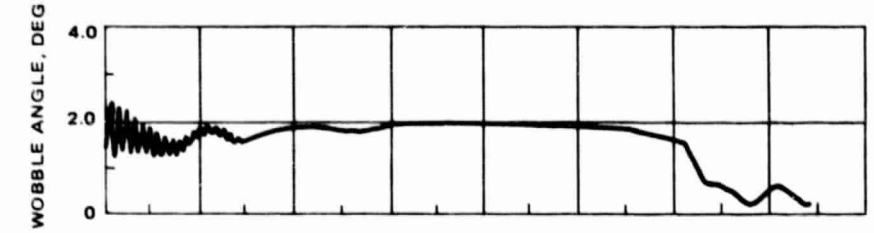
Figure 4.1.3.3-1. Effect of Spacecraft Wobble on Venus Orbit Insertion Maneuver Performance



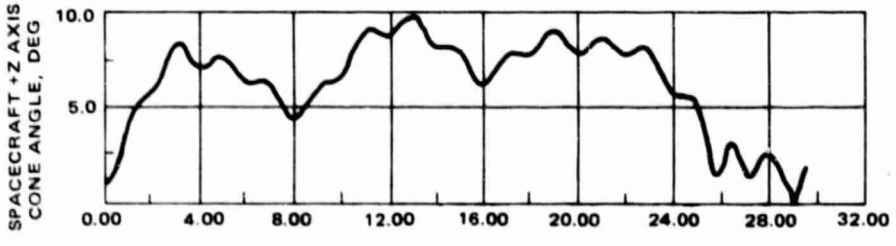
a) THRUST, LB



b) MOMENTUM TILT ANGLE, DEG

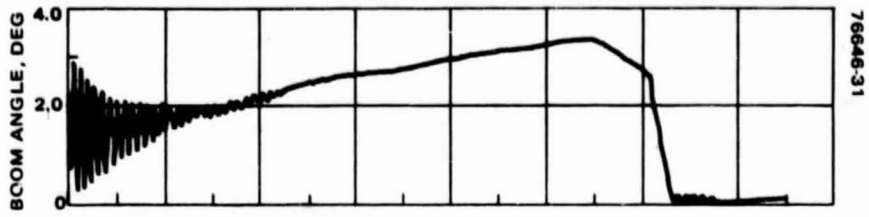


c) WOBBLE ANGLE, DEG

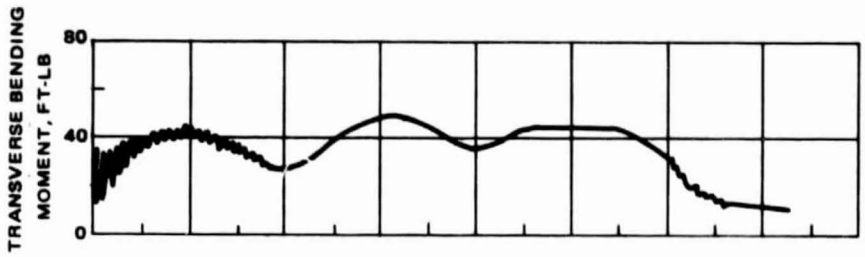


d) SPACECRAFT Z-AXIS TILT, DEG

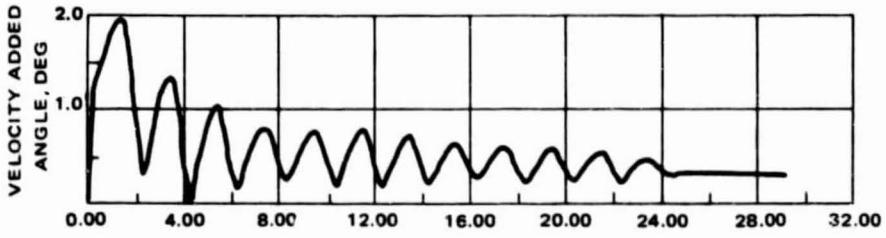
FIGURE 4.1.3.3-2. VENUS ORBIT/MANEUVER SPACECRAFT PERFORMANCE 1



a) BOOM AXIAL DISPLACEMENT, DEG



b) BEARING TRANSVERSE BENDING MOMENT, FT-LB



c) VELOCITY ADDED ANGLE, DEG

FIGURE 4.1.3.3-3. VENUS ORBIT INSERTION MANEUVER SPACECRAFT PERFORMANCE 2

4.2 KEY MISSION EVENTS

The step-by-step sequence of actions for each of several key mission events is described below.

4.2.1 Separation and Initial Spinup

4.2.1.1 Introduction. Orbiter spacecraft separation from the Centaur launch vehicle occurs at a zero spin rate and out of view of ground stations with command capability. Depending on launch date and time, ground station(s) become visible anywhere between 20 minutes and 4 hours after launch. To achieve attitude stability and prevent tumbling, spacecraft spinup must be accomplished as soon as possible after separation; much before initial ground station acquisition.

Orbiter spacecraft spinup is effected with simultaneous use of the spacecraft stored command processors. The spinup sequence of commands is loaded into the command memories prior to launch, and initiated upon spacecraft/Centaur separation. The two spinup thrusters are fired to effect a 6.5 rpm spin rate and the spacecraft is then configured for initial ground station acquisition, all via stored command. Centaur reorientation requirements and Centaur/Orbiter separation dynamics are presented in Section 4.1 of this document.

4.2.1.2 Command Memory Loading and Verification. The two Orbiter command memories will be loaded with their respective spinup command sequences about three hours prior to liftoff. This corresponds to the beginning of the standard 60-minute built-in hold in the countdown. Both memories will have their contents verified after loading. A final verification will be made approximately 15 minutes before liftoff, after the spacecraft has been switched to internal power. (Switching from external power to internal power does not affect the contents of either command memory). Detailed procedures for loading and verifying the command memories are delineated in Section 3.6.3.2 of this document.

4.2.1.3 Spacecraft Subsystem Launch Configuration. At liftoff, the Orbiter subsystems will be configured in the following operational status modes as listed in Table 4.2.1.3-1.

4.2.1.4 Command Memory Initialization. At liftoff the Stored Command Logic (SCL) 1 and 2 are in the "Arm Separation Switch" state. Upon closure of the spacecraft/attach fitting separation switches at separation, the SCL states switch to "immediate start" (initiation of the stored command sequences). Figure 4.2.1.4-1 depicts the functional spinup sequence implementation. One contact of each separation switch shorts its respective ADP "ON" command to ground while the switch is in the launch configuration. Each switch prohibits its respective command processor from executing the spinup sequence prior to separation, and circumvents a mission catastrophic failure due to a possible single point failure in the SCL. This failure causes a change of state in the SCL; e.g., "arm" to "run". (See Section 3.6.3.2 for further delineation of this failure mode). Separation Switches 1 and 2 inhibit ADP 1 ON and ADP 2 ON (ADP19 via COM assignment 5D28 and ADPB9 via COM assignment 6D31), respectively. ADPs 1 and 2 may still be commanded ON prior to launch via commands ADPA9 and ADP29 (via COM assignments 6D28 and 5D31), respectively. The stored sequences will only send ADP 1 commands via COM 5, and ADP 2 commands via COM 6.

4.2.1.5 Stored Command Sequence. Table 4.2.1.5-1 presents the detailed sequence of commands stored in each SCP along with time of execution relative to separation. Reasons for time delays, etc., are given in the "comments" column of the Table.

4.2.1.6 Initial Acquisition. The spacecraft Configuration at initial ground station acquisiton is shown in Table 4.2.1.6-1.

TABLE 4.2.1.3-1

ORBITER SUBSYSTEM LAUNCH CONFIGURATION

Subsystem/Unit	Operational Configuration
COMMUNICATIONS	
Transponders	• Exciter 1 ON/2 OFF • Restore Coherent Mode (ON) • Hi Mod Index Select
Power Amplifiers	• Power Amp 1 ON • Power Amps 2, 3, 4 OFF
X-Band Transmitter	• X-Band Transmitter OFF
Switch Drivers	• Exciter 1 Select • Power Amp 3 Select • Forward Omni Select • Amp 1 Low Power Select • Low Power to Despun Antennas Select • Backup/Forward Omni Antenna Select • Receivers Normal Select (via command Processor Configure Command)
COMMAND	
Command Processors	• Command Processors 1 and 2 ON (Both memories in "Arm separation switch" state)
COMs	• All 6 COMs ON
PCU	• PCU Disarmed
CONTROLS	
Attitude Data Processors	• ADP 1 OFF • ADP 2 OFF • JCE Output Buffers Disabled
Despin Control Electronics	• DCE 1 and 2 OFF
BAPTA	• Motor Drives 1 and 2 OFF
High Gain Antenna Positioner	• HGA Positioner OFF
Star Sensor	• All Star Sensor Channels OFF
DATA HANDLING	
Telemetry Processor	• Telemetry Processor 1 ON/2 OFF • Orbiter Engineering Format • 256 Bits/Sec • Data ON • Convolutional Encoding OFF • DSU 1 and 2 OFF
PCM Encoder	• PCM Encoder 1 ON/2 OFF
DIM	• All 8 DIMs ON
POWER	
Bus Limiters	• All 7 Bus Limiters Enabled
Charge/Discharge Controller	• Lo Rate Charge Select (both relays - both batteries) • Primary Discharge Reg Select (both batteries)
UV/OL Control	• UV/OL Power System Protection OFF • Pre-Charge OFF • Science Bus ON (to Power OIR Heater)
PROPULSION/THERMAL	
Latch Valves & Heaters	• All Jet Heaters ON • Primary Tank/Line Heaters Select • OIM Heater ON • Latch Valves 1 and 2 Closed
SCIENTIFIC INSTRUMENTS	• All Science OFF • OIR Heater ON

TABLE 4.2.1.5-1

FLIGHT SEPARATION/SPINUP SEQUENCE

Step No.	Relative Start Time	Command Processor 1			Command Processor 2			Comments	Telemetry Verification	
		Command	Command Mnemonic	Hex	Command	Command Mnemonic	Hex		Mnemonic	Title & Status
	0:00:00.000	Separation (All times below referenced to this event)			Separation (All times below referenced to this event)			Separation. Separation switch signals initiate SCL command sequences and enable ADP on commands.		Telemetry verification channels in the event that downlink data stream is received.
1	0:00:00.000	Time Delay (= 2.250 sec)		24 00 00	Time Delay (= 2.250 sec)		24 00 00	• Input time delay code equivalent to 2.250 sec. to circumvent an SCP failure mode in which time delay codes in one SCP are ignored and remainder of stored commands executed in 1/8 sec intervals. Second SCP will then reconfigure (after 2.250 sec delay) to execute nominally desired spinup. • This time delay also allows for clearance (& an expected tipoff) from the Centaur before the S/C jets are fired.		

TABLE 4.2.1.5-1 (Continued)

Step No.	Relative Start Time	Command Processor 1			Command Processor 2			Comments	Telemetry Verification	
		Command	Command Mnemonic	Hex	Command	Command Mnemonic	Hex		Mnemonic	Title & Status
2	0:00:02.375	ADP 1 ON	ADP19	A8 00 1C	ADP 2 ON	ADPB9	B0 00 1F		AADP1S AADP2S	(ADP 1 ON/OFF = ON (1) (ADP 2 ON/OFF = ON (1)
3	0:00:02.500	ADP 1 Mode Select • PLL Spin Range: 4 - 8.85 rpm. • PLL Loss of Lock Inhibited. • Star Gate A: Channel 1 • Star Gate B: Channel 1 • Midrange Sun Sensor • Sun Select • SRR Select • Star Normal • Disable Sun Gate • SRR Normal	ATQ63	AE 41 98	Latch Valve 2 Open	VALB9	90 00 1D	For ATQ63 (& ATQ6C, ahead): • PLL loss of lock inhibited (Cannot track large spin rate changes) • All other parameters - "don't care."	VALV2S 	Latch Valve 2 Open/ Closed = Open (1) Cannot verify ADP status - TM is meaningless with both ADPs ON.
4	0:00:02.625	ADP 1 Jet Control • Enable Jet 1 • Enable Jet 3 • Normal Fire Mode • Pulse Width = 512 msec • Time Count • Continuous Fire Sel. • Inhibit Spin Rate Detector.	ATQ62	AE 8A 1A	ADP 2 Mode Select • PLL Spin Range: 4 - 8.85 rpm. • PLL loss of lock Inhibited. • Star Gate A: Cnl. 1 • Star Gate B: Cnl. 1 • Midrange Sun Sen. • Sun Select • SRR Select • Star Normal • Disable Sun Gate • SRR Normal	ATQ6C	B6 41 98	For ATQ62 (& ATQ6B ahead): • Select R1, R3 (spinup jets) • Select time count • Select cont. mode • Spin rate detector inhibit (spinup begins below 3.7 rpm)		Cannot verify ADP status - TM is meaningless with both ADPs ON.

TABLE 4.2.1.5-1 (Continued)

Step No.	Relative Start Time	Command Processor 1			Command Processor 2			Comments	Telemetry Verification	
		Command	Command Mnemonic	Hex	Command	Command Mnemonic	Hex		Mnemonic	Title & Status
5	0:00:02.750	Latch Valve 1 Open	VAL19	88 00 1B	ADP 2 Jet Control • Enable Jet 1 • Enable Jet 3 • Normal Fire Mode. • Pulse Width = 512 msec. • Time Count • Continuous Fire Select. • Disable Spin Rate Detector.	ATQ07	B6 8A 1A		VALV1S	Latch Valve 1 Open/ Closed = Open (1)
6	0:00:02.875	Latch Valve 2 Open	VAL29	88 00 1D	ADP 2 Jet Count Down • Jet Countdown Magnitude = 45 counts = 23.040 sec	ATQ0D	B6 C8 40		VALV2S	Latch Valve 2 Open/ Closed = Open (1).
7	0:00:03.000	ADP 1 Jet Countdown • Jet Countdown Magnitude = 45 counts = 23.040 sec.	ATQ04	AE CB 40	ADP 2 JCE Buffer Enable.	ATQ01	B6 10 00	• Input time code equivalent to 23.04 sec jet firing duration to bring S/C from 0 to 6.5 rpm.		Not Verifiable.
8	0:00:03.125	ADP 1 JCE Buffer Enable.	ATQ09	AE 10 00	Latch Valve 1 Open	VALA9	90 06 1B			Not Verifiable.
9	0:00:03.250	ADP 1 Jet Fire Interlock	ATQ11	AE 50 00	ADP 2 Jet Fire Interlock.	ATQ0K	B6 50 00			Not Verifiable.

TABLE 4.2.1.3-1 (Continued)

Step No.	Relative Start Time	Command Processor 1			Command Processor 2			Comments	Telemetry Verification	
		Command	Command Mnemonic	Hex	Command	Command Mnemonic	Hex		Mnemonic	Title & Status
10	0:00:03.375	ADP 1 Jet Fire	ATC12	AE D0 00	ADP 2 Jet Fire	ATQ01	B6 D0 00	• Spinup nominally begins.	VJET1T VJET3T	Radial Jet 1 Temp Increase. Radial Jet 3 Temp Increase.
11	0:00:03.500	Time Delay (= 24.0 sec)	CMTQ1	01 80 00	Time Delay (= 24.0 sec)	CMTQ2	01 80 00			
12	0:00:27.625	ADP 1 JCE Buffer Disable.	ATQ1Ø	AE 90 00	ADP 2 JCE Buffer Disable.	ATQØJ	B6 90 00	• To backup nominal JCE countdown register timeout for thruster shutdown.		
13	0:00:27.750	ADP 1 Jet Countdown • Jet Countdown Magnitude = 0°.	ATQØ4	AE C0 00	Latch Valve 2 Close.	V/-1 BØ	90 00 1E	• Further backup of ADP 1 countdown.		
14	0:00:27.875	ADP 1 Jet Control • Disable All Jets • Normal Fire Mode • Pulse Width = 0.128 Sec. • Time Count Select • Pulse Select • Enable Spin Rate Detector.	ATQØ2	AE 80 11	ADP 2 OFF	ADPBØ	B0 00 20	• ADP 2 turnoff permits valid attitude measurement & ADP status TM. • Further backup of Jets shutoff.	AADP2S	ADP 2 ON/OFF = OFF (0).
									AJMAGC	JCE Countdown = All zeros.
									AJET1S thru AJET7S	Jet Select Bits = All zeros.
									AJETMS	Normal/Alternate Fire Mode = Normal (1)
									APUI LS	Pulse Width Select = 128 msec (0)
									AMAGCS	Pulse/Time Count Select = Time (0).
									AJETMS	Cont./Pulse Fire Select = Pulse (1).
									ASDETS	Spin Rate Detector = Enabled (1).
									AJCE1S	JCE 1 Buffer Output State = OFF (=0)
									AJCE2S	JCE 2 Buffer Output State = OFF (=0).

TABLE 4.2.1.5-1 (Continued)

Step No.	Relative Start Time	Command Processor 1			Command Processor 2			Comments	Telemetry Verification	
		Command	Command Mnemonic	Hex	Command	Command Mnemonic	Hex		Mnemonic	Title & Status
14 (Cont'd)									AJCEFS	Jet Fire Enable Status = Disabled (=0).
15	0:00:28.000	ADP 1 Measurement Select • SRR to SRR • SRR to PSI2	ATQ01	AE 00 05	Command Processor 2 Configure • SCL 2 Standby	CPCQ2	B5 00 07	• SCI 2 is set to Standby to insure no repeat of spinup sequence. • SCI 2 is not yet turned OFF in order to retain memory contents in the event the spinup did not occur, & it is elected to repeat execution of the SCI 2 contents. If the Standby command is effective, the subsequent commands in this column will not get executed; only a real-time command will change the SCI 2 state.	AM1ADS	Measurement A Address = SRR to SRR.
									AM2ADS	Measurement B Address = SRR to PSI2.
									ATTM	Attitude Measurement Data is Plausible.
									CLOG2S	SCI 2 State Status = Standby (=1111).
16	0:00:28.125	Latch Valve 1 Close.	VAL10	86 00 1C	Command Processor 2 Configure. • SCL 2 STOP.	CPCQ2	B5 00 1B	• SCL 2 STOP is backup to the previous command.	VALV1S	Latch Valve 1 Open/Closed = Closed (0).
									CLOG2S	SCL 2 State Status = STOP 1 (=0010), only if standby command is ineffective. Otherwise, STOP command has no effect in Standby State.
17	0:00:28.250	DSU 1 ON	DSU19	98 00 20	Command Processor 2 Configure • SCL 2 OFF.	CPCQ2	B5 00 3F	• SCL 2 OFF is backup to previous command.	CSCL2S	SCL 2 Status = OFF (0).

TABLE 4.2.1.5-1 (Continued)

Step No.	Relative Start Time	Command Processor 1			Command Processor 2			Comments	Telemetry Verification	
		Command	Command Mnemonic	Hex	Command	Command Mnemonic	Hex		Mnemonic	Title & Status
18	0:00:28.375	Time Delay (=1.875 Seconds)	CMTQ1	78 00 00	SCI 2 OFF	MEM28	90 00 2B	• Time Delay allows DSU 1 to initialize. • SCI 2 OFF discrete command is backup to the previous OFF cmd.	CSCI2S	SCI 2 Status = OFF (0).
19	0:00:30.375	Telemetry Processor 1 Control. • ACS Format • 256 bps • Store DSU 1 • Subcarrier ON • Convolutional Encoder OFF • Data ON	TPCQ1	9E 46 87	NO FURTHER COMMANDS IN CP2			• ACS Format Selected. • DSU 1 Commanded to STORE.	FORMAT BIT RATE	ACS Format Selected 256
20	0:00:30.500	Time Delay (=2M10S)	CMTQ1	04 10 00				Store 2 Major Frames of ACS Format Data.		
21	0:02:40.625	Telemetry Processor 1 Control • Engineering Format • 128 bps • Store DSU 1 • Subcarrier ON • Convolutional Encoder OFF • Data ON	TPCQ1	9E 04 87				• Return to Engr. Format. • Reduce Bit Rate to extend time for data.	FORMAT BIT RATE	Engineering Format Selected 128
22	0:02:40.750	DSU 2 ON	DSU29	98 00 22				• DSU 2 Turned ON for additional capacity.		
23	0:02:40.875	Time Delay (=4M49S)	CMTQ1	08 48 00				• Time Delay allows DSU 2 to initialize.		
24	0:07:30.000	ADP 1 Roll Index Delay • Roll Index Delay Magnitude = 270°	ATQ06	AE A0 0C				• To initialize RIP (No preferred state for RIP delay upon ADP turn ON	ARIPAD	Roll Index Delay Magnitude = 270 Degrees.

TABLE 4.2.1.5-1 (Continued)

Step No.	Relative Start Time	Command Processor 1			Command Processor 2			Comments	Telemetry Verification	
		Command	Command Mnemonic	Hex	Command	Command Mnemonic	Hex		Mnemonic	Title & Status
25	0:07:30.125	Time Delay (=19.875S)	CMTQ1	7C 80 00				• Allows at least one s/C revolution for RIP to stabilize.		
26	0:07:50.125	DCE 1 ON/2 OFF.	DCE19	88 00 28				• DCE Turned ON for thermal integrity.	ADCE1T ADCE1V	DCE 1 Temp rises & settles. DCE 1 Power Supply Voltage = 5 V.
27 to 65		Various DCE Configure, Science Instrument Turn ON and Configure, TP Control, and Time Delay Commands follow - All subject to change.								
66	1:03:00.500	Command Processor 1 Configure • SCL 1 Standby.	CPCQ1	AD 00 07				Analogous to Steps 16, 17 and 18.		
67	1:03:00.625	Command Processor 1 Configure • SCL 1 STOP.	CPCQ1	AD 00 1B						
68	1:03:00.750	Command Processor 1 Configure • SCL 1 OFF.	CPCQ1	AD 00 3F						
69	1:03:00.875	SCL 1 OFF	MEM1#	88 00 2B						

TABLE 4.2.1.6-1

ORBITER SUBSYSTEM ACQUISITION CONFIGURATION

Subsystem/Unit	Operational Configuration
Communications	• Same as Launch (See Table 4.2.1.3-1)
Controls	• Same as Launch, except ADP 1 ON
Command	• Same as Launch, except SCL 1 and 2 "OFF"
Data Handling	• Same as Launch, except DCU will be turned ON for storing pre-acquisition engineering data
Power	• Same as Launch (see Table 4.2.1.3-1)
Propulsion/Thermal	• Same as Launch (see Table 4.2.1.3-1)
Scientific Instruments	• TBD

4.2.2 Magnetometer Boom Deployment. The purpose of this section is to describe the Orbiter Spacecraft mission operations sequence from the time of initial ground station acquisition, through magnetometer boom deployment, and up to the beginning of the cruise phase. This entire command/event sequence is delineated in Table 4.2.2-1.

TABLE 4.2.2-1

ORBITER MAGNITOMETER BOOM DEPLOYMENT SEQUENCE OF EVENTS

Relative Time	CMD Mnemonic	Command Title or Event	Telemetry Verification	
			TLM Mnemonic	TLM Title
A = 0 (≈ L - 1H30M)		Initial Ground Station Acquisition.		DSN Ground Station will acquire non-coherent spacecraft telemetry downlink. Coherent link will be subsequently established.
A - 3 Min.		Switch over to 2048 bps TM rate; then Power System Protection ON.		
A - 10 Min.		Verify spacecraft operating modes.		Spacecraft configuration and operating modes as specified in 4.2.1.6.
A - 20 Min.		Perform attitude and/or spin rate trim maneuvers (if necessary). See Sections 4.3.1.1 and 4.3.1.3.		
A - 30 Min.		Perform attitude determination (see Section 4.3.2) (if trim maneuver had been performed).		
A - 40 Min.	ORD13	Arm Mag Boom, OETP boom deployment, ONMSB/O Hats, and ORAD.		No "Arm" status TM exists - Min. 6 sec and max 14 sec between ARM & FIRE commands.
A - 40M12S	ORD14	Release Mag Boom.	SBOOMS	Mag Boom deployed status (= 1) NOTE: Boom will require approximately 10 sec to deploy.
			ATTM1Z ATTM2Z	ATTM1Z and ATTM2Z measuring $\Psi\Psi$ & $\Psi\Psi_2$ will change, showing spin rate change and nutation as described in Section 4.1.1.2.
A - 40M24S	ORD10	Disarm PCU (To backup automatic disarm function).		
A - 40M36S	MAG19	OMAG Power ON.		
A - 41 Min.		Perform attitude and spin rate determination (See Section 4.3.2).		
A - 55 Min.		Perform spinup to 15 rpm (see Section 4.3.1.3).		
A - 1.0 to 5.0 Hrs.		Recheck attitude and spin rate (after residual nutation damps out) (See Section 4.3.2).		
A - 1 Hr.	UVS10	OUVS Power OFF.		
	IMS10	OIMS Power OFF.		
A - 1 H30 M	CPP19	OCPP Power ON.		
		OCPP Checkout.		
A - 5.0 Hrs.		Perform precession to north ecliptic "Cruise" attitude (see Section 4.3.1.1).		

Upon initial DSN ground station downlink telemetry acquisition, the Orbiter subsystems will be configured in the operational status mode listed in Section 4.2.1.6. This initial acquisition mode incorporates the attitude measurement (ACS) data format with $\Psi\Psi$ and $\Psi\Psi_2$ as the selected measurements. Attitude and spin rate determinations are begun by utilizing these two measurements to calculate spin speed and sun aspect angle respectively. A complete attitude determination is then performed as described in Section 4.3.2. The star Altair is visible for all nominal attitudes. Attitude measurements may be intermittent during this initial acquisiton phase due to combined effects of the 9-deg coning angle (because of the stowed boom) coupled with the approximately 10-deg of residual nutation resulting from initial post-separation spinup. The effect of spacecraft dynamics on attitude measurements at this point in the mission are further discussed in Section 4.1.1.2.1.

The boom will nominally be deployed when sun angle, communications angle (Earth look angle) spin speed and residual nutation are within satisfactory limits as listed below. Also, a DSN ground station command uplink must be established prior to boom deployment.

- Average sun angle -- 62 to 80 deg (sun angle must be less than 80 deg to maintain OMAG boom and hinges above 0°F)
- Average Earth look angle -- 43 to 71 deg (must be less than 100 deg to insure use of forward omni antenna)
- Spin rate -- 6.5 ±0.3 rpm
- Residual nutation ≤ 10 deg
- Coning angle of spacecraft +z axis about spin axis ≤ 9°

If the above criteria are not met, an attitude and/or spin trim maneuver will be necessary as shown in Table 4.2.2-1.

The magnetometer boom will be deployed by ground commands as listed in Table 4.2.2-1. Upon receipt of the specified commands, pin pullers

release the caged boom assembly, and centrifugal force developed by the Orbiter's spin drives the three hinged boom segments outward. Detent pins at each hinge automatically lock the deployed assembly approximately 10 seconds after start. The boom root hinge should be the last of the three hinges to lock into place and as such, it utilizes a microswitch to indicate the stowed/deployed status of the magnetometer boom. Monitoring the telemetry status of this switch along with the change in spin rate described below will be the most immediate and absolute indications of successful boom deployment. The average spacecraft attitude remains the same after deployment. However, the spin rate will decrease to nominally 5.4 rpm and deployment of the boom will introduce a 22-deg average nutation. The attitude measurement data nominally expected at this time is described in Section 4.1.1.2.1.

Nutation damper performance at this low spin rate and high nutation angle is limited; time constant >4 hours. The nominal mission plan described in Table 4.2.2-1 calls for spinup to 15 rpm after deployment. Increasing the spin speed by a factor of three reduces the average nutation angle from 22 to 7 degrees. The higher spin speed results in a nutation damping time constant <30 minutes even at the larger nutation angles and the residual nutation is <0.1 deg within 2 hours of the secondary spinup.

In the event the magnetometer boom fails to deploy, an assessment of telemetry data should be undertaken. No change in the attitude measurement data indicates that the boom was not uncaged, in which case the appropriate "arm" and "fire" commands listed in Table 4.2.2-1 should be repeated.

A change in attitude measurement data without the root hinge switch indicating "deployed" status indicates the centrifugal force (i.e., spin rate) was not great enough for the boom to fully deploy. In this event, the spacecraft spin rate will gradually be increased (in increments of

approximately 1 rpm) until the root hinge switch indicates "deployed".

After the steady-state spin about the +Z axis is achieved, the attitude will be nominally reoriented to within 2 deg of the north ecliptic pole in preparation for the cruise phase. Prior to this 19 ±9 deg precession, star sensor measurements are obtained to define attitude as well as maneuver parameters necessary in executing the sequence in Table 4.2.2-1. The maneuver to the north ecliptic pole will nominally be based on attitude information so that when attitude measurements are obtained near the pole, an initial calibration of axial jet(s) pulsed performance can be made.

4.2.3 Orbit Insertion Maneuver. The purpose of this section is to describe the Orbiter Spacecraft mission operations sequence from approximately two days before arrival at Venus, through firing of the Orbit Insertion Motor (OIM), and up to the beginning of nominal on-orbit operations. This entire command/event sequence is delineated in Table 4.2.3-1.

The orbit insertion maneuver begins with the Orbiter spacecraft in the interplanetary cruise configuration as detailed below and in Section 4.2.1.6. The nominal spacecraft attitude is one with the positive spin axis oriented toward the north ecliptic pole. Spacecraft spin rate is 15 rpm and the high gain antenna (HGA) is despun and pointing towards earth to maintain uplink and downlink communications.

The spacecraft +Z axis attitude required at orbit insertion is nominally directed parallel and opposite to the spacecraft velocity at closest approach to Venus. Approximately 8 minutes before closest approach, the spacecraft will be occulted from earth by Venus. The celestial longitude and latitude of the insertion attitude depend on the day of launch and time of arrival. The range of possible attitudes with corresponding sun and earth look angles are presented in Section 4.1.1.2.3. The precession

TABLE 4.2.3-1

ORBIT INSERTION MANEUVER SEQUENCE OF EVENTS

Relative Time	Command Mnemonic	Command Title or Event	Telemetry Verification	
			TLM Mnemonic	TLM Title
E - 48 Hours		• Begin Orbit Insertion Maneuver		
		• Verify Spacecraft Operating Modes (All Science OFF in anticipation of being exclusively in ACS or Engineering TM formats)		Spacecraft in cruise configuration with all scientific instruments turned OFF, (including RF low power mode, HGA selected, 2048 bps selected, engrg. format).
E - 47.5 H		• Transfer to DSN 64-meter ground station.		Lockup of uplink and downlink at 64-meter ground station.
		• Perform Attitude & Spin Rate Determination (See Section 4.3.2).		
E - 47 H		• Switch to Forward Omni Antenna, RF HI Power, & highest usable bit rate (assumed to be 16 bps in this sequence):		
	TPCQ1 or TPCQA	Telemetry Processor Control - Bit rate select (to 16 bps)		• Loss of bit sync. • Bit Sync re-established.
			Bit rate Selection	Indicates 16 bps.
	MIL11 or MILA1	Low Mod Index Select		Momentary disruption in data in worst case - likely demod sync, bit sync, & frame sync will be retained as carrier suppression is decreased by 6.4 dB.
	AMP1Ø or AMPAØ	Power Amplifiers 1 and 2 OFF. • Verify loss of downlink before proceeding.		Loss of downlink carrier.
	ANT12 or ANTA2	High Power to Despun Antennas Select		No downlink - not verifiable in real time.
	ANT22 or ANTB2	Backup/Forward Omni Antenna Select		No downlink - not verifiable in real time.
	AMP12 or AMPA2	Amplifier 1 to High Power Select		No downlink - not verifiable in real time.
	AMP19 or AMPA9	PA 1 ON/2 OFF.		Carrier may be detectable at multiple of rest frequency of VCO in Rcvr 1 (should be phase-locked).
	AMP39 or AMPC9	PA 3 ON/4 OFF.		Sufficient downlink should be present for ground receiver lock on the carrier demod sync with the subcarrier; & bit sync on the data. Frame sync will follow later because of the low bit rate.
		• Verify Comm S/S telemetry status before proceeding.	RANT1S	Aft Omni/despun antennas to HI/LO pwr = 0 (despun to HI pwr)
			RAMP1S	AMP 1/2 to HI/LO pwr = 1 (AMP 1 to HI).
			RANT2S	HGA/Low Gain antennas switch pos. = 0 (Low gain ant. selected)
			RAMP1T	PA 1 temperature settles at "ON" level.
			RAMP1W	RF power output 1 = 9.5 watts (Nom.).
			RAMP3T	PA 3 temperature rises & settles at "ON" level.
			RAMP3W	RF power output 3 = 9.5 watts (Nom.).

TABLE 4.2.3-1 (Continued)

Relative Time	Command Mnemonic	Command Title or Event	Telemetry Verification	
			TLM Mnemonic	TLM Title
E - 46.5 Hours		Perform precession to orbit insertion attitude. (See Section 4.3.1.1). (Requires ≈15 minutes).		
F - 46.25 H		Perform Attitude Determination (See Section 4.3.2) (Requires ≈45 minutes at 16 bps for data collection and processing).		
E - 45.5 H		Perform Attitude Trim (if necessary) (See Section 4.3.1.1).		
E - 45 H		Spinup the HGA:		HGA is set to spinning mode during OIM burn in order to minimize the effect of transverse torque resulting from antenna static imbalance & OIM thrust along the S/C -Z axis. HGA is spun up at this time in preparation for spinup of entire S/C to 30 rpm.
	MTR1Ø or MTRAØ	Motor Drives 1 and 2 OFF	AMTR1S	Despin Motor Drive 1 ON/OFF = 0.
			AMTR2S	Despin Motor Drive 2 ON/OFF = 0.
			PBUSLI	S/C loads current decreases 150 milliamps if one motor drive was ON.
			ATTM1Z ATTM2Z	SRR to MIP time interval measurement increases until SRR to MIP phase angle becomes 360°; then decreases in one big step as phase angle exceeds 360° and resumes increasing in a repeat of this pattern. When HGA is spinning at rotor speed, no MIP is generated & this measurement, if still selected, will cause a hang-up on the last valid value of either ADP measurement.
		While HGA is spinning up, select new P.I. spin range for later use:		
	ATQØ3 or ATQØC (Address #2)(Quantitative Command)	ADP Mode Select. (Select 16.0 to 35.4 rpm Spin Range)	ASPINS	SRR Spin Range Select = 01 (16.0 to 35.4 rpm selected).
		Perform S/C Spin-up to 30 rpm. (See Section 4.3.1.3). (Requires ≈1.5 Minutes).		
		Perform Spin Speed Determination and Trim, if necessary. (See Sections 4.3.2 and 4.3.1.3).		
		Despin HGA, but retain Comm link via Omni:		Despin from 30 rpm is done at this time in order to obtain more accurate attitude determination.
	MTR19 or MTR29	Motor Drive 1 ON.	AMTR1S	Despin Motor Drive 1 ON/OFF = 1.

TABLE 4.2.3-1 (Continued)

Relative Time	Command Mnemonic	Command Title or Event	Telemetry Verification	
			TLM Mnemonic	TLM Title
[E] - 45 H (Continued)		Antenna will despin and assume a position in azimuth and elevation based upon what had been last commanded in the DCE configure commands.	PBUSLI	S/C Loads current increases 150 milliamp.
			ATTM1Z ATTM2Z	SRR to MIP time interval measurement reads the same as when HGA was last despun.
			ARATEQ	Rate Torque - Assumes a nominal 2.75V (precalibrated) with no torque commands present.
			APOSTQ	Position Torque - Assumes a nominal 2.5V (precalibrated) with no torque commands present.
			AMOTRI	Despin Motor Drive Current - Assumes nominal zero amps with no torque commands present.
[E] - 44 H		Perform Final Attitude Determination and Trim (if necessary). (See Sections 4.3.2 and 4.3.1.1).		
[E] - 42.5 H		Allow HGA to spin up for OIM burn. (Repeat step for HGA spinup as at [E] - 45 Hours).		HGA again is spunup for OIM burn for reason stated earlier.
		Determine precise occurrence of next 4096 second clock transition. Resolve in Universal Time UT relative to time of closest approach.	DCLOK1 DCLOK2 DCLOK3	Spacecraft clock (24 bits): Calculate nearest multiple of 4096 seconds. (Second MSB of DCLOK2 is 4096 second clock bit).
[E] - 42 H		Load and Verify OIM firing sequence as shown in Figure 4.2.3.-1. (See Section 3.6.3.2).		
[E] - ≥40.5 H = t_0	CPCQ1, STORCMD - TIMED	Command Processor 1 Configure (Timed Start).	CMEM1C CLOG1S	Verified at next transition of 4096 second clock bit from "1" to "0": Command Memory address pointers will increment one, and SCL state will change from "TIMED START 1" (=0101) to "RUN" (=1100).
t_0 12 Sec.	CPCQ2, STORCMD - TIMED	Command Processor 2 Configure (Timed start).	CMEM2C CLOG2S	
[E] - 40 H		Begin monitoring Spacecraft Engineering status and assimilating tracking data.		Monitor Engineering Format Telemetry data points to insure there is no anomalous behavior.
[E] - 6 H		Reload and verify Command Processor 1 with updated OIM firing time. (See Section 3.6.3.2). Start via 4096 second clock and verify as to t_0 [E] - 40.5 H).		
[E] - 5 H		Reload and verify Command Processor 2 with updated OIM firing time. (See Section 3.6.3.2). Start via 4096 second clock and verify as at t_0 - 12 sec.		
[E] - ≥8 Min.		Begin Occultation.		Loss of Downlink Telemetry.

TABLE 4.2.3-1 (Continued)

Relative Time	Command Mnemonic	Command Title or Event	Telemetry Verification	
			TLM Mnemonic	TM Title
E - 10 Sec.	ORD11 or ORDA1 (Stored Command)	ARM Orbit Insertion Motor. (See Section 3.6.2.3)		Minimum 6 sec & maximum 14 sec between ARM & FIRE cmds.
E - 0	ORD12 & ORDA2 (Stored Commands)	FIRE Orbit Insertion Motor.		
E + ≈25 Sec.		End of OIM Burn.		
				PCU Disarm command is not required as ARM OIM cmd had armed only OIM fire squibs.
E + ≈12 Min.		Exit occultation. Reacquire communications links with DSN 64-meter ground station. Begin tracking to verify proper orbit insertion.		
E + 30 Min.		Verify Spacecraft Operating Modes.		Monitor engineering format TM for changes from pre-insertion mode.
		Despin HGA as before.		To obtain more accurate attitude determination.
E + 1 Hour		Performing Attitude and Spin Rate Determination (See Section 4.3.2).		
		Spinup HGA as before.		In preparation of despinning the entire S/C to 15 rpm.
E + 2 Hours		Perform despin to 15 rpm (See Section 4.3.1.3) (Requires ≈1.5 minutes).		
E + ≈12 Hours		FIRST APOAPSIS Perform reorientation to South Ecliptic Pole (See Section 4.3.1.1) (Requires ≈70 Minutes).		
E + ≈13 Hours		Despin HGA and point in elevation to earth, then transfer RF communication links to the HGA. (See Section 4.3.4).		

required to achieve the insertion attitude ranges from 20 to 35 degrees. The sun look angles range from 85 to 96 degrees so that no time limit due to power or thermal constraints exist. The earth look angles range from 78.5 to 64.5 degrees so that communications will be maintained using the forward omni antenna and the DSN 64-meter network.

Because of spacecraft occultation from earth view at closest approach, the OIM must be fired by a pre-stored command sequence from the spacecraft Stored Command Logic (SCL). Orbit insertion is the most critical maneuver of the entire mission. As such, it is desirable to be at the insertion attitude for a sufficient length of time prior to arrival to allow for proper attitude and spin rate determination, and the other necessary operations listed in Table 4.2.3-1. Precession to the insertion attitude 48 hours prior to closest approach (☒ -48 hours) allows ample time to perform all necessary operations with a margin of at least 40 hours of operation time.

The earth look angle at the orbit insertion attitude dictates the use of the forward omni antenna exclusively. No other antenna can provide coverage at that attitude. If the receiver connected to the forward omni should fail while the orbiter is in the insertion attitude, 36.4 hours will pass before communications with the spacecraft can be established. The 36.4 hours is the time period of the automatic receiver reverse function which is detailed further in Section 3.6.2.1. All necessary spacecraft command operations should be effected well in advance of 36.4 hours before closest approach. Beginning the orbit insertion maneuver sequence at ☒ - 48 hours as shown in Table 4.2.3-1 is compatible with protecting against a receiver failure in the insertion attitude.

The orbit insertion maneuver sequence begins at ☒ - 48 hours by switching transmit and receive functions from the HGA to the forward omni antenna. This transfer is made while

communicating on a DSN 64-meter net ground station and with the communications subsystem in the high power mode. The telemetry downlink is at 16 bits per second (worst case) in the forward omni/nigh power configuration. The spacecraft spin axis is then precessed to the insertion attitude. For launches prior to May 27, Mirac is the sole stellar reference at the insertion attitude. Capella is the only stellar reference for the second half of the launch opportunity. Both stars will have been visible during the cruise period. Attitude measurements are made on the available star(s) under cruise conditions; 15 rpm and HGA despun.

With the spacecraft attitude within at least 2 to 3 degrees of the target, despin control of the HGA is terminated and the HGA is allowed to spin up. The spacecraft is then spun up to 30 rpm to provide the proper gyroscopic stiffness during OIM burn. After verification of a proper spin speed change and a spin speed trim, if necessary, the antenna is again despun to perform a final attitude determination. (Attitude data taken with the HGA spinning is biased due to its static and dynamic imbalances).

After final attitude measurements and trim, if necessary, despin control is again terminated in preparation for OIM firing. During injection, the spacecraft is configured in the all-spun mode to minimize the effect of the transverse torque resulting from antenna static imbalance and OIM thrust along the +Z axis.

The spacecraft command memories will subsequently be loaded at E - 42 hours with the OIM firing sequence and verified as shown in Table 4.2.3-1. The relative timing and commands loaded into the two SCLs are diagrammed in Figure 4.2.3-1. The sequences are timed to be initiated with respect to the nearest 4096 second clock transition at the time of loading/verification. This eliminates the necessity of sending the memory "START" command at a precise time from the ground station. Pyrotechnic Control Unit (PCU) timing and use of time delays as shown in Figure 4.2.3-1

are to preclude certain failure modes in the SCL as discussed in Section 3.6.3.2.

From E - 40 hours on, spacecraft tracking data is assimilated so that the "near-Venus" effects on trajectory and approach velocity may more precisely be accounted for in calculating the exact time required for OIM burn. At approximately E - 6 hours the SCPs are reloaded and restarted (one at a time) with the most current OIM firing time.

During the time remaining until closest approach, the spacecraft functions and "housekeeping" data are monitored for anomalous behavior. OIM burn will occur at the closest approach to Venus, approximately 6 minutes after the beginning of earth occultation. The spacecraft will exit occultation approximately 12 minutes after OIM burn and communications links will be re-established by the DSN ground station.

Post-insertion attitude, spin rate and residual nutation should be monitored to verify no significant changes have occurred. Tracking data is obtained to define orbital period and periapsis altitude. As soon as spacecraft status is verified as being satisfactory, the HGA is spun up to the rotor rate of 30 rpm, then the entire spacecraft is despun to ≈15 rpm in preparation for the inversion maneuver near the first apoapsis to position the spin axis pointing toward the south ecliptic pole. (If the rotor were despun to 15 rpm while the HGA was despun, the HGA may move erratically and unpredictably in azimuth. Spinning up to the HGA to the 30 rpm rotor rate first requires only the single motor drive OFF command and a small transfer of momentum from the rotor to the HGA). Reorientation to the south ecliptic pole establishes the nominal on-orbit angle of attack and spacecraft viewing geometry for the initial periapsis pass. After the inversion, the HGA is despun and attitude data is taken to plan and execute an attitude trim if required. After the latter maneuver, an HGA elevation trim is made

and telemetry and command functions are switched to the HGA, as shown in Table 4.2.3-1.

Initial periapsis and second apoapsis maneuver strategies are discussed in Section 4.1.1.1. Once the orbit parameters (periapsis altitude and orbit period) are properly adjusted, the spacecraft is despun to 5 rpm and the on-orbit mission phase proceeds.

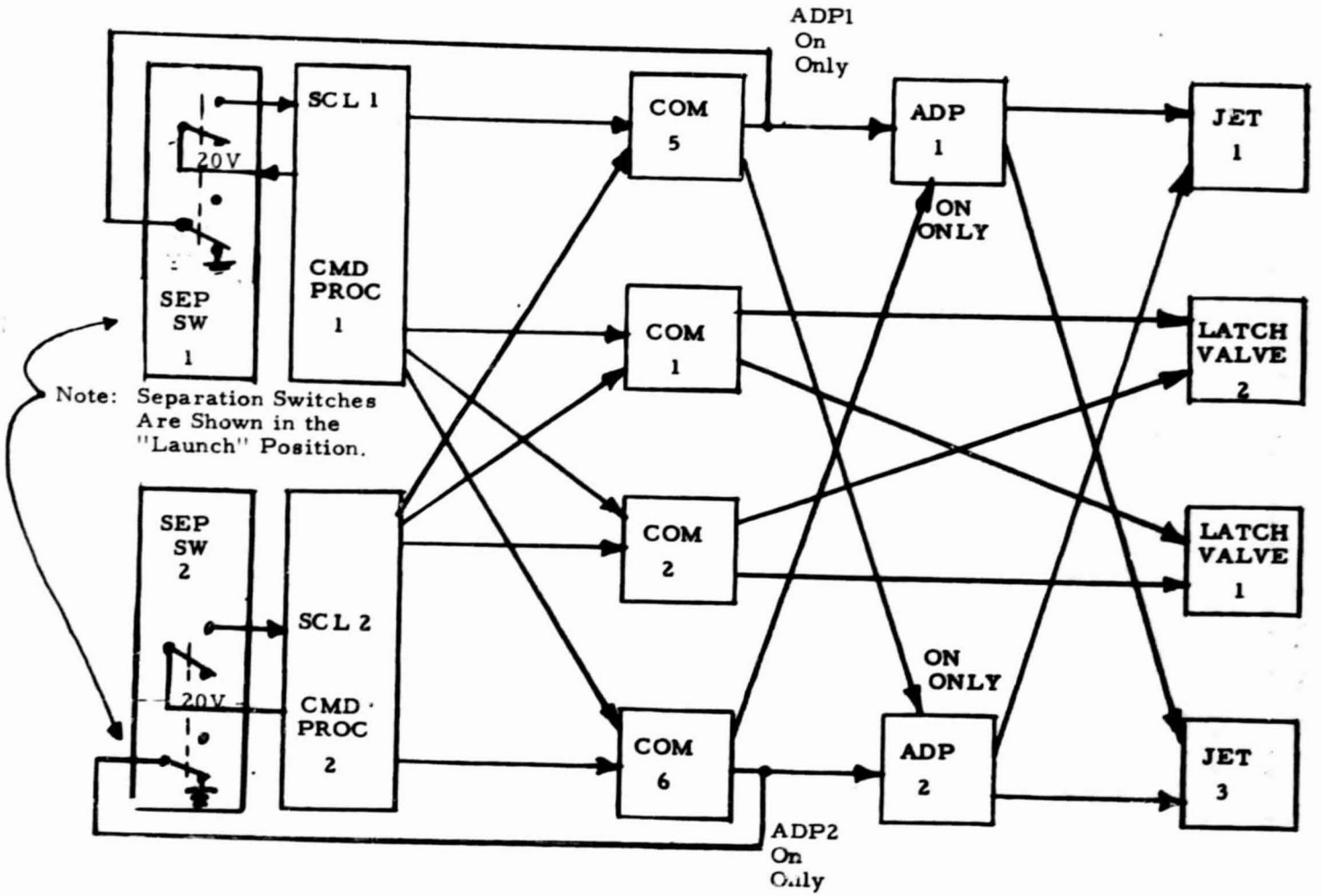
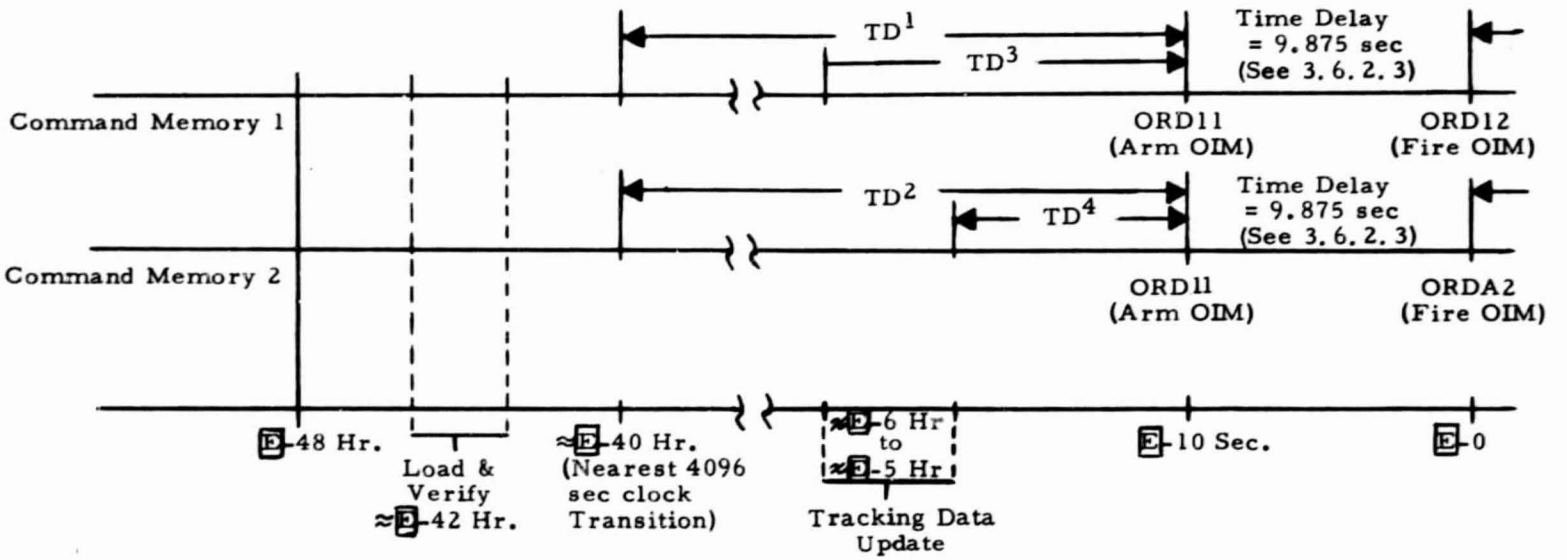


Figure 4.2.1.4-1. Spinup Sequence Functional Implementation



TD^1 = Time delay equivalent to time between E-0 and nearest 4096-second clock transition after E-42 Hours with Command Processor 1 clock stability bias incorporated.

TD^2 = Same as TD^1 except with command processor 2 clock stability bias incorporated.

TD^3 = Same as TD^1 except use nearest 4096-second clock transition after E-6 Hours.

TD^4 = Same as TD^2 except use nearest 4096-second clock transition after E-5 Hours.

Command processor accuracy (stability) determined for each command processor during interplanetary cruise; preferably in last 20 days of cruise. Each command processor's clock timing is calibrated against real time on ground. Calibration is effected by storing an exact 40.0 hour time delay code followed immediately by an RF modulation index change command, and measuring the time between start of the memory and the observed change in downlink modulation index.

Figure 4.2.3-1. Orbit Insertion Command Firing Sequence.

4.3 SPACECRAFT OPERATIONS

Descriptions of Primary and Backup Operational Modes and Operational Limitations, for Maneuvers, Roll Reference Transfer, Attitude Determination, HGA Pointing, Spacecraft Rover and Spacecraft Thermal Disciplines, are described below.

4.3.1 Maneuvers. A General Description; Common Maneuvers Strategy and Tactics; Strategy and Tactics unique to each of precession, ΔV, and spin rate maneuvers; and typical command sequences with corresponding telemetry verification for each of these three types of maneuvers are described ahead.

4.3.1.1 General Description. Spacecraft maneuvers include all commanded changes in spacecraft attitude (via precession), or velocity (via translation), or spin rate (change in angular momentum). Maneuvers are performed to establish and maintain attitude and spin rate during cruise, to correct errors in velocity (trajectory correction maneuvers), to adjust spin rate and attitude before and after orbit insertion motor burn, and to correct perturbations in velocity, spin rate and attitude during the in-orbit phase.

Execution of any one maneuver will result in a deficiency or excess in the magnitude of the desired maneuver. This is due primarily to impulse uncertainty, and is caused by variations in thruster(s) performance. Any one maneuver will result also in small changes as if each of the remaining two types of maneuvers had been executed as well. These cross-coupling errors, as they are called, are caused not only by imperfect mechanical alignment and unbalance of the thrusters, but also by imperfect timing in thruster(s) burn; and will vary as mass properties change due to depletion of liquid and solid (OIM) fuel, deployment of equipment, and reorientation of equipment (HGA). Another unwanted effect, nutation will also result; nutation is the inertial coning motion of the spin axis about the angular momentum vector.

Strategy and tactics common to the three types of maneuvers are described in Sections 4.3.1.2 and 4.3.1.3, respectively. The approach to maneuvers, and the significance of impulse uncertainty, coupling errors and nutation are described in these sections.

Each of the three types of maneuvers, including its unique strategy, tactics, characteristics, and typical command sequence, are detailed in Sections 4.3.1.4 through 4.3.1.6.

4.3.1.2 Common Strategy. Strategy is the basic plan, independent of the spacecraft design and the point in the mission. Strategy common to the three types of maneuvers consists essentially of (1) defining the pre-maneuver conditions of spacecraft mechanics and propulsion performance parameters, the desired post-maneuver conditions, and the best way to accomplish the post-maneuver conditions; (2) doing the maneuver, and (3) verifying the adequacy of the result.

Specific strategy begins with determining the pre-maneuver spacecraft attitude, velocity, position, spin rate, and nutation (if any is present). Determining velocity and position are relatively time-consuming for any spacecraft design. The degree of accuracy to which each of these five characteristics are to be determined varies according to the maneuver type and required magnitude. It is sometimes constrained by lack of available time for determination, which in turn, may be due to spacecraft thermal, power, or communications link performance constraints.

The pre-maneuver propulsion performance parameters are updated next, using pre-launch measured values and any in-flight calibration data that may be available. These are the impulse characteristics of each thruster, and the spacecraft mass properties. This information is used to define, via appropriate offline software, maneuver parameters, i.e., the burn duration (continuous or number of pulses), jet start angle (pulse maneuvers), the expected cross-coupling

and the expected nutation associated with any required post-maneuver conditions.

Maneuver strategy will emphasize: (1) Minimum fuel usage, (2) minimum disturbance to the communication links, power profile, and thermal profile, (3) maneuver accuracy and (4) simple and safe operation, including performance of the maneuver event in the shortest amount of time. The time element becomes increasingly significant as the one-way communications time between earth and spacecraft increases (one way communications time when the orbiter spacecraft begins orbiting Venus is approximately three minutes. The one way communications time gradually increases until, 230 days later, it becomes approximately fourteen minutes). All of the ideal goals are not attainable simultaneously. Their relative priority of importance depends in part on the point in the mission, the condition of the spacecraft, and the mission impact resulting from each ideal goal that is compromised or omitted.

The maneuver is then executed, and the result is assessed to determine if it is acceptably close to the required results. Post-maneuver attitude, velocity, position, spin rate, and nutation should all be determined for the first usage of each thruster to refine the models of mass properties and thruster impulse until they are sufficiently accurate. Subsequent post-maneuvers assessments need then be only partial. If the result is inadequate, then the cycle should be repeated using revised performance parameters.

4.3.1.3 Common Tactics. Tactics consist of the specific implementation of the strategy, unique to the spacecraft design and to the point in the mission.

Successful completion of a maneuver requires the use of all the Orbiter's subsystems. The controls subsystem supplies the attitude measurements and thruster control signals; the propulsion subsystem supplies the impulse required to move the spacecraft; the command and data handling subsystems process the commands

necessary to perform the maneuver and the telemetry conditioning necessary to verify the maneuver; the communications subsystem receives the commands, transmits the telemetry and indicates certain maneuver performance by doppler shift; and the power subsystem supplies the electrical power necessary to operate all of the required units.

Tactics for the Orbiter begin with determining the pre-maneuver spacecraft attitude, spin rate, and nutation (if any is present). Attitude and spin rate determinations are described in Sections 3.3.3.6, 3.3.3.7 and 4.3.2. The preferred method for determining attitude, via use of sun-to-star(s) time measurements, is functionally independent of the point in the mission, but the choice of star(s) varies during the mission as described in Sections 3.0 and 4.3.2. The preferred method for determining the spin rate is essentially independent of the point in the mission, as long as the sun is within the FOV of a sun sensor.

Nutation will have reduced to negligible levels if the previous maneuver had occurred many hours ago. It should have been measured immediately after the previous maneuver to aid in updating mass properties and thrust level for the forthcoming maneuver. (The Nutation angle (half-cone angle between the spacecraft spin axis and the angular momentum vector about which the spin axis is coning) should be $<1^{\circ}$ in magnitude before beginning any maneuver). Telemetered $\psi - \psi_2$ time delay (ATTM1Z and ATTM2Z) will vary approximately sinusoidally when nutation is present. The equivalent spin angle delay variation can be converted to the sun aspect angle variation by use of the appropriate equation shown in Figures 3.3.2.1-1 or 3.3.2.1-2. Resolution in sun L.O.S. angle measurement varies with spin rate and offset from the boresight axis of the sun sensor. For the nominal cruise attitude and spin rate of 15 rpm, the telemetered resolution of ±0.25 millisecond time delay translates into a sun L.O.S. (line-of-sight) resolution of $\pm 0.0230^{\circ}$. For the nominal on-orbit attitude and spin rate

of 5 rpm, the sun L.O.S. resolution will be one-third as small.

Determination of pre-maneuver velocity and position is done conventionally via the onboard phase-lockable S-band system and doppler measurements. The onboard phase-lockable X-band transmitter permits more accurate velocity measurements. It is intended for science use during occultation, but can be used for velocity measurements during the transit phase, provided thermal constraints are met (see Section 3.7.3.4).

Updating of mass properties and impulse for accurate prediction of the forthcoming maneuver is important due to the orbiter propulsion system design. The orbiter uses an unregulated mass-expulsion propulsion system. Steady depletion of fuel results in decreasing pressure and corresponding thrust level for the thrusters (A "blowdown" system - refer to Figure 3.4.2.1-3 through 3.4.2.1-5, and Figure 3.4.2.3-1). OIM burn results in a step decrease in spacecraft weight and the spacecraft moments of inertia.

The arrangement of the seven thrusters on the Orbiter is shown pictorially in Figures 3.4.1-3 and 3.4.1-4. Table 3.4.1-2 further specifies thrusters geometry. Three of the seven thrusters are aligned parallel to and offset from the spin axis. These axial thrusters, one called forward and the other two called aft, are used for precession and velocity change maneuvers. The four remaining thrusters are aligned offset from and approximately perpendicular to the spin axis. These four radial thrusters are divided into two pairs; the plane of the thrust vectors of each pair is canted relative to the plane perpendicular to the spin axis in order to pass through the center of mass during a particular phase of the mission. In particular, radial thrusters R3 and R4 are canted through the center of mass at the time of the first trajectory correction maneuver and radial thrusters R1 and R2 are canted through the on-orbit center of mass. Each thruster can be operated in either

the continuous or pulse mode. In the pulse mode, the pulse on time can be either 128 milliseconds or 512 milliseconds. Table 4.3.1.3-1 details the types of maneuvers that can be performed with various thruster combinations.

The essential significance of each mass property parameter and other pertinent parameters for each of the three types of maneuvers is shown in Table 4.3.1.3-2. (The relationships are detailed in later sections). Nominal mass parameters are listed in Table 3.1.2-3. Tolerances on these and thruster related parameters are used to predict the expected crosscoupling errors associated with each candidate set of thrusters for the forthcoming maneuver.

There are a large number of possible thruster and thruster operating mode combinations. Not all of the possible combinations are practical, and certain combinations are more advantageous than others for some maneuvers. There are four criteria used for selecting the thruster or thrusters and operating mode used in a particular maneuver. These are efficiency, granularity, maneuver duration and the disturbances caused by the maneuver. Efficiency is defined as the size of maneuver that can be performed per unit of propellant; it is affected by spacecraft spin rate, mass, moments of inertia, propulsion subsystem pressure, cant angle of the radial thrusters and pulse width in the pulse mode. Since several of these parameters change during the mission, the efficiency of a given maneuver changes throughout the mission. Granularity is defined as the minimum maneuver that can be performed by one pulse or one second of continuous burn. Granularity is affected by the same parameters as efficiency and thus, also changes throughout the mission. Maneuver duration is affected by the size of the maneuver, the granularity of the maneuver, the number of thrusters used, and the thruster operating mode used. The disturbances caused during a maneuver include uncertainties in the planned maneuver, nutation and cross-coupling of other maneuver types such as precession and spin speed changes

during velocity change maneuvers. Disturbances are caused by thrust misalignments, cant angles of the radial thrusters and uncertainties in spacecraft mass properties and thruster performance. Table 4.3.1.3-3 lists the thrusters and operating modes selected for the Orbiter mission and includes the efficiency and granularity of each maneuver. The rationale for these selections is discussed in the following paragraphs.

Table 4.3.1.3-1 lists three unique ways to perform an axial ΔV maneuver, i.e., the axial velocity component of a total (vector) ΔV maneuver. These are a single axial thruster (forward or aft) in the continuous mode, two aft axial thrusters in the continuous mode and a single axial thruster (forward or aft) pulsed twice per spin period. The basic method for performing an axial ΔV maneuver is to use a single axial thruster in the continuous mode since only a single thruster is available at the forward end of the spacecraft. Because the axial thrusters are offset from the spin axis, the use of a single axial thturster causes precession and nutation in addition to the desired axial ΔV. Coupled radial ΔV and spin speed changes are caused by misalignments and principal axis tilt. The use of two aft axial thrusters eliminates the moment due to a single thruster and thus reduces precession and nutation to functions of misalignments and uncertainties. Efficiency is the same whether one or two thrusters are used, but granularity is increased by a factor of two and maneuver duration is reduced by a factor of two by using the two aft axials. Assuming that the increased granularity can be tolerated, both aft axial thrusters should be used for aft axial ΔV maneuvers in order to minimize precession and nutation. The two aft axials are nominally fired together to execute all required orbit correction maneuvers. The axial ΔV granularity is nearly the same for the cruise and on-orbit phases because the reduction in thrust due to pressurant blowdown is nearly offset by the reduction in mass during orbit insertion. The efficiency of axial ΔV maneuvers increases on-orbit because the

reduction in spacecraft mass is greater than the reduction in specific impulse due to propellant blowdown.

The final method of performing an axial ΔV maneuver is by pulsing a single axial jet twice per spin period. This method is used on-orbit in the event of an aft axial thruster failure. Relative to the nominal two-jet mode, this 180° bang-bang mode increases maneuver duration by a factor of ≈15 and requirs 21% more propellant to execute the same maneuver.

A radial ΔV maneuver, i.e., the radial velocity component of a total ΔV maneuver, can only be performed in a pulse mode. The pair of radial thrusters canted closest to the center of mass at the time of the maneuver (R3 and R4 prior to orbit insertion, and R1 and R2 after orbit insertion) is used in order to minimize precession. Radial ΔV maneuvers tend to be lengthy since only one pulse pair can be fired per spin period. Thus, the 512 millisecond pulse width is used to minimize the duration of the maneuver. Radial ΔV maneuvers are less efficient than comparable axial ΔV maneuvers because of the lower thruster Isp in the pulse mode, the trigonometric loss due to the angle swept during the pulse on time and the trigonometric loss due to the cant angle of the radial thrusters. Because of the cant angle, there is an axial velocity change whenever a radial ΔV maneuver is performed. In addition, if the thrust vector plane does not pass through the spacecraft center of mass, precession will occur. Uncertainties in radial ΔV in the desired and transverse direction, axial ΔV, spin change, precession and nutation all occur due to misalignments and thruster mismatches and performance uncertainties.

TABLE 4.3.1.3-1

THRUSTER USAGE

MANEUVER	JET CONTROL MODES (PRIMARY & BACKUP)
Spin Control	
Spinup	1) R1 and R3 continuous 2) R1 continuous 3) R3 continuous
Spindown	1) R2 and R4 continuous 2) R2 continuous 3) R4 continuous
Canted Radial ΔV	1) R1 and R2 pulsed simultaneously at spin frequency 2) R3 and R4 pulsed simultaneously at spin frequency
Axial ΔV	1) A5 continuous 2) A6 continuous 3) A5 pulsed at twice spin frequency 4) A6 pulsed at twice spin frequency 5) A7 continuous 6) A6 and A7 continuous 7) A7 pulsed at twice spin frequency
Attitude Precession	1) (R1 and R2) or (R3 and R4) pulsed simultaneously at spin frequency 2) (R1 and R4) or (R2 and R3) pulsed simultaneously at spin frequency 3) A5 and A6 pulsed simultaneouly at spin frequency 4) A5 pulsed at spin frequency 5) A6 pulsed at spin frequency 6) A7 pulsed at spin frequency

TABLE 4.3.1.3-2

MANEUVERS PARAMETERS

Units	Parameters	Equation Symbol	Parameter Affects the Maneuver: ΔV Radial	ΔV Axial	Spin Change	Precession
Lb	S/C Weight	W	Yes →	→	→	→
Slug-Ft^2	Spin Moment of Inertia	I_3	Yes →	→	→	→
None	Spin to Transverse Inertia Ratio	σ	Yes →	→	→	→
None	No. of Radial Jets Fired	Q = NRJETS-1	No	No	Yes	No
In.	Radius to Radial Jets	R	Yes	No	Yes	No
In.	Axial location of Radial Jets relative to c.g.	L	Yes ↓	↓	↓	↓
Deg.	Tilt Angle of Jets 1R & 2R	η_1	↓	↓	↓	↓
Deg.	Tilt Angle of Jets 3R & 4R	η_2	↓	↓	↓	↓
Deg.	Location in transverse plane of radial jets	β	↓	↓	↓	↓
Lb.	Radial Jet Thrust	F	↓	↓	↓	↓
None	No. of Axial Jets Fired	K	No ↓	Yes ↓	No ↓	Yes
In.	Radius to Axial Jets	R	↓	↓	↓	Yes
In.	Axial location of axial jets fired relative to cg	L	↓	↓	↓	No
Lb.	Axial Jet Thrust	F	↓	↓	↓	Yes
Sec.	Specific Impulse	I_{SP}	Yes →	→	→	→
RPM	Initial Spin Speed	ω_0	→	→	→	→
Lb.	S/C Weight	δW	→	→	→	No
Deg.	Thrust Misalignment	ϵ	→	→	→	→
Deg.	Thruster Misalignment	$\delta\beta$		No	Yes	Yes
Deg.	Principal Axis Tilt	τ	→	→	→	→
Deg.	Jet Pulse Timing Circuit	$\delta\theta$		No	No	Yes
In.	Radius of Jets	δR		Yes	Yes	Yes
In.	Axial Location of Jets	δL		No	Yes	No
In.	Sun Pulse	$\delta\alpha$		No	No	Yes
Sec.	Thrust Centroid	δt_c		No	No	Yes
Deg.	Initial attitude (Rotation about inertial X Axis)	X		No	No	No
Deg.	Tilt Angle of Jets	$\delta\eta$	→	→	→	→
Lb.	Thrust Magnitude	δF	→	→	→	→
Slug-Ft^2	Spin Inertia	δI_3	No	No	Yes	Yes
Sec.	Pulse Burn Duration	T				
MPS	Desired Radial ΔV	ΔV_{Radial}				
Deg.	Initial attitude (rotation about inertial Y Axis)	Y				
Sec.	Uncertainty in Burn Duration	δT				
MPS	Desired Axial ΔV	ΔV_{Axial}				
RPM	Final Spin Speed	ω_f				
Sec.	Uncertainty in Burn Duration	δT				
RPM	Uncertainty in Spin Speed	$\delta\omega_f$				
Sec.	Burn Duration	T				
Deg.	Desired Precession	None				

Uncertainties

NOTE: These parameters are used in Figures 4.3.1.4.2.3-1, 4.3.1.5.2.2-1, 4.3.1.5.2.2-2, & 4.3.1.6.2.2-1.

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TABLE 4.3.1.3-2

MANEUVERS PARAMETERS

Units	Parameters	Equation Symbol	Parameter Affects the Maneuver: ΔV Radial	ΔV Axial	Spin Change	Precession
Lb	S/C Weight	W	Yes →	→	→	→
Slug-Ft^2	Spin Moment of Inertia	I_3	Yes →	→	→	→
None	Spin to Transverse Inertia Ratio	σ	Yes →	→	→	→
None	No. of Radial Jets Fired	Q = NRJETS-1	No	No	Yes	No
In.	Radius to Radial Jets	R	Yes	No	Yes	No
In.	Axial location of Radial Jets relative to $c.g._0$	l	Yes ↓	↓	↓	↓
Deg.	Tilt Angle of Jets 1R & 2R	γ_1	↓	↓	↓	↓
Deg.	Tilt Angle of Jets 3R & 4R	γ_2	↓	↓	↓	↓
Deg.	Location in transverse plane of radial jets	ϕ	↓	↓	↓	↓
Lb.	Radial Jet Thrust	F	↓	↓	↓	↓
None	No. of Axial Jets Fired	K	No	Yes	No	Yes
In.	Radius to Axial Jets	R	↓	↓	↓	Yes
In.	Axial location of axial jets fired relative to $c.g._0$	l	↓	↓	↓	No
Lb.	Axial Jet Thrust	F	↓	↓	↓	Yes
Sec.	Specific Impulse	I_{SP}	Yes →	→	→	→
RPM	Initial Spin Speed	ω_0	→	→	→	→
Lb.	S/C Weight	δW	→	→	→	No
Deg.	Thrust Misalignment	ϵ	→	→	→	→
Deg.	Thruster Misalignment	$\Delta\gamma$	→	No	Yes	Yes
Deg.	Principal Axis Tilt	ϵ	→	→	→	→
Deg.	Jet Pulse Timing Circuit	$\delta\phi$	→	No	No	Yes
In.	Radius of Jets	δR	→	Yes	Yes	Yes
In.	Axial Location of Jets	δl	→	No	Yes	No
In.	Sun Pulse	$\Delta\epsilon$	→	No	No	Yes
Sec.	Thrust Centroid	δt_c	→	No	No	Yes
Deg.	Initial attitude (Rotation about inertial X Axis)	X	→	No	No	No
Deg.	Tilt Angle of Jets	$\delta\gamma$	→	→	→	→
Lb.	Thrust Magnitude	δF	→	→	→	→
Slug-Ft^2	Spin Inertia	δI_3	No	No	Yes	Yes
Sec.	Pulse Burn Duration	τ				
MPS	Desired Radial ΔV	ΔV_{Radial}				
Deg.	Initial attitude (rotation about inertial Y Axis)	Y				
Sec.	Uncertainty in Burn Duration	$\delta\tau$				
MPS	Desired Axial ΔV	ΔV_{Axial}				
RPM	Final spin Speed	ω_f				
Sec.	Uncertainty in Burn Duration	$\delta\tau$				
RPM	Uncertainty in Spin Speed	$\Delta\omega_f$				
Sec.	Burn Duration	τ				
Deg.	Desired Precession	None				

NOTE: These parameters are used in Figures 4.3.1.4.2.3-1, 4.3.1.5.2.2-1, 4.3.1.5.2.2-2, & 4.3.1.6.2.2-1.

TABLE 4.3.1.3-3

SUMMARY OF NOMINAL MANEUVER CHARACTERISTICS

Maneuver	Jet Selection	Mode (Pulse Width, ms)	Nominal Performance During Cruise At 15 rpm		Nominal Performance On-Orbit At 5 rpm	
			Control	Efficiency	Control	Efficiency
Attitude Precession	1) Axial Jet Couple	Pulse (128)	0.15°/ Pulse	106°/lb.	0.30°/ Pulse	292°/lb.
	2) Single Axial Jet	Pulse (128)	0.08°/ Pulse	106°/lb.	0.15°/ Pulse	292°/lb.
	3) Radial Jet ΔV Pair with thrust offset from Center of Mass.	Pulse (128)	0.02°/ Pulse	12°/lb.	0.04°/ Pulse	34°/lb.
Spin Speed	1) Radial Jet Spinup or Despin Pair	Contin-uous	0.19 rpm/sec	17 rpm/lb.	0.13 rpm/sec	18.5 rpm/ lb.
	2) Single Radial Jet Spinup or Despin Thruster	a) Cont.	0.09 rpm/sec	17 rpm/ lb.	0.06 rpm/sec	18.5 rpm/lb.
		b) Pulse (128) at twice spin freq.	--	--	0.005 rpm/ pulse	14.4 rpm/lb.
Velocity	1) Aft Axial Jet Pair	Contin-uous	0.02 MPS/ sec	1.7 MPS/ lb	0.02 MPS/ sec	2.8 MPS/lb.
	2) Single Axial Jet	a) Cont.	0.01 MPS/ sec.	1.7 MPS/ lb.	0.01 MPS/ sec.	2.8 MPS/lb.
		b) Pulse (512) at twice spin freq.	--	--	0.004 MPS/ pulse	2.2 MPS/lb.
	3) Radial Jet ΔV Pair	Pulse (512)	0.01 MPS/ Pulse	1.6 MPS/ lb.	0.01 MPS/ Pulse	2.2 MPS/lb.

Spin speed changes can be accomplished by firing a diagonally opposite pair of radial thrusters in the continuous mode, firing a single radial thruster in the continuous mode or by firing a single radial thruster in the pulse mode. The normal method is to use the diagonally opposite pair. This method is efficient and minimizes attitude disturbances. The single radial thruster fired in the continuous mode is used in failure mode situations. This method improves granularity but increases attitude disturbances. The single thruster fired in the pulse mode can be used for on-orbit spin speed trims. It has the advantage of improved granularity and it minimizes the HGA pointing transients associated with phase lock-loop tracking of a variable spin speed, but it suffers an efficiency loss of 22%.

Spin speed granularity is better in the on-orbit configuration than the cruise condition because the reduction in thrust due to pressurant blowdown is greater than the decrease in spacecraft spin moment of inertia during orbit insertion. However, spin maneuvers are slightly more efficient on-orbit because the reduced spin moment of inertia more than offsets the degradation in specific impulse.

Attitude precession maneuvers are always performed in the pulse mode using one of three methods: The Forward and Aft axial thrusters as a couple, a single axial thruster or the radial thruster pair that is not aligned through the spacecraft center of mass at the time of the maneuver. The use of the axial couple is efficient and minimizes velocity disturbances. The single axial thruster precession maneuver has the same efficiency as the axial couple method, but there is an inherent axial velocity change. The granularity of the single thruster maneuver is improved by a factor of two, is consistent with nominal attitude maintenance requirements and generates a ΔV ($\approx$0.001 mps/pulse) that can be used to trim periapsis altitude. Attitude precession granularity is worse during the on-orbit phase because the decrease in spacecraft angular momentum is greater than the reduction in

thrust due to pressurant blowdown. Precession efficiency is improved in the on-orbit phase since the reduced momentum state more than offsets the degradation in specific impulse.

Precession maneuvers can also be performed using the radial thruster pair whose thrust vector plane is offset from the center of mass. This type of precession maneuver offers the smallest granularity but is only 11% as efficient as a precession maneuver performed with the axial thrusters. There is also a radial velocity change inherent in this maneuver. In addition to the cross-coupling errors mentioned above, each of these precession maneuvers causes attitude disturbances and velocity changes due to misalignments and thruster performance uncertainties.

A set of thrusters and its operating mode have been selected for each of nominal mission maneuvers. The expected performance for each mission maneuver is shown in Table 4.1.1.4-1.

The increasing one-way communications link time delay cited earlier is an inducement to use the onboard command memories for operational safety, wherever timing in execution of commands is critical. Spacecraft commanding should be done in real time wherever possible, but the onboard command memories should be used to execute backup commands that will return the spacecraft to a safe state. The command memories must be used for ΔV corrections required while the spacecraft is occulted, beginning with OIM burn. For these events, the onboard Data Storage Units should also be used to aid in post-maneuver performance analysis. Detailed descriptions of how to use the command memories and DSUs are provided in Section 3.6 and 3.5 respectively.

Detailed instructions and usage restrictions for structuring the ADP configure quantitative command for measurements and jets control during a maneuver event ((ATQ01 or ATQ0A) through (ATQ12 or ATQ0L), respectively), are contained in Sections 3.3.3.5 and 3.3.3.11, respectively.

The maneuver is then executed, and the result is assessed for accuracy. Any thruster maneuver also offers the opportunity for post-maneuver resolution between thruster tolerances and observed cross-coupling errors.

4.3.1.4 Precession Maneuver

4.3.1.4.1 Precession Maneuver Strategy. The initial and final (desired) spin axis orientations are each specified by two angles referenced to a convenient inertial frame. The spacecraft is precessed from the initial to the final orientation by applying an inertially referenced moment normal to the spin axis. This precession moment is supplied by pulsing axial (or radial) jets that are offset from the spacecraft center of mass; to obtain the proper inertial direction, these jets are fired at a fixed lag angle from an inertial reference (sun or star).

A convenient inertial frame for a reorientation maneuver starts with the spacecraft at the origin and with one axis directed towards the sun (primary inertial reference with stars as backup). The second axis is orthogonal to the sunline and in the plane defined by the sunline and north ecliptic pole. Note that the sunline and north elliptic pole are not typically orthogonal because the spacecraft flies outside of the ecliptic plane. The third axis completes the triad. The two angles that specify the orientation of the spin axis are shown in Figure 4.3.1.4.1-1: ϕ being the angle between the sunline and the spin axis, and θ being the dihedral angle between the plane defined by the sunline/north ecliptic pole and the sunline/spin-axis plane (right handed rule). Relative to the unit sphere shown in Figure 4.3.1.4.1-1 with the sun as a pole, the tip of the unit spin axis vector is at co-latitude ϕ and on meridian (or at longitude) θ.

An arbitrary infinitesimal precession of the spin axis is shown schematically in Figure 4.3.1.4.1-2; dP is the magnitude of the precession and the direction is given by the angle ψ. ψ is the

angle of rotation about the spin axis measured, as shown in Figure 4.3.1.4.1-2, from the sunline/spin axis plane to the direction of precession. ψ, called phase angle hereafter, differs by a $\pm 90^{\circ}$ from the jet-lag angle since jet torque is orthogonal to jet thrust. In practice, the jet-lag angle is measured from the sun pulse; and this angle, and accordingly, ψ, are held constant during an ideal precession maneuver. On the unit sphere (shown in Figure 4.3.1.4.1-1), the tip of the unit spin axis vector describes a rhumb line, i.e., crosses successive meridians at the same angle; in the Mercator projection, the rhumb line appears as a straight line.

The basic kinematical equations follow from Figure 4.3.1.4.1-2:

$$\Delta\theta = \theta_F - \theta_I = \tan\psi \ln\left[\frac{\tan\frac{\phi_F}{2}}{\tan\frac{\phi_I}{2}}\right] \qquad (1)$$

$$\Delta\phi = \phi_F - \phi_I = -P\cos\psi \qquad (2)$$

where the subscripts I and F refer to initial and final orientations, respectively. For given initial and final orientations, the phase angle, ψ, is obtained from Equation (1) and the total required precession angle, P, from Equation (2).

4.3.1.4.2 Precession Maneuver Tactics

4.3.1.4.2.1 Implementation. The actual precession maneuver is conducted by pulsing axial (or radial) jets at a fixed lag angle (from the sunline) as determined from ψ (Equation (1)). An uncertainty in this lag angle, equivalent to $\delta\psi$, will obviously affect the final orientation of the spacecraft. Each pulse precesses the spacecraft through a nominal small angle. P, obtained from Equation (2), determines the number of pulses and an uncertainty in the precession increment leads to an uncertainty in the actual P, denoted by δP,

which, in turn, leads to uncertainties in the final orientation of the spacecraft.

In addition to the two uncertainties -- $\delta\psi$ and δP -- there are also two uncertainties in the initial orientation of the spacecraft. The initial uncertainty in co-latitude, β, will be denoted by $\delta\beta_I$ and will be called an in-plane uncertainty -- in-plane referring to the sunline/spin-axis plane. This uncertainty is due to sun sensor measurement errors. The second initial attitude uncertainty will be defined as an uncertainty normal to this plane, will be denoted by ϵ_I, and will be called an out-of-plane uncertainty. This uncertainty is due to star sensor measurement errors.

Uncertainties in the final orientation of the spacecraft will also be defined as the in- and out-of-plane uncertainties -- denoted by $\delta\beta_F$ and ϵ_F, respectively; in-plane now refers to the final sunline/spin-axis plane. In particular,

$$\delta\beta_F = e_1\,\delta\beta_I + e_2\,\epsilon_I + e_3\,\delta\psi + e_4\,\delta P$$

$$\epsilon_F = f_1\,\delta\beta_I + f_2\,\epsilon_I + f_3\,\delta\psi + f_4\,\delta P$$

where the e's and f's are error coefficients (or partial derivatives) defined in Table 4.3.1.4.2-1. Note that final values of the output uncertainties in each plane are obtained by root-sum-squaring the corresponding contributions shown in the equations above. The total attitude error is given by

$$\sqrt{(\delta\beta_F)^2 + (\epsilon_F)^2}.$$

To obtain the values to achieve an open-loop precession along a rhumb line path, a number of steps must be taken:

(a) The values of longitude (θ_I, θ_F) and co-latitude (β_I, β_F), which describe the initial and final orientation, are used to obtain the rhumb line parameters, ψ and P. These describe the angle to the sun, and the length of

the ideal rhumb line. The logic for determining the correct phase angle ψ, and quadrant Q is defined in Figure 4.3.1.4.2-1.

(b) The propulsion performance characteristics (calibrated impulse and impulse centroid location) are then used to define the required number of pulses to achieve P and the appropriate jet start angle corresponding to ψ.

(c) An estimate of the total attitude error resulting from the maneuver is made from known tolerances to insure that the range of actual attitudes achieved is acceptable. In addition, various attitudes along the rhumb line path are examined for compatibility with any power, thermal and communications constraints.

(d) The required set of spacecraft commands is constructed, transmitted and verified prior to maneuver execution.

The process is best illustrated by a representative example, the nominal/mission scheduled recr to orbit insertion attitude. Figure 4.3.1.4.2-2 illustrates the relevant geometry as well as celestial latitude and longitude of the sunline, initial attitude and final attitude. ϕ_i, θ_f and $\Delta\theta$ are first defined as follows. From Figure 4.3.1.4.2-2, the initial (ϕ_i) and final (ϕ_f) sun angles are determined to be

$$\cos \phi_i = \cos(-0.6^\circ)\cos(90^\circ)\cos(\text{arbitrary}) + \sin(-0.6^\circ)\sin(90^\circ) \Rightarrow \phi_i = 90.6^\circ$$

$$\cos \phi_f = \cos(-0.6^\circ)\cos(65^\circ)\cos(348^\circ - 264^\circ) + \sin(-0.6^\circ)\sin(65^\circ) \Rightarrow \phi_f = 84.6^\circ$$

The great circle precession, ΔP, is given by:

$$\cos \Delta P = \cos(65^\circ)\cos(90^\circ)\cos(\text{arbitrary}) + \sin(65^\circ)\sin(90^\circ) \Rightarrow \Delta P = 25^\circ.$$

The net rotation of the spin axis/sunline plane, $\Delta\theta$, is obtained from the law of cosines, i.e.,

$$\cos(360^\circ - \Delta\theta) = \frac{\cos 25^\circ - \cos 90.6^\circ \cos 84.6^\circ}{\sin 90.6^\circ \sin 84.6^\circ} \Rightarrow$$

$$\Delta\theta = 335.7^\circ \quad \text{(long route)}$$
$$\Delta\theta = -24.3^\circ \quad \text{(short route)}$$

The rhumb line Equations ((1) and (2) of Section 4.3.1.4.1) can now be used to solve for the nominal rhumb line precession, P, and phase angle, Ψ, along the shortest route:

$$\tan\Psi = \frac{-\frac{24.3^\circ}{57.3^\circ}}{\ln\left[\frac{\tan 42.3^\circ}{\tan 45.3^\circ}\right]} \Rightarrow \Psi = 76.7^\circ$$

$$P = -\frac{(\beta_f - \beta_i)}{\cos\Psi} \Rightarrow P = 26.1^\circ$$

Using the nominal parameters under discussion, the in-plane and out-of-plane errors are defined from Table 4.3.1.4.2-1 to be

$$\delta\beta_f^2 \approx \delta\beta_i^2 + (0.2)\,\delta\psi^2 + (0.05)\,\delta P^2$$

$$\epsilon_i \approx \epsilon_i^2 + (0.95)\,\delta P^2$$

where initial and final errors are measured in degrees. Assuming initial in plane and out of plane errors of $\pm 0.5^\circ$, a phase angle error of $\pm 3^\circ$ and a 3% impulse error (i.e., $\delta P = 0.8^\circ$) results in a final in-plane error of 1.4° and final out of plane error of 0.9°. The total attitude error after the maneuver is then 1.7°; the maneuver adds $\pm 1^\circ$ to the initial total attitude uncertainty of 0.7°.

TABLE 4.3.1.4.2-1

ERROR COEFFICIENTS

UNCERTAINTY (OUTPUT) / ERROR SOURCE (INPUT)	$\delta\phi_F$	ϵ_F
$\delta\phi_I$	e_1 1	f_1 $\tan \Psi \left(1 - \frac{\sin \phi_F}{\sin \phi_I}\right)$
ϵ_I	e_2 0	f_2 $\frac{\sin \phi_F}{\sin \phi_I}$
$\delta\Psi$	e_3 $P \sin \Psi$	f_3 $\frac{1}{\cos \Psi}\left(\frac{\Delta\theta \sin \phi_F}{\sin \Psi} + P \sin^2 \Psi\right)$
δP	e_4 $-\cos \Psi$	f_4 $-\sin \Psi$

Impulse characteristics of the thrusters are used to define the required number of pulses (see Section 4.3.1.4.2.2). Impulse centroid characteristics and thruster/sum sensor geometry are employed to convert from the phase angle, ψ, to an appropriate jet start angle.

4.3.1.4.2.2 Precession Dynamics. Precession of a spinning spacecraft such as the Orbiter is accomplished by firing gas jets that generate an applied torque normal to the spin axis. Successive torque pulses will cause the angular momentum vector to precess and will induce a wobble motion of the spin axis about the angular momentum vector.

The angular momentum step size dP is given by the following equation (also shown on Figure 4.3.1.4.2.3-1):

$$dP = \frac{K \cdot F \cdot R \cdot T}{H_s} \left(\frac{\sin \theta/2}{\theta/2} \right)$$

where

K = Number of Axial Jets fired.

F = Axial jet thrust (lb.)

R = Radius to axial jets (feet)

T = Pulse burn duration (seconds)

H_s = Spinning angular momentum [(slug-ft^2) x (rad/sec)]

= $I_{ZT}\omega_{OT}$ = Total spinning Inertia times the total spacecraft spin rate for an all-spun configuration

OR:

$= I_{ZR}\omega_{OR}$ = Rotor spinning inertia times the rotor spin rate for the despun configuration.

θ = Burn angle (degrees) $= \omega_{OT}T$ or

$= \omega_{OR}T$.

$\left(\frac{\sin \theta/2}{\theta/2}\right)$ = efficiency of the thrusters.

The command resolutions for a precession, i.e., the angular step size in degrees per pulse (128 ms) are shown in Table 4.3.1.3-3, for several thruster combinations and the two basic phases in the mission.

4.3.1.4.2.3 Precession Maneuver Parameters and Algorithms. Maneuver parameters that are significant for a precession are shown in Table 4.3.1.3-2. Their first order effect and algorithms relationship are shown in Figure 4.3.1.4.2.3-1.

4.3.1.4.2.4 Restrictions. Restrictions unique to a precession maneuver include: 1) a time limitation, depending on sun L.O.S. and solar intensity, according to Figure 3.2.3-1, to avoid excessive thermal imbalance, 2) sun L.O.S. restriction to insure power balance. This also varies with solar intensity, batteries level of charge, and spacecraft load level (see Sections 3.8.3 and 4.3.5), 3) sun L.O.S. restriction on usage of any one of the three sun sensors as the SRR. If each of the initial and final spin axis angles w.r.t. the ecliptic normal is <30°, the mid-range sun sensor should be used. If the final spin axis angle is ≥30°, then the appropriate extended sun sensor should be selected prior to the start of the precession. 4) Earth l.o.s. restrictions on usage of any one of the four available antennas. The High Gain Antenna is usable only when despun. Its effective field-of-view, as well as that for each

of the other antennas, is described in Sections 2.3.2.1 and 2.3.2.2. Switchover to transmitter Hi power, then to the highest bit rate usable on the most effective omni, followed by switchover to that omni prior to the start of the precession is the most-likely required action for most precessions, and 5) the nutation angle should be <1° before beginning any maneuver.

4.3.1.4.3 Typical Command Sequence and Telemetry Verification. Initial and changed spacecraft configurations for a typical precession maneuver are shown in Table 4.3.1.4.3-1. The full description of the initial configuration is shown in the first column ("Assumed Steady State" column). Only the changes in configuration, from the preceding column, are shown in subsequent columns.

A detailed typical command sequence for a precession, with associated telemetry verification, for an in-transit precession from ecliptic-normal attitude, is shown in Table 4.3.1.4.3-2.

4.3.1.5 Trajectory Modification Maneuvers (ΔV)

4.3.1.5.1 ΔV Maneuver Strategy. A ΔV maneuver, accomplished in the cruise or on-orbit attitude, does not significantly disturb the prevailing communications links, thermal profile, and power profile. By resolving the required velocity change into velocity components that are parallel to an orthogonal set of spacecraft thrusters (vector mode), these disturbances are avoided. The operational complexity of pre- and post-maneuver precessions required for a ΔV maneuver with axial jets only (axial mode) are avoided by executing two successive, but operationally simpler, maneuvers. Figure 4.3.1.5.1-1 illustrates that the weight penalty incurred using the vector mode as a function of the ΔV required and the angle between the spin axis and ΔV direction. Use of the vector mode for the first midcourse ($\Delta V \leq 5$ m/sec) incurs a maximum weight penalty of 0.0 lbs.

TABLE 4.3:1.4.3-1

SPACECRAFT CONFIGURATIONS FOR A TYPICAL

PRECESSION MANEUVER

Time Relative to Maneuver / Equipment	Assumed Steady-State (Pre-Maneuver)	Just Prior to First Jet Fire *1	Between First Jet Fire & Last Jet Fire *1	Immediately After Last Jet Fire *1	Assumed Steady-State (Post-Maneuver)*1
Earth Station(s) In Use:	26 M.	64 M.			
Communications S/S: Mod Index (Radians)	1.18 (HGA; Lo XMTR pwr; 26 M. DSN).	0.65 (Fwd Omni; Hi pwr; 64 M DSN)			
Transmit Antenna	HGA.	Fwd Omni.			
Receive Antenna	Rcvr 1 to HGA; Rcvr 2 to aft omni.	Rcvr 1 to Fwd Omni.			
Exciter	1 ON; 2 OFF. Coh. Mode Enabled.				
Power Amplifier(s)	1 ON; 2, 3, & 4 OFF. (=Lo Power).	1 & 3 ON; 2 & 4 OFF. (=Hi Power)			
States of Transfer Switches	① Lo pwr to despun antennas. ② HGA Selected. ③ Fwd Omni selected. ④ Exciter 1 selected. ⑤ AMP1 to Lo. ⑥ Rcvr Norm Select. ⑦ AMP3 Selected.	① Hi pwr to despun antennas. ② Backup/Fwd Omni Selected ⑤ Amp 1 to Hi.			
X-Band Transmitter	OFF				
Controls S/S: Star Sensor Channels:	• ϕ^* & ϕ_2^* Both ON. • BP & Threshold Settings: as Required.				

*1 - NOTE: Only the changes from the preceeding column are shown.

Table 4.3.1.4.3-1 (Continued)

Equipment / Time Relative to Maneuver		Assumed Steady-State (Pre-Maneuver)	Just Prior to First Jet Fire	Between First Jet Fire & Last Jet Fire	Immediately After Last Jet Fire	Assumed Steady-State (Post-Maneuver)
Controls S/S (Cont'd) Sun Sensors		Mid-Range Selected	If final spin axis att. is $\geq 30^\circ$ off eclip. norm., then approp. extend. sun sensor should be select. prior to jet firing.			
Attitude Data Processor	ON/OFF State	ADP1 ON.				
	Measurement Select	A & B Channels in any configuration.	Configure for spin rate & sun aspect ang. measurements: A: ϕ to ϕ B: ϕ to ϕ_2			
	Jet Control	① All Jets Disabled. ② Spin rate detector enabled. ③ All other choices: DNA	Preferred: (① A5 & A6 selected; or A5 selected, or A6 selected, or A7 select; or Backup Mode choices: (R1 & R2 selected) or (R3 & R4 selected) or (R1 & R4 selected) or (R2 & R3 selected). ③ Pulse fire; pulse count; 128 ms pulse width; normal fire mode - all selected. *1			Same as Column 1.
	ADP Mode Select	① SRR adv. inhibited. ② Sun gate enabled. ③ Normal (Gate A = 11.25° wide) selected; SRR selected; sun selected; (Continued)				

*1 - NOTE: For Large precession maneuvers, the time count mode will be used.

Table 4.3.1.4.3-1 (Continued)

Equipment	Time Relative to Maneuver	Assumed Steady-State (Pre-Maneuver)	Just Prior to First Jet Fire	Between First Jet Fire & Last Jet Fire	Immediately After Last Jet Fire	Assumed Steady-State (Post-Maneuver)
Controls S/S Cont'd						
	ADP Mode Select (Cont'd)	④ Star Gates A & B; Channel 1 (ϕ*) selected. ⑤ PLL Loss of Lock enabled. ⑥ PLL spin range of 8.0 - 17.7 rpm selected.				
Attitude Data Processor (Cont'd)	ADP Configure Commands 3 through 11	① Jet countdown: All zeros. ② ACS angle: Delay magnitude as required, if Star Gate B is in use. ③ Roll Index delay magnitude as required. ④ PLL spin period magnitude - same as prevailing spin period. ⑤ JCE buffer output disabled.	① Jet countdown magnitude & ACS angle delay magnitude both loaded. ② JCE buffer enabled. ③ Fire interlock, then fire.	① Jet countdown magnitude decreasing by one per S/C revolution.	① Jet countdown magnitude at all binary ones (decimal 4095), & no longer changing. ② JCE buffer disabled at fixed time delay relative to 1st pulse firing. (This is backup turnoff of jets. Normal turnoff of jets occurs when JCE countdown register counts down to all zero & cycles once to all ones).	Same as Column 1.
	Spin Rate	15 RPM.				

Table 4.3.1.4.3-1 (Continued)

Equipment	Time Relative to Maneuver	Assumed Steady-State (Pre-Maneuver)	Just Prior to First Jet Fire	Between First Jet Fire & Last Jet Fire	Immediately After Last Jet Fire	Assumed Steady-State (Post-Maneuver)
Controls S/S Cont'd						
Despin Control Elect. (DCE)	ON/OFF State	1 ON; 2 OFF				
	Motor Drive	1 ON; 2 OFF				Either 1 ON or 2 ON in order to retain HGA in despun state. (Spinning HGA biases the attitude determination).
	DCE Configure Commands 0 Through 5	① AZ pointing slew: All Zero. ② HGA Elevation Positioner Slew: All Zeros. ③ Azimuth Position: as required. ④ HGA electrical boresight despun and pointed to Earth Station. ⑤ HGA elevation positioner OFF. ⑥ RAM VPD and NADIR (Not Used). VPD Reset	① Phase reset disable.			ENABLE
Command S/S: Command Processor		① 1 & 2 ON. ② Receiver normal selected. ③ SCL 1 & 2 cleared.				

Table 4.3.1.4.3-1 (Continued)

Time Relative to Maneuver / Equipment	Assumed Steady-State (Pre-Maneuver)	Just Prior to First Jet Fire	Between First Jet Fire & Last Jet Fire	Immediately After Last Jet Fire	Assumed Steady-State (Post-Maneuver)
Command S/S (Cont'd)					
Command Output Modules	1 Through 6 ON.				
Pyrotechnics	PCU Disarmed.				
Data Handling S/S:					
Nominal Bit Rate (bps)	2048 (via HGA)	To highest supportable bit rate via fwd omni & 64 M DSN.			
Data Format	Orbiter Eng.	ACS			Orbiter Eng.
PCM Encoder	1 ON; 2 OFF				
TM Processor	1 ON; 2 OFF. Subcarrier conv. enc. & data all ON.				
Data Input Modules	All 8 ON.				
Data Storage Units	TBD				
Power S/S:					
Batteries D.O.D.	0%				
Batteries Charge/ discharge State (typical)	(1) Lo chg. rate selected. (2) Prim. disch. reg. selected				

Table 4.3.1.4.3-1 (Continued)

Equipment / Time Relative to Maneuver	Assumed Steady-State (Pre-Maneuver)	Just Prior to First Jet Fire	Between First Jet Fire & Last Jet Fire	Immediately After Last Jet Fire	Assumed Steady-State (Post-Maneuver)
Power S/S (Cont'd)					
Protective Circuits	(1) All 7 limiters enabled. (2) UV/OL protection ON. (3) Precharge OFF.				
Science Bus	ON (To power OIR heater).				
Propulsion S/S:					
Latch Valves	1 & 2 closed.	1 &/or 2 Opened as required.			Both Closed.
Heaters	(1) Primary tanks, fwd & aft jet heaters all ON. (2) OIM heater 1 ON; 2 OFF.				

TABLE 4.3.1.4.3-2

TYPICAL PRECESSION SEQUENCE

Command Mnemonic	Command or Action	Remarks & Verification	
		Telemetry Mnemonic	Remarks & TM Data
(1)	Typical In-Transit Precession from Ecliptic Normal Attitude. • Determine up-to-date pre-maneuver S/C attitude, spin rate, & velocity for use in determining required precession magnitude & for comparison to post-maneuver values for performance evaluation.		
(2)	• Via an offline computer program, the up-to-date $I_{bit(s)}$ for the selected thruster(s), the ACS angle delay magnitude required for correct spin axis precession direction (including biases for sun sensor reference pulse & engines thrust vectors), & number of thruster(s) pulses (jet countdown magnitude) required to complete the precession shall be calculated. • An offline computer program may also be used to predict the expected cross-coupling errors in spin rate & ΔV resulting from the planned maneuver.		Input data and calculated data: Fuel usage to date (i.e. remaining fuel); tanks pressure; tanks temperatures; prelaunch test data for pulse firing of engines; updated S/C moments of inertia; updated engines moment arms; present S/C attitude in inertial space; total impulse required to change S/C attitude to desired orientation.
(3)	• Configure for spin rate & sun L.O.S. measurements throughout the impending maneuver:		
ATC#1: MEA1=SRR; MEA2=S2S (PSI2).	ADP1 configure - measurement select.	AM1ADS	Measurement A address: SRR to SRR.
		AM2ADS	Measurement B address: SRR to PSI2.
(4)	• Based on latest measured spin period, update PLL spin period magnitude (below):		Update is recommended if on-board loaded magnitude differs from measured true magnitude by more than ±1 bit (±0.25 m.s.).

Table 4.3.1.4.3-2 (Continued)

Command Mnemonic	Command or Action	Remarks & Verification	
		Telemetry Mnemonic	Remarks & TM Data
(4A) ATQ97 (or ATQ78, which is combination of ATQ97 & ATQ98 (see ahead).	Typical In-Transit Precession from Ecliptic Normal Attitude (Cont'd) ADP1 Configure - PLL Spin period magnitude (Eight MSBs)		MSBs values of AM1ADS (measured spin period) are commanded here for potential simulated SRR use during ensuing pulse firing.
		ASIMSZ (word 9)	Corresponds to that value commanded.
(4B) ATQ98 SIMSPNPDLSB	ADP1 Configure - PLL Spin period magnitude (Eight LSBs)		LSBs value of AM1ADS (measured spin period) here for potential simulated SRR use during ensuing pulse firing.
		ASIMSZ (word 10)	Corresponds to that value commanded
(5)	• Configure S/C for jet firing short of JCE buffer enable & opening of latch valve(s):		
(5A) ATQ92: RnE, RnD, AmE, AmD, as requfred (n=1 thru 4; m=5 thru 7); FMN; P/8; CP; FP;SRDE.	ADP Configure - Jet Control ([Preferred: (A5 & A6 selected) or (A5 selected) or (A6 selected) or (A7 Selected))] OR [Backup Mode Choices: (R1 & R2 Selected) or (R3 & R4 selected) or (R1 & R4 selected) or (R2 & R3 selected)] ; normal fire mode, 128 ms pulse width, pulse count, pulse fire, and spin rate detector enabled - all selected.	AJET1S Through AJET7S	= 1 for selected jets; = 0 for remaining jets.
		ASDETS	= 1 (spin rate detector enabled)
		AJETMS (word 2)	= 0 (Pulse fire selected)
		AMAGCS	= 1 (pulse count)
		APULLS	= 0 (128 ms. pulse width)
		AJETMS (word 3)	= 1 (normal fire select)

Table 4.3.1.4.3-2 (Continued)

Command Mnemonic	Command or Action	Remarks & Verification	
		Telemetry Mnemonic	Remarks & TM Data
(5B) ATQ#4 MAGSEC	Typical In-Transit Precession from Ecliptic Normal Attitude (Cont'd) ADP1 Configure - Jet countdown. (load number of pulses to be fired)	AJMAGC	JCE countdown - magnitude corresponds to that value commanded
(5C) ATQ#5 ACSANG	ADP1 Configure - ACS angle delay magnitude (azimuthal angle delay between occurance of sun reference pulse and initiation of one pulse firing per spin period).		Notes: (1) Each pulse firing of the total number of pulses loaded above will occur at this constant delay. A large precession may warrant two or more blocks of pulses, each block having a unique angle delay magnitude for all of its pulses. (2) There is a one count bias, i.e., the actual delay is one count greater than the commanded value.
(6)	• Verify ADP mode telemetry is as indicated at the right:	ADVANS	= 0 (SRR advance inhibited).
		ASUNGS	= 1 (sun gate enabled)
		A*ACQS	= 0 (normal)
		ASRRMS (Bit 6)	= 1 (SRR selected)
		ASRRMS (Bit 5)	= 1 (sun selected)
		ASUNSS	Sun Sensor: Midrange selected. (Assuming precession magnitude ≠60°)
		A*GASS	Star Gate A = 1 (channel 1 selected)
		A*GBSS	Star Gate B = 1 (channel 1 selected)
		ALOLES	PLL Loss of lock = 1 (enabled)
		ASPINS	PLL Spin range select: 8.0-17.7 RPM

Table 4.3.1.4.3-2 (Continued)

Command Mnemonic	Command or Action	Remarks & Verification	
		Telemetry Mnemonic	Remarks & TM Data
(7)	Typical In-Transit Precession from Ecliptic Normal Attitude (Cont'd) • Verify JCE buffer is disabled before proceeding:	AJCEFS	= 0 (JCE fire disabled).
(8)	• Configure downlink for forward OMNI & Hi Xmtr pwr usage:		
(9)	• Transfer command control to 64M DSN station. Tune up-link frequency to produce phase-locked downlink corresponding to rest frequency of VCO in receiver 1.		
(10)	• Decrease bit rate to highest supportable via fwd OMNI, HI XMTR power, & 64 M. DSN. Station usage (worst case of 16 BPS assumed ahead):		
(10A) TPCQ1 or TPCQA	Telemetry Processor control - bit rate select (to 16 BPS)		• Loss of bit sync. • Bit sync re-established
		Bit rate selection	Indicates 16 BPS.
(10B) MIL11 or MILA1	Low Mod. index select		Momentary disruption in data in worst case - likely demod sync, bit sync, & frame sync will be retained, as carrier suppression is decreased by 6.4 DB.
(10C) AMP1A or AMP2A	Power amplifiers 1 and 2 OFF. • Verify loss of downlink before proceeding.		Loss of downlink carrier.
(10D) ANT1A or ANTA2	Hi power to despun antennas select.		No downlink - not verifiable in real time.

Table 4.3.1.4.3-2 (Continued)

Command Mnemonic	Command or Action	Remarks & Verification	
		Telemetry Mnemonic	Remarks & TM Data
(10E) ANT22 or $\overline{\text{ANTB2}}$	Typical In-Transit Precession from Ecliptic Normal Attitude (Cont'd) Backup/forward OMNI antenna select		No downlink - not verifiable in real time.
(10F) AMP12 or $\overline{\text{AMPA2}}$	Amplifier 1 to Hi power select.		No downlink - not verifiable in real time.
(10G) AMP19 or $\overline{\text{AMPA9}}$	PA 1 ON/2 OFF		Carrier may be detectable at multiple of rest frequency of VCO in RCVR 1 (should be phase-locked).
(10H) AMP39 or $\overline{\text{AMPC9}}$	PA 3 ON/4 OFF		Sufficient downlink should be present for ground receiver lock on the carrier; demod sync with the subcarrier; & bit sync on the data. Frame sync will follow later because of the low bit rate
(11)	*Verify Comm S/S telemetry status before proceeding.	RANT1S	AFT OMNI/despun antennas to HI/LO pwr = 0 (despun to HI pwr).
		RAMP1S	AMP 1/2 to HI/LO pwr = 1 (AMP 1 to HI)
		RANT2S	HGA/Low gain antennas switch pos. = 0 (low gain antennas selected)
		RAMP1T	PA 1 temperature settles at "ON" level.
		RAMP1W	RF Power output 1 = 9.5 watts (nom.)
		RAMP3T	PA 3 temperature rises & settles at "ON" level
		RAMP3W	RF power output 3 = 9.5 watts (nom.)

Table 4.3.1.4.3-2 (Continued)

Command Mnemonic	Command or Action	Remarks & Verification	
		Telemetry Mnemonic	Remarks & TM Data
(12)	Typical In-Transit Precession from Ecliptic Normal Attitude (Cont'd) • Minimize coupled spin speed change effects on PLL frequency tracking:		• To avoid HGA pointing transients during jet firing period. (HGA is kept essentially despun to avoid biasing post-maneuver attitude determination.
DCEQ6	DCE configure - mode CMD (Phase reset disable).	ADSPNS	Despin Phase Reset = 0 (disabled) • This will keep the HGA pointed W.R.T. the last remembered location of the sun l.o.s., & the last remembered spin period. If the spin speed changes slightly due to coupling during the forthcoming maneuver, the HGA boresight will rotate about the spin axis at a rate equal to the difference between the remembered spin period and the present true spin period.
(13)	• To produce higher sampling of pertinent data during the maneuver, switch to the ACS format:		
TPCQ1 or $\overline{\text{TPCQA}}$	Telemetry processor control - format select (to ACS format)		• Loss of frame sync. • If 3 successive minor frames are required to re-establish frame sync, then 96 seconds (worst case) will elapse before it occurs, due to 16 BPS rate.
(14)	• According to the selected thruster(s), open the pertinent latch valves:		

Table 4.3.1.4.3-2 (Continued)

Command Mnemonic	Command or Action	Remarks & Verification	
		Telemetry Mnemonic	Remarks & TM Data
(14A) VALI9 or VALA9	Latch valve 1 open (if required - for use of any combination of R3, R4, A5 or A7).	VALV1S	Latch Valve 1 Open/Closed = 1 (open)
(14B) VAL29 or VALB9	Latch valve 2 open (if required - for use of any combination of R1, R2, or A6).	VALV2S	Latch Valve 2 Open/Closed = 1 (open)
(15)	• Verify proper latch valve status before proceeding.		
ATQ09	ADP 1 Configure - JCE Buffer Output Enable.	AJCE1S	JCE Buffer = 1 (Enabled)
(16)	• Verify that Buffer is enabled before proceeding.		
ATQ11	ADP1 Configure - Jet fire interlock		
(17)	Immediately after executing the following Jet fire command, monitor the indicated telemetry data. As ground backup to the onboard system, be ready to execute the JCE buffer output disable command (ahead), if telemetry indicates a significant deviation from expected values.		The S/C is protected against erratic performance by: 1. PLL loss of lock enabled (two successive out-of-lock occurances terminates the maneuver). 2. Spin rate detector enabled (if spin rate exceeds the range 3.7 to 67 rpm - the maneuver is terminated). 3. On-board automatic firing duty cycle detector (if firing duration per spin period exceeds 28% for certain, & exceeds 13% possibly, the maneuver is terminated). 4. Jet countdown circuit. (After the number of pulses loaded in the count-down circuit are fired, JCE buffer and JCE fire enable latch (latter enabled in next step) are both automatically disabled).

Table 4.3.1.4.3-2 (Continued)

Command Mnemonic	Command or Action	Remarks & Verification	
		Telemetry Mnemonic	Remarks & TM Data
	Typical In-Transit Precession from Ecliptic Normal Attitude (Cont'd)		
ATQ12	ADP1 configure - jet fire	AJCEFS	JCE fire enable = 1 (firing). Remains 1 only for firing duration, i.e. during pulse firing, likely it will telemeter as zero.
		VJET1T	Radial Jet 1 Temp
		VJET2T	Radial Jet 1 Temp
		VJET3T	Radial Jet 3 Temp
		VJET4T	Radial Jet 4 Temp
		VJET5T	Fwd Axial Jet 5 Temp.
		VJET6T	Aft Axial Jet 6 Temp.
		VJET7T	Aft Axial Jet 7 Temp.
			Increase for each thruster previously selected
		VTANKP	Tank Pressure - may show a detectable gradual change, depending on magnitude of precession.
		PLIMTI	Bus voltage limiter current
		PBUS LI	S/C Loads current
			If telemetry sampling of either of these occurs during a jet firing, a change will be seen (increase for PLIMTI; decrease for PBUSLI). (With use of ACS format & 16 BPS, PBUSLI is telemetered once every 16 sec.; PLIMTI, once every 2048 secs. With firing occuring for a 128 MSEC. Duration once every 4 sec., probability of a change being detected is very low).

Table 4.3.1.4.3-2 (Continued)

Command Mnemonic	Command or Action	Remarks & Verification	
		Telemetry Mnemonic	Remarks & TM Data
	Typical In-Transit Precession from Ecliptic Normal Attitude (Cont'd)	AM2ADS/ ATTM1Z	SRR to PSI2 - changes as expected (increase or decrease), and at expected rate.
		AM1ADS/ ATTM1Z	SRR to SRR - Changes as expected (spin rate) - based on estimated misalignments of thrusters.
		ALOCKS	PLL Loss of lock = 1 (no loss of lock): if PLL loss of lock has been enabled, a loss of the roll reference pulse for two successive S/C revolutions (and not before) will cause this to drop to zero and terminate the maneuver. If PLL Loss of Lock has been inhibited, this will stay at "1" at all times.
		AJMAGC	JCE countdown - decreases by one count per spin period.
(18)	If a problem occurs during firing, send the next command immediately. Otherwise, send it after a time delay from the previous jet fire command so as to be executed on board immediately after the last pulse firing has occurred.		
ATQ1S	ADP1 Configure - JCE buffer output disable.	AJCE1S	JCE buffer = 0 (disabled)
(19)	Verify completion of firing via the telemetry indication at the right.	AJMAGC	JCE countdown has counted down to all zeros, then counter has recycled to all binary ones (decimal 4095) and stopped changing.
		AJCEFS	JCE fire enables = 0 (not firing)

Table 4.3.1.4.3-2 (Continued)

Command Mnemonic	Command or Action	Remarks & Verification	
		Telemetry Mnemonic	Remarks & TM Data
	Typical In-Transit Precession from Ecliptic Normal Attitude (Cont'd)	AM2ADS/ ATTM1Z	SRR to PS12 - no change since termination of firing.
		AM1ADS/ ATTM1Z	SRR to SRR- no change since termination of firing.
		AJCE1S	JCE buffer = 0 (disabled)
(20)	• Begin trajectory determination to establish ΔV caused by precession.		
(21)	• Reconfigure the S/C for post-maneuver steady-state operations (assumes no subsequent imminent maneuver operations);		
(21A) VAL1S or VALA9	Latch Valve 1 closed (if open)	VALV1S	Latch Valve 1 open/closed = 0(closed)
(21B) VAL2S or VALB9	Latch Valve 2 closed (if open)	VALV2S	Latch Valve 2 open/closed = 0 (closed)
(21C) ATCS RaD and AmD (n = 1 thru 4; m = 5 thru 7); FMN; P18; CP; FP; SRDE.	ADP1 configure - jet control (all jets disabled)	AJET1S Through AJET8S	= 0 for all jets.
		ASDETS	= 1 (spin rate detector enabled)
		AJETMS	Continuous/pulse fire select = 0 (Pulse)
		AMAGCS	Pulse/time count select = 0 (Time)
		APULLS	Pulse width select = 0 (128 ms)
			(Continued)

Table 4.3.1.4.3-2 (Continued)

Command Mnemonic	Command or Action	Remarks & Verification	
		Telemetry Mnemonic	Remarks & TM Data
	Typical In-Transit Precession from Ecliptic Normal Attitude (Cont'd)		(Continued)
		AJETMS (word 3)	Normal/Alternate Fire Mode = 1 (Normal)
(21D) ATQ74	ADP1 Configure - Jet countdown. (= All zeros)	AJMAGC	JCE Countdown - Magnitude of all zeros.
(21E) ATQ97 (or ATQ78, which is combination of ATQ97 & ATQ98 (see ahead)	ADP1 Configure - PLL spin period magnitude (MSBs)		MSBs values of AM1ADS (measured spin period) are commanded here for potential simulated SRR use.
		ASIMSZ (word 9)	Corresponds to that value commanded.
(21F) ATQ98	ADP1 Configure - PLL spin period magnitude - (LSBs)		LSBs value of AM1ADS (measured spin period) are commanded here for potential simulated SRR use.
		ASIMSZ (word 10)	Corresponds to that value commanded
(22)	• Stay in the ACS format, & with HGA despin (a spinning HGA will bias attitude measurements) while performing an attitude determination (refer to section 4.3.2), then return to orbiter engineering format:		
TPCQ1 or $\overline{\text{TPCQA}}$	Telemetry processor control - format select (to orbiter eng. format)		• Loss of frame sync. • If 3 successive minor frames are required to re-establish frame sync, then 96 secs. (worst case) will elapse before it occurs, due to 16 BPS rate.
(23)	• Resume steady-state operations as defined in Table 4.3.1.4.3-1 - monitor batteries states-of-charges & comm. S/S temperatures in face of HI XMTR Pwr operation.		

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4.3.1.5.2 ΔV Maneuver Tactics

4.3.1.5.2.1 Implementation of a ΔV Maneuver. The axial and radial thrusters onboard the orbiter constitute the orthogonal set required to perform any ΔV without changing the prevailing attitude. The geometric arrangement of axial and radial thrusters relative to the spin axis, and the design of JCE control in the attitude Control Subsystem, permit the spacecraft to be translated successively in axial and radial directions with operational simplicity.

The thrust vectors of radial thrusters R3 and R4 are canted -4.46° toward the -Z axis from a plane parallel to the spacecraft X-Y plane, in order to pass through the center of mass that is estimated to apply at the required time for TCM #1. This set should be used for radial ΔV maneuvers prior to OIM burn. As the center of mass shifts toward the +Z axis due to expenditure of hydrazine fuel, subsequent usage of thrusters R3 and R4 will cause a coupled positive precession about a precession axis at $\theta = 322°30'$ (+X axis is at $\theta = 0°$).

The thrust vectors of radial thrusters R1 and R2 are canted +4.02° toward the +Z axis from a plane parallel to the spacecraft X-Y plane in order to pass through the center of mass that is estimated to apply on-orbit as an average. This set should be used for radial ΔV maneuvers after OIM burn. As the center of mass shifts toward the +Z axis due to expenditure of hydrazine fuel, subsequent usage of thrusters R1 and R2 will cause a coupled negative precession about the precession axis at $\theta = 322°30'$. The thrust vectors of axial thrusters A5, A6 and A7 are parallel to the +Z axis.

Using the axial thruster(s) in continuous burn mode for an axial ΔV will result in a small precession if the angle burn duration is not an integer multiple of the prevailing spin period.

The command resolution for a ΔV maneuver, given the sensitivities shown in Table 4.3.1.3-3, and

0.512 second for the LSB in the ADP configure quantitative command for continuous firing mode (ATQØ4 or ATQØD), is 0.0102 MPS/0.512 sec. for the aft axial jet pair, and 0.0051 MPS/0.512 sec. for a single axial jet.

Pulsed mode resolution is 0.004 MPS/pulse for single axial jet pulsed (512 ms) twice per spin period, and 0.01 MPS/pulse for a radial jet pair at 512 ms pulse duration.

4.3.1.5.2.2 Maneuver Parameters and Algorithms. Parameters that are significant for a ΔV maneuver are shown in Table 4.3.1.3-2. Their first order effect and algorithmic relationship are shown in Figures 4.3.1.5.2.2-1 and 4.3.1.5.2.2-2 for axial and radial ΔV maneuvers, respectively. Figure 4.3.1.5.2.2-3A shows that impulse magnitude error (essentially defined by thruster impulse uncertainty) is 2 percent and <3 percent in the axial and vector modes, respectively. Figure 4.3.1.5.2.2-3B shows that the impulse direction errors (function of attitude uncertainty, open loop precession errors and execution errors) are <3 deg and <1.7 deg in the axial and vector modes, respectively.

4.3.1.5.2.3 Restrictions. There are no restrictions unique to a ΔV maneuver beyond those general restrictions already cited in Section 4.3.1.3. Neither firing the radial engine(s) with 512 ms pulse width per revolution or firing the axial(s) continuously to accomplish a ΔV are expected to cause intolerable disturbances to the DCK control of the HGA.

4.3.1.5.3 Typical Command Sequence and Telemetry Verification. Initial and changed spacecraft configurations for a typical radial ΔV maneuver (no precessions required), axial ΔV maneuver (no precessions required), total ΔV maneuver with axial jets only (precessions required), and total ΔV maneuver with radial jets only (precessions required) are shown in Tables 4.3.1.5.3-1, -3, -5 and -7, respectively.

Revision

A detailed typical command sequence for each of these three types of ΔV maneuvers, with associated telemetry verification, starting with the spacecraft intransit and in ecliptic-normal attitude, is shown in Tables 4.3.1.5.3-2, -4, -6 and -8, respectively.

TABLE 4.3.1.5.3-1

SPACECRAFT CONFIGURATIONS FOR A TYPICAL RADIAL ΔV MANEUVER

This is the same as Table 4.3.1.4.3-1 (Typical Precession Maneuver) with the following exceptions:

1. Pre-maneuver steady-state configurations for comm. S/S & bit rate prevail throughout the radial ΔV maneuver (HGA can be maintained pointed to earth station by DCE even in the face of 512 MS pulsing of radial jets for a radial ΔV).

2. For "Attitude Data Processor - Jet Control - Just Prior to First Jet Fire", replace jet select combinations by:

 (R1 & R2 selected) or (R3 & R4 selected). - whichever produces a thrust vector directed closer to the prevailing S/C Center of Mass (i.e. that produces the smaller magnitude precession because of change in location of center-of-mass since launch).

3. For "DCE - Just Prior to First Jet Fire - delete "Phase Reset Disable"

TABLE 4.3.1.5.3-2

TYPICAL RADIAL ΔV MANEUVER SEQUENCE

This is the same as Table 4.3.1.4.3-2 (Typical Precession maneuver Sequence) with the following exceptions:

1. Replace all references to "precession" by "radial ΔV maneuver."

2. Replace jet select combinations for ATQ#2 command (ADP1 configure - Jet Control) by "(R1 & R2 selected) or (R3 & R4 selected." and selection of 128 ms pulse width by selection of 512 ms pulse width (APULLS telemetry should read logical 1 (= 512 ms pulse width)).

3. Delete all changes to comm. S/S & Data Handling S/S - operation on HGA, 2048 (or highest supportable) bps rate, Hi mod index select, & Xmtr, low power.

 Delete DCE configure - mode control - disabling of HGA Phase Reset.

 Retain scheduled switchover to 8 from ACS Data Format, however.

TABLE 4.3.1.5.3-3

SPACECRAFT CONFIGURATIONS FOR A TYPICAL AXIAL ΔV MANEUVER

This is the same as Table 4.3.1.4.3-1 (Typical Precession Maneuver) with the following exceptions:

1. Pre-maneuver steady-state configurations for comm. S/S & Bit Rate Prevail throughout the axial ΔV maneuver (HGA can be maintained essentially pointed to earth station by DCE even in the face of 512 ms pulsing or continuous firing of axial jet(s) for an axial ΔV; however, DCE Phase Reset should be disabled to avoid HGA pointing transients during the Jets Firing).

2. For "Attitude Data Processor - Jet Control - Just prior to first jet fire":

 (a) Replace jet select combinations by: (A5 or A6 or A7 or (A6 & A7) selected for continuous firing) or (A5 or A6 or A7 selected for pulsing at once per half spin periods) or ((A6 & A7 selected for pulsing at once per spin period).

 (b) For continuous firing: Continuous fire select & Time Count Select.

 (c) For Pulse firing: Pulse Fire, Pulse Count, Pulse width likely 512 ms., and either normal (one firing per spin period) or alternate mode (one firing per half spin period) - all selected.

3. For "ADP Configure - Commands ATQ04 through ATQ12 - for continuous fire:

 (a) Replace all references to "Jet Countdown" by "Time Countdown".

4. For "DCE - immediately after last jet fire" - Add: "Phase reset enable."

TABLE 4.3.1.5.3-4

TYPICAL AXIAL ΔV MANEUVER SEQUENCE

This is the same as Table 4.3.1.4.3-2 (Typical precession maneuver sequence) with the following exceptions:

1. Replace all references to "precession" by "axial ΔV maneuver."

2. Replace jet select combinations for ATQ∅2 command (ADP1 Configure - Jet Control) by "(A5 or A6 or A7 or (A6 7 A8) selected. For continuous fire, select continuous fire & time count. For pulsed fire - as shown, but select 512 ms. pulse width, and normal or alternate mode. Listed telemetry should indicate corresponding to that selected by command.

3. For a continuous firing: (a) ATQ∅4 command (ADP1 Configure - jet countdown) must represent the time count in 0.512 second steps; (b) ATQ∅5 command (ACS Angle Delay Magnitude) is omitted; (c) the on-board duty cycle detector is inhibited during continuous firing & the reference to "pulse loaded into the countdown circuit" is to be replaced by "... 0.512 msec, time counts loaded into the countdown circuit"; (d) PLIMTI & PBUSLI will both definitely show a change in current level during a continuous firing; (e) AJMAGC (JCE Countdown) will be decreasing by one count per 0.512 msec.

4. Delete all changes to comm S/S & Data Handling S/S - retain operation on HGA, 2048 (or highest supportable) bit rate, Hi Mod Index select, & Xmtr low-power. Retain scheduled switchover to & from ACS format, however.

5. After the post-maneuver issuance of ATQ∅7 & ATQ∅8- ADP 1 Configure - PLL spin period magnitude - add (to restore full DCE control): DCEQ6 - DCE configure - mode cmd (phase reset enable). ADSPNS should then = 1 (enabled).

TABLE 4.3.1.5.3-5

SPACECRAFT CONFIGURATIONS FOR A TOTAL ΔV MANEUVER WITH AXIAL JETS ONLY - (AXIAL MODE)

This will consist of a precession to the ΔV attitude followed by axial jet(s) burn continuously, followed by precession back to the steady state attitude. S/C Configuration changes shall conform to:

1. Precession to ΔV attitude - same as Table 4.3.1.4.3-1, except do not close latch valves needed for steps 2 & 3 ahead.

2. Continuous axial jet(s) burn - same as Table 4.3.1.5.3-3, except

 (a) Comm. S/S & data handling S/S shall be same status during the burn as for post precession, and
 (b) Do not close latch valves needed for step 3 ahead.

3. Precession back to steady - state attitude: Same as Table 4.3.1.4.3-1, except Comm. S/S & bit rate switchover is in reverse order (from (FWD OMNI, HI XMTR PWR, 16 bps) to (HGA, LOW XMTR PWR, 2048 bps)).

TABLE 4.3.1.5.3-6

SEQUENCE FOR A TOTAL ΔV MANEUVER WITH AXIAL JETS ONLY - (AXIAL MODE)

1. Precession to ΔV attitude - same as Table 4.3.1.4.3-1 (typical precession maneuver sequence)

2. Continuous axial jet(s) burn - same as for Table 4.3.1.5.3-4 - except Comm. S/C & Data Handling S/S shall be the same status during the burn as for post-precession (already on FWD OMNI, XMTR HI PWR, 64 M. DSN, & 16 BPS).

3. Precession back to steady state attitude: Same as Table 4.3.1.4.3-1, except Comm. S/S & bit rate switch-over is in reverse order - complementary commands needed for this that are not shown in the Table.)

TABLE 4.3.1.5.3-7

SPACECRAFT CONFIGURATIONS FOR A TOTAL ΔV MANEUVER WITH RADIAL JETS ONLY - (RADIAL MODE)

This will consist of a precession to the ΔV attitude followed by radial jet(s) burn inpulsed mode, followed by precession back to the steady state attitude. S/C Configuration changes shall conform to:

1. Precession to ΔV attitude - same as Table 4.3.1.4.3-1, except do not close latch valves needed for steps 2 & 3 ahead.

2. Pulsed Radial jet(s) burn - same as Table 4.3.1.5.3-1, except

 (a) Comm. S/S & data handling S/S shall be same status during the burn as for post precession, and
 (b) Do not close latch valves needed for step 3 ahead.

3. Precession back to steady - state attitude: Same as Table 4.3.1.4.3-1, except Comm. S/S & bit rate switchover is in reverse order (from (FWD OMNI, HI XMTR PWR, 16 bps) to (HGA, LOW XMTR PWR, 2048 bps.).

TABLE 4.3.1.5.3-8

SEQUENCE FOR A TOTAL ΔV MANEUVER WITH RADIAL JETS ONLY - (RADIAL MODE)

1. Precession to ΔV attitude - same as Table 4.3.1.4.3-2 (typical precession maneuver sequence)

2. Pulsed Radial jet(s) burn - same as for Table 4.3.1.5.3-2 - except Comm. S/C & Data Handling S/S shall be the same status during the burn as for post-precession (already on FWD OMNI, XMTR HI PWR, 64 M. DSN, & 16 BPS).

3. Precession back to steady state attitude: Same as Table 4.3.1.4.3-1, except Comm. S/S & bit rate switchover is in reverse order - complementary commands needed for this that are not shown in the Table.)

4.3.1.6 Spin Rate Change and Trim Maneuvers

4.3.1.6.1 Spin Rate Change Maneuvers. Spin change maneuvers are most expediently done by firing radial thrusters continuously. However, the 180° bang-bang is preferred during HGA operation in order to limit spin speed rate of change and insure continuous pointing. Firing a pair of spinup or despin thrusters rather than a single thruster is always preferred in either continuous or pulse mode, in order to minimize attitude and velocity perturbations.

4.3.1.6.2 Unique Tactics for a Spin Rate Change

4.3.1.6.2.1 Implementation of a Spin Rate Change. The cant angles of the spin-up thrusters pair (R1 and R3) and spindown thrusters pair (R2 and R4) on the Orbiter are slightly different (+4.02° and -4.46°, respectively) so as to result in a small precession about the precession axis at θ 322°30', and a small ΔV toward the -Z axis. Using any single thruster for spin rate change will result in a larger precession and an accompanying larger axial ΔV that that obtained by use of the appropriate pair.

The command resolution for a spin rate change, given the sensitivities shown in Table 4.3.1.3-3, and 0.512 second for the LSB in the ADP Configure quantitative command for continuous firing mode (ATQØ4 or ATQØD), ranges from 0.098 RPM/0.512 sec. to 0.067 RPM/0.512 sec. for a radial jet pair, and from 0.046 RPM/0.512 sec. to 0.0307 RPM/0.512 sec. for a single radial jet. Pulsed firing from a single jet on-orbit has a resolution of 0.05 RPM/pulse.

4.3.1.6.2.2 Spin Rate Change Parameters and Algorithms. Parameters that are significant for a spin rate change are shown in Table 4.3.1.3-2. Their first order effect and algorithmic relationship are shown in Figure 4.3.1.6.2.2-1.

4.3.1.6.2.3 Restrictions. Beyond the general restrictions cited in Section 4.3.1.3, restrictions unique

to a spin rate change include those described in Section 3.3.2.3.5 and 3.3.3.5 that are summarily reported here for emphasis: (1) the PLL loss of lock detection should be disabled except for spin trims within the PLL selected spin range since the PLL cannot track large spin speed changes, and will terminate the spin change maneuver if loss of lock is detected (2) the sun gate should be disabled, except for spin trims within the PLL selected spin range since the spin period is changing, and an enabled sun gate would transfer the SRR to the simulated SRR after two consecutive missed sun pulses. A disabled sun gate also allows normal sun acquisition. (4) In order to maintain the HGA boresight axis approximately pointed to a DSN station during a spin speed change, the maximum firing duty cycle permissible is that obtained via use of two 128 ms pulses per revolution.

4.3.1.6.3 Typical Command Sequence and Telemetry Verification. Initial and changed spacecraft configuration for a typical 128 ms pulsed spin change maneuver (likely a spin trim) within the PLL selected spin range are shown in Table 4.3.1.6.3-1. A corresponding detailed sequence of commands and telemetry verification is shown in Table 4.3.1.6.3-2.

Initial and changed spacecraft configurations for a typical continuous or 512 ms pulsed (or 128 ms pulsed to produce final spin rate outside of the initial PLL selected spin range) spin change maneuver are shown in Table 4.3.1.6.3-3. A corresponding detailed sequence of commands and telemetry verification is shown in Table 4.3.1.6.3-4.

TABLE 4.3.1.6.3-1

SPACECRAFT CONFIGURATIONS FOR A TYPICAL 128 MS. PULSED SPIN CHANGE MANEUVER (LIKELY A SPIN TRIM) WITHIN THE SELECTED PLL SPIN RANGE

This is the same as Table 4.3.1.4.3-1 (Typical Precession Maneuver) with the following exceptions:

1. Pre-maneuver steady state configurations for comm. S/S and bit rate prevail throughout the spin trim maneuver (HGA can be maintained pointed to earth station by DCE even in the face of 128 MS pulsing once or twice per spin period of radial jets for a spin trim - refer to Section 3.3.3. However, DCE configure - mode control - phase reset should be disabled to avoid HGA pointing transients during the jets firings.

2. For "attitude data processor - jet control - just prior to first jet fire" replace jet select combinations by: R1 OR R2 OR R3 OR R4 OR (R1 & R3) OR (R2 & R4) selected.

3. If the expected final spin rate is outside the pre-maneuver steady state selected spin range for "ADP Mode Select", then the maneuver described in Tables 4.3.1.6.3-3 and 4.3.1.6.3-4 should be followed.

4. For "ADP Configure Commands ATQ04 through ATQ12 - just prior to first jet fire" - add: PLL Spin period magnitude - set to expected final spin rate.

5. For "Attitude Data Processor - Spin Rate" -

 (a) "Between first jet fire and last jet fire" - add: "changing toward expected final spin rate."
 (b) "Immediately after last jet fire" - add: "at expected final spin rate."

6. For "DCE - immediately after last jet fire" - add: Phase reset enable.

TABLE 4.3.1.6.3-2

TYPICAL SPIN TRIM (128 MS PULSED - WITHIN THE PLL SELECTED SPIN RANGE) MANEUVER SEQUENCE

This is the same as Table 4.3.1.4.3-2 (Typical Precession Maneuver Sequence) with the following exceptions:

1. Replace all references to "precession" by "spin trim maneuver."

2. ATQ#7 & ATQ#8 (or, equivalently, ATQ#8) commands (ADP1 Configure - PLL Spin Period Magnitude - MSBs & LSBs) should load the expected final spin period not the initial spin period.

3. Replace jet select combinations for ATQ#2 command (ADP1 configure - jet control) by "R1 or R2 or R3 or R4 or (R1 & R3) or (R2 & R4) selected."

4. Delete all changes to Comm. S/S & Data Handling S/S - retain operation on HGA, 2048 (or highest supportable) bps rate, Hi Mod. index select & Xmtr low power. Retain scheduled switchover to and from ACS Data format and disabling of (DCE configure - Mode Control -) Phase reset (to avoid HGA pointing transients during the jets firings), however.

5. Just after the post-maneuver issuance of ATQ#7 & ATQ#8 (ADP1 Configure - PLL Spin Period Magnitude), add:

CMD Mnemonic	Command or Action	Remarks & Verification	
		TM Mnemonic	Remarks & TM Data
ATQ#3 SRXX as req'd; PLE; SGAL1; [illegible] SSRM; REFSUN; STN; SUGE; SRRNRM	ADP 1 Configure - ADP Mode Select (PLL Loss of Lock enabled; sun gate enabled).	ASUNGS	= 1 (Sun Gate Enable)
		ALOLES	PLL Loss of Lock = 1 (Enabled)
			All other TM - as before.
DCEQ6	DCE Configure - Mode CMD (Phase Reset Enable)	ADSPNS	Despin Phase Reset = 1 (Enabled)

TABLE 4.3.1.6.3-3

SPACECRAFT CONFIGURATIONS FOR A TYPICAL CONTINUOUS OR 512 MS PULSED (OR 128 MS PULSED THAT REQUIRES CHANGE IN PLL SELECTED SPIN RANGE) - SPIN CHANGE MANEUVER

This is the same as Table 4.3.1.4.3-1 (Typical Precession Maneuver) with the following exceptions:

1. For "Attitude Data Processor - Jet Control" - Just prior to first jet fire":

 (a) Replace jet select combination by: R1 or R2 or R3 or R4 or (R1 & R3) or (R2 & R4) selected.

 (b) For continuous firing: Continuous fire select & time count select.

 (c) For pulse firing: As shown, but 512 ms (or 128 ms) pulse width selected, as applicable.

2. For "Attitude Data Processor - ADP Mode select" -

 (a) "Just prior to first jet fire", add:

 (1) PLL loss of lock inhibited.

 (2) Sun gate disabled

 (3) PLL spin range - selected to encompass expected final spin rate if it is outside of pre-maneuver steady state selected spin range.

 (b) "Assumed steady-state-post maneuver", add:

 (1) PLL loss of lock enabled.

 (2) Sun gate enabled

TABLE 4.3.1.6.3-3 (Continued)

3. For "ADP Configure Commands ATQ#4 through ATQ12" - Just Prior to First Jet Fire" - add:

 PLL Spin period magnitude - set to expected final spin rate.

4. For "DCE - Immediately after last jet fire" - add: "Phase reset enable"

5. For "ADP Configure - Commands ATQ#4 through ATQ12" - for continuous fire:

 (a) Replace all references to "Jet Countdown" by "Time Countdown"

6. For "Attitude Data Processor - Spin Rate" -

 (a) "between first jet fire & Inst jet fire" - add: "changing toward expected final spin rate."

 (b) "Immediately after last jet fire" - add: "At expected final spin rate".

TABLE 4.3.1.6.3-4

TYPICAL SPIN CHANGE (CONTINUOUS OR 512 MS PULSED) (OR 128 MS PULSED THAT REQUIRES CHANGE IN PLL SELECTED SPIN RANGE) - MANEUVER SEQUENCE

This is the same as Table 4.3.1.4.3-2 (Typical Precession Maneuver Sequence) with the following exceptions:

1. Replace all references to "precession" by "spin change maneuver."

2. ATQ$\emptyset$7 and ATQ$\emptyset$8 (or equivalently, ATQ78) commands (ADP1 Configure - PLL Spin Period Magnitude - MSBs and LSBs) should load the expected final spin period, not the initial spin period.

3. Replace jet select combinations for ATQ$\emptyset$2 command (ADP1 Configure - Jet Control) by "R1 or R2 or R3 or R4 or (R1 and R3) or (R2 and R4) selected." For continuous fire, select Continuous Fire and Time Count. For pulsed fire - as shown, but select 512 ms or 128 ms pulse width, as applicable; and Normal or Alternate Mode. Listed telemetry should indicate corresponding to that selected by Command.

4. Just before configuring downlink for forward omni usage, add the following:

CMD Mnemonic	Command or Action	Remarks & Verification	
		TM Mnemonic	Remarks & TM Data
ATQ$\emptyset$3: SRXX as required; PLD; SGA1; SGB1; SSRM; REFSUN; STN; SUGD; SRANRM.	ADP 1 Configure – ADP Mode Select (New spin rate selected, if expected final spin rate exceeds present selected PLL spin range; PLL loss of lock inhibited; Sun Gate disabled).	ASUNGS	= 0 (sun gate disabled)
		ALOLES	PLL Loss of Lock = 0 (Inhibited - remains so throughout the maneuver).
		ASPINS	PLL spin range - encompasses expected final spin rate.
			All other TM - as shown in Table 4.3.1.4.3-2.

TABLE 4.3.1.6.3-4 (Continued)

For a continuous firing: (a) ATQ#4 Command (ADP1 Configure - Jet Countdown) must represent the time count in 0.512 second steps; (b) ATQ#5 Command (ACS Angle Delay Magnitude) is omitted; (c) both the onboard duty cycle detector and PLL Loss of Lock circuit are inhibited during continuous firing and the reference to "... pulses loaded into the countdown circuit" is to be replaced by "... 0.512 msec. time counts loaded into the countdown circuit"; (d) PLIMTI and PBUSLI will both definitely show a change in current level during a continuous firing; (e) AJMAGC (JCE countdown) will be decreasing by one count per 0.512 msec.

Just after the post-maneuver issuance of ATQ#7 and ATQ#8, ADP1 Configure - PLL Spin Period Magnitude, add:

CMD Mnemonic		Remarks & Verification	
		TM Mnemonic	Remarks & TM Data
ATQ#3: SRXX as Required; PLE;SGA1; SGB1;SSRM; REFSUN; STN;SUGE; SRANRM.	ADP1 Configure - ADP Mode Select (PLL Loss of Lock enabled; sun gate enabled).	ASUNGS	= 1 (Sun Gate Enable)
		ALOLES	PLL Loss of Lock = 1 (Enabled)
			All Other TM - As Before.
DCEQ6	DCE Configure - Mode (Phase Reset Enable)	ADSPNS	Despin Phase Reset = 1 (Enabled)
			HGA should despin to stationary inertial reference orientation.
	Transfer of comm. link from fwd omni back to the HGA may be done after assurance that HGA AZ and EL positions are acceptable for the post-maneuver attitude. (Attitude determination is made ahead).		

4.3.2 Attitude Determination. Attitude determination strategy and tactics are described below. A discussion of attitude estimation, the overall mission timeline and the typical command sequence with corresponding telemetry verification are included.

4.3.2.1 General Description. Attitude determination includes all data collecting and data processing necessary to establish the spacecraft attitude (direction of the angular momentum vector) w.r.t. inertial (fixed) coordinates in three-dimensional space. The spacecraft attitude is determined as precisely as practical in order to (1) insure optimum thrust vector orientation for the hydrazine thrusters, and, more importantly, for the Venus Orbit Insertion Maneuver and (2) achieve satisfactory pointing of the HGA and scientific instruments.

Attitude determination may be done (1) in a "quick-look" manner, i.e., with a few samples of data to establish approximately that a desired attitude prevails, or (2) in a "refined" manner, where numerous samples of data from various spatial sources are statistically processed with estimates of sensors biases in order to achieve an accuracy of 0.5° or better. The "refined" approach seeks to minimize random errors by appropriate data averaging; and to minimize systematic errors (i.e., biases) by quantitatively identifying their up-to-date values. The biases may change due to several causes such as thermal effect on mechanical alignment, fuel-depletion effect on mass properties, and change-in-spacecraft-location effect on intensity of, and angle subtended by the spatial reference(s) (such as the sun or earth).

4.3.2.2 Strategy. A general description and the use of two or more spatial references are described below.

4.3.2.2.1 General. Strategy is the basic plan.

Specific strategy for "quick-look" attitude determination consists essentially of (1) collecting a few samples of data of two spatial references, (2) defining the approximate spatial position of the spacecraft and angles between lines of sight to the two spatial references, and (3) processing the data minimally by on-line computer or graphical aids to determine the approximate spacecraft attitude.

Specific strategy for "refined" attitude determination consists essentially of (1) collecting many successive samples of data of lines-of-sight to at least two and preferably three or more point-source spatial references, (2) defining the accurate spatial position of the spacecraft, the angles between lines-of-sight to all spatial references, and up-to-date estimates of biases to be eliminated from the telemetered data, and (3) statistically processing the data via off-line computer to determine the spacecraft attitude as accurately as practical. (The required accuracy does vary with mission events, e.g., TCMs, orbit insertion, on-orbit operation).

4.3.2.2.2 Use of Two or More Spatial References. Collection of data on the line-of-sight to two or more spatial references is sufficient to define spacecraft attitude in inertial (fixed) coordinates.

The accuracy of the results is generally commensurate with the numbers of spatial references beyond two that are used, the amount of data collected and processed to minimize the random errors, and the accuracy of estimation of the systematic errors. Additionally, the accuracy will be degraded when lines-of-sight to spatial references are nearly colinear such that measurements become redundant.

The effect of random errors can be minimized by appropriate statistical averaging, hence the need for collection of many successive samples of

data. The systematic errors include contributions by:

(a) Orbit or trajectory determination errors and their effect on choosing the correct ephemeris data.
(b) The angle subtended by the spatial reference, such as the sun. A point source such as a star is inherently a more accurate spatial reference than the sun.
(c) Mechanical alignments and electrical boresight alignments of sensors, and the effect of thermal distortion upon them.
(d) Sensor calibration uncertainty.
(e) Variation in sensor threshold detection.
(f) Spacecraft wobble.

Representative values for these contributors, excluding the effect of item (a) above, are shown in Table 4.3.2.2-1.

4.3.2.3 Tactics. A general description, attitude sensors and data types, attitude determination, attitude estimation, sun and stars availability during the nominal mission, and an overall mission timeline for attitude determination/estimation are described ahead.

4.3.2.3.1 General. Tactics consist of the specific implementation of the strategy, unique to the spacecraft design and to the point in the mission.

The preferred method for determining spacecraft attitude, via use of sun-to-star(s) roll angle and aspect angle measurements, is functionally independent of the point in the mission, but the choice of stars varies during the mission, as described in Section 3.0 and ahead. Further, the accuracy of determination will vary during the mission for reasons already cited in Section 4.3.2.2.2.

TABLE 4.3.2.2-1

REPRESENTATIVE VALUES FOR ATTITUDE DETERMINATION ERROR CONTRIBUTIONS (EXCLUDES TRAJECTORY ERRORS)

Bias Error Source	Via ψ - $\psi 2$ Measurement:		Via ψ to ψ^* or $\psi 2^*$ Measurement:	
	Midrange (Deg)	Extended Range (Deg)	Midrange (Deg)	Extended Range (Deg)
Sensor Alignment Uncertainty	0.009	0.027	0.061	0.071
S/C Structural Deformation	0.086	0.268	0.171	0.400
Calibration Uncertainty	0.105	0.255	0.085	0.135
Leading Edge Uncertainty (Note 1)	0.1	0.1	0.244	0.331
Threshold Detection Error	0.052	0.052	0.037	0.037
Spacecraft Wobble (0.2^o)	0.140	0.535	0.176	0.754
ΣRSS	0.255	0.661	0.363	0.929
Random Errors				
Jitter (All Sources)	0.148	0.148	0.376	0.387
Nutation (0.1^o)	0.070	0.268	0.088	0.377
ΣRSS	0.164	0.306	0.387	0.540
Total Error (Bias + Random)	0.389	0.967	0.750	1.469

Note 1: Assumes compensation for sun size variation per Figure 3.3.2.1-5 at Earth and Venus.

Note 2: Worst case values are assumed for all components determined for the following separate conditions (Any system condition would produce less values, since some of the separate conditions cannot occur simultaneously):

- Sun is at edge of FOV of each sensor.
- 5 RPM for jitter determination.
- 50 RPM for fixed time errors (i.e., biases plus random errors).
- Maximum wobble and nutation angle as specified in PC-410.

Spacecraft attitude sensors and data types, attitude determination, attitude estimation, availability of spatial references during the mission, and overview timelines are described ahead.

4.3.2.3.2 Attitude Sensors and Data Types. The attitude control subsystem uses sun and star sensors to effect attitude and spin rate measurements. Each sensor has a vertical and a canted slit; Ψ and Ψ_2 for the sun sensor; Ψ^* and Ψ_2^* for the star sensor. Any one of the Ψ, Ψ^*, or Ψ_2^* slits can be used as an azimuthal reference (SBR-selected roll reference) for the purpose of making azimuthal angle measurements (via telemetry of time intervals between occurrences) to Ψ_2, and to any star, including the reference star, using the Ψ^* or Ψ_2^* slit. The geometry of attitude sensor measurements is summarized in Figure 4.3.2.3-1. Further description of the sun and star sensors is detailed in Section 3.3.

The primary and secondary data types employed in the attitude determination process are also shown in Figure 4.3.2.3-1. The three primary data types are sun aspect angle ($\Psi\Psi_2$), sun-to-star roll angle with the vertical star slit (Ψ_2^*), and sun-to-star roll angle with the canted star slit ($\Psi\Psi_2^*$). The ACS mechanization also permits direct measurements of star aspect angle ($\Psi^*\Psi_2^*$), star-to-star roll angle with the vertical star slit ($\Psi^*(1)$ $\Psi^*(2)$), star-to-star roll angle with the canted slit ($\Psi_2^*(1)$ $\Psi_2^*(2)$), and star-to-star roll angle with the vertical and canted star slits ($\Psi^*(1)$ $\Psi_2^*(2)$). These latter four data types can also be derived from the three primary data types.

There are two kinds of errors associated with each data type: random and bias. The random component is caused by threshold jitter, pulse-to-pulse jitter, residual spacecraft nutation and quantization effects. The bias component is caused by leading edge and threshold uncertainty, sensor misalignment, calibration uncertainty and spacecraft wobble. All error sources except the leading edge and threshold uncertainty generate a

bias which is a function of the sun and star aspect angles. However, the effects of all these error sources on the bias of any data type can be obtained from errors in the pairs of angles that define the orientations of all sensor slits. In other words, if each slit is represented by an azimuth angle (defined by the intersection of the slit with the actual spin plane) and a cant angle (defined by rotation about this line of intersection), then errors in these two angles for each slit involved in the measurement of any data type can always be made equivalent to any of the error sources noted above.

4.3.2.3.3 Attitude Determination. Attitude determination, aside from estimation of attitude errors (discussed ahead), employs three schemes that are shown in Table 4.3.2.3-1, along with associated data types, required sensors, and minimum number of required visible stars and measurements. Advantages and disadvantages associated with each scheme are shown in Table 4.3.2.3-2.

4.3.2.3.4 Attitude Estimation. The need for bias estimation is based on the following essential considerations:

(a) Relative azimuth bias in the spin plane between two slits distorts measurements.
(b) Relative cant bias with respect to spin plane between two slits distorts measurements.
(c) Relative azimuth and cant between those slits employed in the measurement process must be estimated to eliminate any out-of-spec attitude error due to sensor biases.
(d) Sun sensor requires estimation at two different attitudes to distinguish between the relative azimuth and cant biases of the ψ and ψ_2 slits.

TABLE 4.3.2.3-1

BASIC SCHEMES FOR ATTITUDE DETERMINATION

SCHEME	DATA TYPES	SENSORS REQUIRED	MINIMUM NO. OF VISIBLE STARS REQUIRED	MINIMUM NUMBER OF MEASUREMENTS REQUIRED
(1) DUAL SLIT: SUN ANGLE + STAR ANGLE	YY_2, $Y^*(i)Y_2^*(i)$	SUN, Y^* SLIT, Y_2^* SLIT	1	2
(2) SINGLE SLIT: SUN ANGLE + SUN TO STAR ROLL ANGLE WITH ONE STAR SENSOR SLIT	YY_2, $YY^*(i)$ OR $YY_2^*(i)$	SUN, Y^* SLIT OR Y_2^* SLIT	1	2
(3) MAPPING: TWO SPIN ANGLES BETWEEN THREE TARGETS	$YY^*(i)$, $YY^*(j)$ OR $YY_2^*(i)$, $YY_2^*(j)$	SUN SENSOR, ONE STAR SLIT	2	2
	OR	OR		
	$Y^*(i)Y^*(j)$ OR $Y_2^*(i)Y_2^*(j)$	ONE STAR SENSOR SLIT	3	
	OR	OR		
	$Y^*(i)Y_2^*(j)$	TWO STAR SENSOR SLITS		

TABLE 4.3.2.3-2

RELATIVE MERITS OF ATTITUDE DETERMINATION SCHEMES

SCHEME	ADVANTAGES	DISADVANTAGES
(1) DUAL SLIT	• USES SUN ANGLE • REQUIRES ONE STAR • GOOD ACCURACY FOR STAR/SUN RELATIVE LONGITUDE > 30°	• SUSCEPTIBLE TO BIAS ERRORS AT LOW SUN ANGLES (w.r.t +Z axis) • USES STAR AS ROLL REFERENCE • REQUIRES BOTH STAR SENSOR SLITS • POOR ACCURACY FOR STAR OPPOSITE SUN
(2) SINGLE SLIT	• USES SUN ANGLE • USES SUN AS ROLL REFERENCE • REQUIRES ONE STAR • REQUIRES ONE STAR SENSOR SLIT • GOOD ACCURACY FOR STAR OPPOSITE SUN	• SUSCEPTIBLE TO BIAS ERRORS AT LOW SUN ANGLE (w.r.t. +Z axis) • POOR ACCURACY FOR STAR/SUN RELATIVE LONGITUDE 60° TO 120°
(3) MAPPING	• CAN USE SUN AS ROLL REFERENCE • CAN USE ONE STAR SENSOR SLIT • THREE TARGETS RELAX ACCURACY DEPENDENCE ON RELATIVE LONGITUDE BETWEEN TARGETS	• REQUIRES AT LEAST TWO STARS • REQUIRES GATING ON AT LEAST TWO STARS • TWO OR MORE STARS MUST SATISFY INTERFERENCE CRITERIA

The need for bias estimation is illustrated quantitatively in Figure 4.3.2.3-2. Regardless of the scheme employed, attitude determination using telemetry alone is not sufficient to reduce the error in spin axis attitude to the 0.5° or less requirement of PC-410 (Reference: Section 1.5.4). Data smoothing and estimation of biases will result in a less than 0.5° error in attitude for the appropriate geometry.

The approach taken in the "refined" attitude determination process is to estimate attitude errors (biases) in those pairs of angles that define slit orientations. While the bias component of the data types is not estimated directly, the mean of the residuals (observed minus computed data) is a measure of how good the estimation is. The attitude estimation algorithm uses this approach.

Four basic attitude estimation schemes, defined by which data types are employed with the estimator, can be identified: The primary scheme (Scheme 1) uses sun aspect angle and sun-to-star roll angles with two star sensor slits; the secondary scheme (Scheme 2) uses sun aspect angle and sun-to-star roll angles with one star sensor slit.

A summary of the various attitude estimation schemes, their data types, number of required measurements (to meet the 0.5 degree attitude determination accuracy) and estimated biases is presented in Table 4.3.2.3-3. Advantages and disadvantages associated with each scheme are shown in Table 4.3.2.3-4. Note that while the first scheme requires the greatest number of measurements, it also requires the fewest number of visible stars, three. Note also that the relative biases estimated using Scheme 1 include or can be used to define those associated with Schemes 2 and 3, and approximate those associated with Scheme 4. Because of these relative advantages, Scheme 1 can be considered the baseline type of attitude estimation. Scheme 3 also employs both star sensor slits and requires four measurements; the latter advantage is offset

by the requirment for four visible stars and added complexity of using a star as the selected roll reference. Scheme 2 requires an additional visible star (relative to Scheme 1) to compensate for the use of only one star sensor slit. Scheme 4 requires the fewest measurements and only one slit; however, four visible stars are required and the fewer measurements relative to Scheme 2 is offset by the added complexity of using a star as the selected roll reference. Schemes 1 and 2 have been examined in detail with respect to the Pioneer Venus mission profile because (i) Scheme 1 essentially solves for all the observable biases with the fewest number of stars; (ii) both schemes employ the sun as an easily detectable selected roll reference, i.e., avoiding a star roll reference by using an increased number of measurements; (iii) "Complete" bias estimation need not be done continuously, only periodically to account for drift and mass properties changes; and (iv) Only one star and one star slit is required to define attitude after each bias "update".

4.3.2.3.5 Sun Availability During the Nominal Mission. For nearly all of the nominal mission, the sun line-of-sight will remain well within the sun sensors' fields-of-view. The period of time when the sun l.o.s. may fall outside the sensors' fields-of-view begins immediately after launch and automatic spacecraft spinup to a nominal 6.5 rpm. At that time, the average sun look-angle may be anywhere between 62° to 80° (Refer to Sections 4.1.1.1 and 4.1.1.2), but additionally, it may be passing out the operating sensor's field-of-view for part of each spin period due to a post-spinup sun look angle variation of 19° maximum. Mag. boom deployment may increase this variation to 30° maximum. Increasing the spacecraft spin rate by _ a factor of 3 (to 15 rpm) will reduce the nutation by a factor of 3, and reduce the nutation damper time constant to <30 minutes. Once the nutation damper has performed its function, and the spacecraft spin axis has been precessed to the nominal cruise attitude of being normal to the ecliptic plane, the sun l.o.s. is expected to remain well within the sun sensor's

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TABLE 4.3.2.3-3

BASIC SCHEMES FOR ATTITUDE ESTIMATION

	SCHEME	DATA TYPES	SENSORS REQUIRED	MINIMUM NO. OF VISIBLE STARS REQUIRED	MIN. NO. OF MEASUREMENTS REQUIRED	BIASES ESTIMATED
①	SUN ANGLE + SUN-TO-STAR ROLL ANGLES WITH TWO STAR SENSOR SLITS	$\Psi\Psi_2$, $\Psi\Psi^*(i)$, $\Psi\Psi_2^*(i)$	SUN, Ψ^* SLIT, Ψ_2^* SLIT	3	7	• RELATIVE AZIMUTH OF Ψ/Ψ^*, Ψ/Ψ_2^* • RELATIVE CANT OF Ψ/Ψ^*, Ψ/Ψ_2^* • RELATIVE AZIMUTH OF Ψ/Ψ_2 OR RELATIVE CANT OF Ψ/Ψ_2
②	SUN ANGLE + SUN-TO-STAR ROLL ANGLES WITH ONE STAR SENSOR SLIT	$\Psi\Psi_2$, $\Psi\Psi^*(i)$ OR $\Psi\Psi_2^*(i)$	SUN, Ψ^* OR Ψ_2^* SLIT	4	5	• RELATIVE AZIMUTH OF Ψ/Ψ^* OR Ψ/Ψ_2^* • RELATIVE CANT OF Ψ/Ψ^* OR Ψ/Ψ_2^* • RELATIVE AZIMUTH OF Ψ/Ψ_2 OR RELATIVE CANT OF Ψ/Ψ_2
③	STAR-TO-STAR ROLL ANGLES WITH TWO STAR SENSOR SLITS	$\Psi^*(i)\Psi_2^*(j)$, $i = j$ AND/OR $i \neq j$	Ψ^* SLIT, Ψ_2^* SLIT	4	4	• RELATIVE AZIMUTH OF Ψ^*/Ψ_2^* • RELATIVE CANT OF Ψ^*/Ψ_2^*
④	STAR-TO-STAR ROLL ANGLES WITH ONE STAR SENSOR SLIT	$\Psi^*(i)\Psi^*(j)$ OR $\Psi_2^*(i)\Psi_2^*(j)$ $i \neq j$	Ψ^* OR Ψ_2^* SLIT	4	3	• CANT OF Ψ^* OR Ψ_2^* WITH RESPECT TO SPIN PLANE

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TABLE 4.3.2.3-3

EASIC SCHEMES FOR ATTITUDE ESTIMATION

SCHEME	DATA TYPES	SENSORS REQUIRED	MINIMUM NO. OF VISIBLE STARS REQUIRED	MIN. NO. OF MEASUREMENTS REQUIRED	BIASES ESTIMATED
① SUN ANGLE + SUN-TO-STAR ROLL ANGLES WITH TWO STAR SENSOR SLITS	YY_2, $YY^*(1)$, $YY_2^*(1)$	SUN, Y^* SLIT, Y_2^* SLIT	3	7	• RELATIVE AZIMUTH OF Y/Y^*, Y/Y_2^* • RELATIVE CANT OF Y/Y^*, Y/Y_2^* • RELATIVE AZIMUTH OF Y/Y_2 OR RELATIVE CANT OF Y/Y_2
② SUN ANGLE + SUN-TO-STAR ROLL ANGLES WITH ONE STAR SENSOR SLIT	YY_2, $YY^*(1)$ OR $YY_2^*(1)$	SUN, Y^* OR Y_2^* SLIT	4	5	• RELATIVE AZIMUTH OF Y/Y^* OR Y/Y_2^* • RELATIVE CANT OF Y/Y^* OR Y/Y_2^* • RELATIVE AZIMUTH OF Y/Y_2 OR RELATIVE CANT OF Y/Y_2
③ STAR-TO-STAR ROLL ANGLES WITH TWO STAR SENSOR SLITS	$Y^*(1)Y_2^*(j)$, $i = j$ AND/OR $i \neq j$	Y^* SLIT, Y_2^* SLIT	4	4	• RELATIVE AZIMUTH OF Y^*/Y_2^* • RELATIVE CANT OF Y^*/Y_2^*
④ STAR-TO-STAR ROLL ANGLES WITH ONE STAR SENSOR SLIT	$Y^*(1)Y^*(j)$ OR $Y_2^*(1)Y_2^*(j)$ $i \neq j$	Y^* OR Y_2^* SLIT	4	3	• CANT OF Y^* OR Y_2^* WITH RESPECT TO SPIN PLANE

TABLE 4.3.2.3-4

RELATIVE MERITS OF ATTITUDE ESTIMATION SCHEMES

SCHEME	ADVANTAGES	DISADVANTAGES
1 SUN ANGLE + SUN TO STAR ROLL ANGLES WITH BOTH SLITS	• USES SUN AS ROLL REFERENCE • REQUIRES LEAST NO. OF STARS (3) • BIASES ESTIMATED FROM ANY OTHER SCHEME CAN BE DERIVED • DONE PERIODICALLY; THEN ONE STAR/ANY SLIT USEFUL	• REQUIRES MOST MEASUREMENTS (7)
2 SUN ANGLE + SUN TO STAR ROLL ANGLES WITH ONE SLIT	• USES SUN AS ROLL REFERENCE • REQUIRES ONE SLIT • DONE PERIODICALLY; THEN ONE STAR/ONE SLIT USEFUL	• REQUIRES 4 STARS • REQUIRES 5 MEASUREMENTS • REQUIRES USE OF PARTICULAR SLIT WHOSE BIASES HAVE BEEN ESTIMATED
3 STAR TO STAR ROLL ANGLES WITH TWO SLITS	• INDEPENDENT OF SUN • CAN INFER ONE SUN SENSOR BIAS; HOWEVER, THIS REQUIRES 5 MEASUREMENTS	• USES STAR AS ROLL REFERENCE • REQUIRES 4 STARS • MINIMUM OF 2 STARS ARE REQUIRED FOR DETERMINISTIC SOLUTION SUBSEQUENTLY
4 STAR TO STAR ROLL ANGLES WITH ONE SLIT	• INDEPENDENT OF SUN • MINIMUM NO. OF MEASUREMENTS (3) • ONE SUN SENSOR BIAS CAN BE INFERRED FROM YY_2 DATA REQUIRES 4 MEASUREMENTS	• USES STAR AS ROLL REFERENCE • REQUIRES 4 STARS • MINIMUM OF 3 STARS ARE REQUIRED FOR DETERMINISTIC SOLUTION SUBSEQUENTLY

field-of-view. This is illustrated in Section 4.1 for the cruise phase (Figure 4.1.1.2-8), the range of nominal orbit insertion attitudes (Figure 4.1.1.2-13), and the On-Orbit phase (Figure 4.1.1.2-16).

4.3.2.3.6 Star Availability During the Nominal Mission. The Orbiter cruise, orbit, and orbit insertion maneuver phases were simulated using the previously introduced estimation techniques to ascertain the feasibility of meeting the attitude determination accuracy specifications. Since the accuracy of the estimator is dependent upon the number of independent measurements, and therefore, the number of stars sighted, bar charts of star availability were created for the Orbiter cruise and in-orbit phases of the nominal mission. Figures 4.3.2.3-3 and 4.3.2.3-4 are time line bar charts showing the availability of visible stars for each star sensor slit when no interference by the sun occurs. Figures 4.3.2.3-3 and 4.3.2.3-5 include time line bar charts indicating the total number of visible stars by either star sensor slit. For the orbit phase, where the attitude is nominally at the south ecliptic normal, it could be advantageous to reorient the spin axis attitude to (ecliptic latitude) $\theta = 178^{\circ}$, and (ecliptic longitude) $\phi = 0^{\circ}$ to potentially acquire Gacrux and Fomalhaut. An additional bar chart for this orientation has been drawn (Figure 4.3.2.3-6) to demonstrate the resulting increased number of stars.

These charts indicate that except for the Orbiter cruise phase, three or more stars are available for more than 90 percent of the time. For the failure mode, when one star sensor slit is inoperative, the required four stars needed for complete observability are available 46.1 percent of the time during the orbit phase (ecliptic normal attitude). If the attitude is reoriented to $\theta = 178^{\circ}$, $\phi = 0^{\circ}$, four stars are available 94.3 percent of the time during orbit phase.

For the Orbiter cruise phase, three or more stars are available 79 percent of the time, and four

stars are available only 7.5 percent of the time for the failure mode.

Figure 4.3.2.3-7 depicts the potential attitudes and available stars for the Orbiter at orbit insertion.

4.3.2.3.7 Overall Attitude Estimation Accuracy. Table 4.3.2.3-5 lists some prominent events in the Orbiter nominal mission when an attitude determination, either "Quick-Look" or "Refined", in conjunction with an attitude estimation, are recommended to occur.

4.3.2.4 Typical Command Sequences and Telemetry Verification. The initial Spacecraft Configuration for a typical attitude determination sequence is shown in the "Assumed Steady-State (Pre-Maneuver)" column of Table 4.3.1.4.3-1.

The feasible telemetered data types for use in determining or estimating attitude are shown in Matrix format in Table 4.3.2.4-1. A specific step-by-step command sequence, that includes corresponding telemetry verification, for each telemetered data type is identified by Table number in each cell of this matrix. All of these tables are included in this Section 4.3.2.4.

TABLE 4.3.2.3-5

EXPECTED ATTITUDE ESTIMATION ACCURACY FOR SOME MISSION EVENTS

Event Start Time	Event	Action	Remarks	Estimation Accuracy (Stars/Slits Available)			
				3/2	4/1	2/2	3/1
≈L+5 Hr (1 Hr After Initial Ground Station Acquisition). L = Launch Time	Just before Mag. Boom deployment.	(1) First "quick-look" Att. Det. w.r.t. sun l.o.s.	• Average sun look angle nominally <80° to maintain OMAG boom and hinges above 0°F.				
≈L+6 Hrs.	Just after Mag. Boom deployment.	(1) "Quick-look" att. det. w.r.t. sun l.o.s., spin rate determination, then: (2) Spin-up to 15 rpm. (3) Wait up to two hours for nutation to diminish to <0.1°. (4) Post-deployment attitude determination that includes mapping of available stars. (5) Reorient spin axis to normal to ecliptic plane and pointed North. (6) Despin HGA. (7) Bias estimation and Attitude Determination using available stars.	Residual nutation is expected to be <0.1° within two hours after spinup to 15 rpm due to shorter time constant (< 30 minutes) of nutation damper at 15 rpm.			0.04° to 0.18°	0.03° to 0.1°
E-2 Days	VOI (Orbit Insertion)	(1) Bias estimation and Attitude Determination just before and after precession to VOI attitude. (2) Spinup to 30 rpm, then perform final attitude determination prior to OIM burn.	• Calibration prior to reor permits use of 1 star to define attitude to <1°. • Option to reor to 2-star attitude, calibrate jets; then execute small reor and use 1 star. • Option to use several dim stars within sensor detection capability.			0.2°	

TABLE 4.3.2.3-5 (Continued)

Event Start Time	Event	Action	Remarks	Estimation Accuracy (Stars/Slits Available)			
				3/2	4/1	2/2	3/1
E+1 Hour	Configure S/C and Trim Orbit.	(1) Perform 1st post-OIM Burn Attitude Determination (2) Despin to 15 rpm. (3) Reorient spin axis to ecliptic normal and pointed South. (4) Bias estimation, attitude determination and attitude trim, as required.				0.06° to 0.25°	0.06° to 0.34°

TABLE 4.3.2.4-1

MEASUREMENTS OF S/C ATTITUDE TELEMETERED DATA TYPES

(NOTE: All sequences assume that the SRR has already been established per Section 4.3.3)

Measure: To: / From (established SRR):	PSI	PSI2	PSI*	PSI2*
PSI	• Produces spin period • Part of sequence in Table 4.3.2.4-2	• Perform sequence in Table 4.3.2.4-2	• Perform sequence in Table 4.3.2.4-3	• Perform sequence in Table 4.3.2.4-4
PSI*	Not possible for S/C design	• Perform sequence in Table 4.3.2.4-2	• For the same star, produces spin period • For different stars, perform sequence in Table 4.3.2.4-5	• Perform the sequence in Table 4.3.2.4-6
PSI2*	Not possible for S/C design	• Perform sequence in Table 4.3.2.4-2	• Perform the sequence in Table 4.3.2.4-6	• For the same star, produces spin period • For different stars, perform sequence in Table 4.3.2.4-5

TABLE 4.3.2.4-2
MEASUREMENT OF SUN ASPECT (ELEVATION) ANGLE ($\psi\ \psi_2$ OR ψ_2* ψ_2 OR $\psi*$ ψ_2)

COMMAND MNEMONIC OR STEP #	COMMAND OR ACTION	REMARKS & VERIFICATION	
		TELEMETRY MNEMONIC	REMARKS & TM DATA
(1)	• Given the initial spacecraft configuration shown in the first column of Table 4.3.1.4.3-1, use the existing SRR, which is assumed to be either the sun or a known star.		
(2)	• If not already so, configure ADP measurements to obtain data for calculation of sun elevation angle:		
ATQ01 (For ADP1 Only) (MEA1 = SRR, MEA2 = S2S)	ADP1 Configure - Measurement Select ((SRR to SRR) and (SRR to PSI2)).	AM1ADS	Measurement A Address: (SRR to SRR)
		AM2ADS	Measurement B Address: (SRR to PSI2)
		ATTM1Z OR ATTM2Z	Measurement Data: Reflects above data according to:
		ATTMSS	Data Measurement = 1 (A) or = 0 (B), alternating no more often than once per spin period, as explained in Section 3.3.2.3.6 and 3.3.3.6.

TABLE 4.3.2.4-2. (Continued)

Command Mnemonic	Command or Action	Remarks & Verification	
		Telemetry Mnemonic	Remarks & TM Data
(3)	• Accumulate at lease 30 valid samples of each measurement. Accept as a valid sample: (1) Only the first ATTM1Z or ATTM2Z sample following a change in ATTMSS. (2) Only samples accompanied by the indicated telemetry at the right:		
		ALOCKS	PLL Loss of Lock = 1 (no loss of lock)
		AMISSC	Missed SRR Count = No change from last sample of this parameter.
		ALOLES	PLL Loss of Lock = 1 (Enabled } Not required but preferred
		ASUNGS	Sun Gate = 1 (Enabled) if the Sun is the SRR. } Not required but preferred
(4)	• Determine the mean of each set of 30 samples – Reject samples outside of 3σ deviation from the mean and recalculate the mean of the remaining samples.		
(5)	• Calculate "spin angle" for SRR to Sun: $AZEL_{SUN}$ = (Mean of (SRR to ψ_2) samples) $\div$ (Mean of (SRR to SRR) samples) - $BIAS_{ELSUN}$, where $BIAS_{ELSUN}$ = value obtained from Table in "Remarks" column.		SRR: Corresponding value (from Tables 3.3.2.4-2 and 3.3.2.4-3 for $BIAS_{ELSUN}$: PSI Zero. PSI* -(1)-(2)-(3)+(A)+(B)+(C). PSI2* -(1)-(2)-(3)+(A)+(C).

TABLE 4.3.2.4-2 (Continued)

COMMAND MNEMONIC OR STEP #	COMMAND OR ACTION	REMARKS & VERIFICATION	
		TELEMETRY MNEMONIC	REMARKS & TM DATA
(6)	• If the SRR is PSI, declare $EL_{SUN} = AZEL_{SUN}$ and proceed to the next step. • If the SRR is PSI* or PSI2*, then calculate: $\theta = AZEL_{SUN} - SEP_{STARSUN}$, where $SEP_{STARSUN}$ = Azimuthal separation angle from the SRR star to the sun, measured in the plane normal to the S/C spin axis (measured in direction of spin). • If $\theta < 0^{o}$, then $EL_{SUN} = 360^{o} + \theta$ • If $\theta \geq 0^{o}$, then $EL_{SUN} = \theta$.		
(7)	• Use any one of the following to determine the sun aspect (elevation) angle (approximate) w.r.t. the S/C spin axis, using the value EL_{SUN} as the known spin angle delay: (1) The appropriate equation in Figure 3.3.2.1-1 or Figure 3.3.2.1-2. OR (2) The appropriate curve in Figure 3.3.2.1-5. OR (3) Any applicable nomogram - or table, when assembled, in the PV Telemetry Data Conversion Handbook (Reference Para. 1.5.16).		

TABLE 4.3.2.4-3

MEASUREMENT OF SUN TO STAR USING THE PSI* SLIT ($\psi\psi^*$)

COMMAND MNEMONIC OR STEP #	COMMAND OR ACTION	REMARKS & VERIFICATION	
		TELEMETRY MNEMONIC	REMARKS & TM DATA
	Because a spin angle measurement using the PSI* slit is dependent on the elevation of a star, as seen in Figures 3.3.2.2-3A and 3.3.2.2-3B, determination of the roll (azimuth) angle to a star requires use of the PSI2* slit as well as the PSI* slit. Once spin angle measurements to the star using both slits are made, the roll (azimuth) and aspect (elevation) angles to the star may be determined concurrently. The following sequence details the first basic step in this procedure and references the reader to the final basic step.		
(1)	• Given the initial spacecraft configuration shown in the first column of Table 4.3.1.4.3-1, use the existing SRR, which is assumed to be the sun.		
(2)	Configure the PSI* and PSI2* channels for the measurement star:		
STRQ1 OR STRQA	Star Sensor Threshold and Bandpass Select. (Both channels set to threshold setting and bandpass compatible with the measurement star.	A*1THS	PSI* Threshold Setting — All Conform to Commanded State
		A*2THS	PSI2* Threshold Setting
		A*1BPS	PSI* Bandpass State
		A*2BPS	PSI2* Bandpass State
(3)	Assign PSI* Channel to Star Gate B:		
ATQØ3 (For ADP1 Only) (SGB = 1)	ADP 1 Configure - ADP Mode Select (Star Gate B = Channel 1 (PSI*); All other ADP Mode States unchanged).	A*GBSS	Star Gate B = 1 (= Channel 1 = PSI*)
(4)	Locate Star Gate B over the measurement Star:		

TABLE 4.3.2.4-3 (Continued)

Command Mnemonic	Command or Action	Remarks & Verification	
		Telemetry Mnemonic	Remarks & TM Data
ATQ05 (For ADP1 Only) (ACSANG = XXX.XX)	ADP1 Configure – ACS Angle Delay Magnitude (Calculate $\theta = SEP_{SUNSTAR} - BIAS_{PSIPSI*} - 0.352^o - 5.625^o$ (to locate star in the center of Star Gate B) where $SEP_{SUNSTAR}$ = estimated azimuthal separation angle measured in plane normal to spin axis from the sun l.o.s. to the star l.o.s. (measured in the direction of $\overline{S/C}$ spin); and $BIAS_{PSIPSI*}$ = Value defined in Remarks column. If θ is positive, then the commanded magnitude = θ. If θ is negative, then the commanded magnitude = $360^o + \theta$.	AACSAD	ACS Angle Delay Magnitude = (same magnitude as that commanded)
			• Bias $_{PSIPSI*}$ (values obtained from Tables 3.3.2.4-2 and 3.3.2.4-3) = -(1)-(2)-(3)+(A)+(B)+(C).
			• Actual delay is 0.352^o larger than the commanded value.
(5)	Configure ATTM for star data:		
ATQ01 (For ADP1 Only) (MEA1 = SRR, MEA2 = GSB)	ADP 1 Configure – Measurement Select (SRR to SRR) and (SRR to Gated Star B).	AM1ADS	Measurement A Address: (SRR to SRR)
		AM2ADS	Measurement B Address: (SRR to Star Gate B)
		ATTM1Z OR ATTM2Z	Measurement Data – Reflects above data according to:
		ATTMSS	Data Measurement = 1(A) or = 0(B), alternating no more often than once per spin period as explained in Sections 3.3.2.3.6 and 3.3.3.6.

TABLE 4.3.2.4-3 (Continued)

Command Mnemonic	Command or Action	Remarks & Verification	
		Telemetry Mnemonic	Remarks & TM Data
(6)	• Accumulate at least 30 valid samples of each measurement. Accept as a valid sample: 1) Only the first ATTM1Z or ATTM2Z sample following a change in ATTMSS. 2) Only samples accompanied by the indicated telemetry at the right:		
		ALOCKS	PLL Loss of Lock = 1(No loss of lock)
		AMISSC	Missed SRR Count = No change from last sample of this parameter.
		ALOLES	PLL Loss of Lock = 1 (Enabled) } not necessary but preferred
		ASUNGS	Sun Gate = 1 (Enabled) } not necessary but preferred
		ASRRMS	Sun/Star Select = 1 (Sun) } not necessary but preferred
(7)	Determine the mean of each set of 30 samples - reject samples outside of 3σ deviation from the mean; then recalculate the mean of the remaining samples.		
(8)	Calculate Spin Angle: $AZEL_{STAR} = \{$(Mean of (SRR to Gated Star B) Samples) $\div$ (Mean of (SRR to SRR) Samples)$\} + BIAS_{PSIPSI*}$ The resulting value is dependent on the elevation of the star as well as its azimuthal separation (pure "roll"angle) from the sun. To determine these values, proceed to Table 4.3.2.4-4.		• $BIAS_{PSPSI*}$ WAS DEFINED IN Step (4)

TABLE 4.3.2.4-4

MEASUREMENT OF SUN-TO-STAR ASPECT ANGLE ($\psi\psi_2$* and $\psi\psi'$)

COMMAND MNEMONIC OR STEP #	COMMAND OR ACTION	REMARKS & VERIFICATION	
		TELEMETRY MNEMONIC	REMARKS & TM DATA
(1)	• Given the initial spacecraft configuration shown in the first column of Table 4.3.1.4.3-1, use the existing SRR, which is assumed to be the sun.		
(2)	Configure the PSI* and PSI2* channels for the measurement star.		
STRQ1 OR STRQA	Star Sensor threshold and bandpass select. (Both channels set to threshold setting and bandpass compatible with the measurement star.	A*1THS	PSI* Threshold Setting
		A*2THS	PSI2* Threshold Setting
		A*1BPS	PSI* Bandpass State
		A*2BPS	PSI2* Bandpass State
			All conform to Commanded State.
(3)	Assign PSI2* Channel to Star Gate B:		
ATQ03 (for ADP1 Only) (SGB = 2)	ADP 1 Configure - ADP Mode select (Star Gate B = Channel 2 (PSI2*); All other ADP Mode States unchanged).	A*GBSS	Star Gate B = 0 (= Channel 2 = PSI2*).
(4)	Locate Star Gate B over the measurement star.		
ATQ05 (for ADP1 Only) (ACSANG = XXX.XX)	ADP 1 Configure - ACS Delay Magnitude (Calculate θ = $SEP_{SUNSTAR}$ - $BIAS_{PSIPSI2*}$ - 0.352^{O} - 5.625^{O} (to locate star in the center of Star Gate B), where $SEP_{SUNSTAR}$ = Estimated azimuthal separation angle measured in plane normal to spin axis from the sun l.o. to the star l.o.s. (measured in the direction of S/C spin); and $BIAS_{PSIPSI2*}$ = value obtained from Remarks column.	AACSAD	ACS Angle Delay Magnitude = (same magnitude as that commanded)
			• $BIAS_{PSIPSI2*}$ (values obtained from Tables 3.3.2.4-2 and 3.3.2.4-3) = -(1)-(2)-(3)+(A)+(C).
			• Actual delay is 0.352^{O} larger than the commanded value.
	• If θ is positive, then the commanded magnitude = θ.		
	• If θ is negative, then the commanded magnitude = 360^{O} + θ.		

TABLE 4.3.2.4-4 (Continued)

COMMAND MNEMONIC OR STEP #	COMMAND OR ACTION	REMARKS & VERIFICATION	
		TELEMETRY MNEMONIC	REMARKS & TM DATA
(5)	Configure ATTM for Star Data:		
ATQØ1 (for ADP1 Only) (MEA1 = SRR, MEA2 = SGB)	ADP 1 Configure - Measurement Select (SRR to SRR) and (SRR to Gated Star B).	AM1ADS	Measurement A Address: (SRR to SRR)
		AM2ADS	Measurement B Address: (SRR to Star Gate B).
		ATTM1Z or ATTM2Z	Measurement Data - Reflects above data according to:
		ATTMSS	Data Measurement = 1(A) or = 0(B), alternating no more often than once per spin period as explained in Sections 3.3.2.3.6 and 3.3.3.6.
(6)	• Accumulate at least 30 valid samples of each measurement. Accept as a valid sample: 1. Only the first ATTM1Z or ATTM2Z sample following change in ATTMSS. 2. Only samples accompanied by the indicated telemetry at the right:	ALOCKS	PLL Loss of Lock = 1 (No loss of lock).
		AMISSC	Missed SRR Count = No change from last sample of this parameter.
		ALOLES	PLL Loss of Lock = 1 (Enabled) — Not Necessary But Preferred.
		ASUNGS	Sun Gate = 1 (Enabled) — Not Necessary But Preferred.
		ASRRMS	Sun/Star Select = 1 (Sun) — Not Necessary But Preferred.
(7)	Determine the mean of each set of 30 samples - reject samples outside of 3∇ deviation from the mean; then recalculate the mean of the remaining samples.		
(8)	Calculated biased "spin angle" for sun-to-star: AZELPSI2* = {Mean of (SRR to Gated Star B Samples) $\div$ (Mean of SRR to SRR) Samples)}.		

TABLE 4.3.2.4-4 (Continued)

COMMAND MNEMONIC OR STEP #	COMMAND OR ACTION	REMARKS & VERIFICATION	
		TELEMETRY MNEMONIC	REMARKS & TM DATA
(9)	• The resulting AZELPSI2* includes an azimuth (roll) component and an elevation (aspect) component for the measurement star, as well as the biases between the PSI slit and the PSI2* slit. • The actual aspect (elevation) angle of the measurement star, measured w.r.t. the S/C spin axis, may be determined as follows: (a) Determine the spin angle delay using the PSI* slit by performing Steps (1) through (7) of the sequence in Table 4.3.2.4-3 (if not already performed). (b) From the mean values determined in Step (7) of Table 4.3.2.4-3, calculate: AZELPSI* = {Mean of (SRP to Gated Star B) Samples} ÷ {Mean of (SRR to SRR) Samples}. (c) Calculate EL_{STAR} = AZELPSI2* - AZELPSI* . (d) Use Figure 3.3.2.2-4A or Figure 3.3.2.2-4B where EL_{STAR} ≡ PSI* to PSI2* spin angle delay in degrees, to obtain elevation angle of the star w.r.t. the S/C spin axis.		
(10)	The actual roll (azimuth) angle of the measurement star, measured w.r.t. the sun in the spacecraft spin plane and in the direction of positive spacecraft spin, may be determined as follows: The roll (azimuth angle) = AZPSI2* = AZELPSI2* + AZBIAS + EL_{STAR}, where AZBIAS = value obtained from Remarks column.		AZBIAS (values obtained from Tables 3.3.2.4-2 and 3.3.2.4-3)= -(1)-(2)-(3)+(C).

TABLE 4.3.2.4-5

MEASUREMENT OF STAR "SPIN" ANGLE USING THE SRR STAR CHANNEL ($\psi^*\psi^*$ OR $\psi_2^*\psi_2^*$)

COMMAND MNEMONIC OR STEP #	COMMAND OR ACTION	REMARKS & VERIFICATION	
		TELEMETRY MNEMONIC	REMARKS & TM DATA
	Because a spin angle measurement using the PSI* slit is dependent on elevation of a star, as seen in Figures 3.3.2.2-3A and 3.3.2.2-3B, determination of the roll (azimuth) angle to a star requires use of the PSI2* slit as well as the PSI* slit. Once spin angle measurements to the star using both slits are made, the roll (azimuth) and aspect (elevation w.r.t. the SRR star) angles to the measurement star may be determined concurrently. The following sequence details the first basic step in this procedure, and references the reader to the final basic step.		
(1)	• Given the initial spacecraft configuration shown in the first column of Table 4.3.1.4.3-1, use the existing SRR, which is assumed to be a known star (call it the "SRR" Star). • This sequence is to be used only for either one of the following two cases: (1) PSI* is the SRR channel, and PSI* will also be used for Star Gate B. (2) PSI2* is the SRR channel, and PSI2* will also be used for Star Gate B.		
(2)	• It is undesirable to change the threshold setting and bandpass setting for the star channel that is being used as the SRR channel. This is to prevent loss of the present SRR and subsequent loss of DCE control.		
(3)	• If not already so, assign the Star channel being used as the SRR channel (i.e., Star Gate A) also to Star Gate B:		
ATQ03 (for ADP1 Only)	ADP 1 Configure - ADP Mode Select (Star Gate A - Retain present channel selection. Star Gate B - Make the same as for Star Gate A (1 for PSI*; 2 for PSI2*).	A*GASS	Star Gate A - No change in TM reading.
		A*GBSS	Star Gate B = Same as for Star Gate A (= 1 for PSI*, = 0 for PSI2*).

TABLE 4.3.2.4-5 (Continued)

COMMAND MNEMONIC OR STEP #	COMMAND OR ACTION	REMARKS & VERIFICATION	
		TELEMETRY MNEMONIC	REMARKS & TM DATA
(4)	• Locate Star Gate B over the star to be measured (call it the measurement star):		
ATQ05 (for ADP1 Only) (ACSANG = XXX.XX)	ADP 1 Configure - ACS Angle Delay Magnitude. Calculate θ = SEP_{STARS} - 5.625° (to locate measurement star in the center of Star Gate B) - 0.352°, where SEP_{STARS} = Estimated azimuthal separation angle measured in the plane normal to the spin axis from the SRR star l.o.s. to the measurement star l.o.s. (measured in the direction of S/C spin). • If θ is positive, then the commanded magnitude = θ . • If θ is negative, then the commanded magnitude = 360° + θ .	AACSAD	ACS Angle Delay Magnitude = (Same magnitude as that commanded).
			• Actual delay is 0.352° larger than the commanded value.
(5)	Configure ATTM for Star Data:		
ATQ01 (for ADP1 Only) (MEA1 = SRR, MEA2 = SGB)	ADP 1 Configure - Measurement Select. (SRR to SRR) and (SRR to Gated Star B).	AM1ADS	Measurement A Address: (SRR to SRR)
		AM2ADS	Measurement B Address: (SRR to Star Gate B)
		ATTM1Z OR ATTM2Z	Measurement Data - Reflects above data according to:
		ATTMSS	Data Measurement = 1(A) or = 0(B), alternating no more often than once per spin period as explained in Sections 3.3.2.3.6 and 3.3.3.6.

TABLE 4.3.2.4-5 (Continued)

COMMAND MNEMONIC OR STEP #	COMMAND OR ACTION	REMARKS & VERIFICATION	
		TELEMETRY MNEMONIC	REMARKS & TM DATA
(6)	• Accumulate at least 30 valid samples of each measurement. Accept as a valid sample: (1) Only the first ATTM1Z or ATTM2Z sample following a change in ATTMSS. (2) Only samples accompanied by the indicated telemetry at the right.	ALOCKS	PLL Loss of Lock = 1 (No loss of lock)
		AMISSC	Missed SRR Count = No change from last sample of this parameter.
		ALOLES	PLL Loss of Lock = 1 (Enabled) — Not Necessary but Preferred.
		ASRRMS	Sun/Star Select = 0 (Star) — Not Necessary but Preferred.
(7)	Determine the mean of each set of 30 samples - reject samples outside of 3σ deviation from the mean; then recalculate the mean of the remaining samples.		
(8)	• Calculate "spin angle" from the SRR star to the measurement star: AZEISAME = (Mean of (SRR to measurement star) samples $\div$ (Mean of (SRR to SRR) samples).		
(9)	• The resulting value is dependent on the elevation of the measurement star, its azimuthal separation (pure "roll" angle) from the SRR star, and the (known) elevation angle of the SRR star. To determine these values, proceed to Table 4.3.2.4-6. If the PSI* slit was used for the SRR in this sequence, then perform the PSI* to PSI2* measurement in Table 4.3.2.4-6. Otherwise, perform the PSI2* to PSI* measurement in Table 4.3.2.4-6.		

TABLE 4.3.2.4-6

MEASUREMENT OF STAR ASPECT (ELEVATION) ANGLE ($\psi^* \psi_2^*$ AND $\psi^* \psi^*$) OR ($\psi_2^* \psi^*$ AND $\psi_2^* \psi_2^*$)

COMMAND MNEMONIC OR STEP #	COMMAND OR ACTION	REMARKS & VERIFICATION	
		TELEMETRY MNEMONIC	REMARKS & TM DATA
(1)	• Given the initial spacecraft configuration shown in the first column of Table 4.3.1.4.3-1, use the existing SRR, which is assumed to be a known star (call it the "SRR" star). • This sequence is to be used only for either one of the following two cases: (1) PSI* is the SRR channel, and PSI2* will be used for Star Gate B. (2) PSI2* is the SRR channel, and PSI* will be used for Star Gate B.		
(2)	• If not already so, configure the star channel that is not being used as the SRR star channel, for the star to be measured (call it the "measurement" star):		• For the star channel that is the present SRR channel, it is undesirable to change the threshold selection and the bandpass selection. This is to prevent loss of the present SRR star and subsequent loss of DCE control.
STRQ1 OR STRQA	Star Sensor Threshold and Bandpass Select. (Threshold and Bandpass selections are retained on the star channel (PSI* or PSI2*) that is being used as the SRR channel; the alternate channels settings are changed to conform to the measurement star's requirements.)	A*1THS	PSI* Threshold Setting (All Conform to Commanded State)
		A*2THS	PSI2* Threshold Setting
		A*1BPS	PSI* Bandpass State
		A*2BPS	PSI2* Bandpass State
(3)	• Select Star Gate B for the measurement star's channel, but retain Star Gate A selection for the SRR Star:		
ATQ#3 (for ADP1 Only)	ADP 1 Configure - ADP Mode Select (Star Gate A - Retain present channel selection. Star Gate B - Channel 1 (if PSI* was selected in Step (2)) or Channel 2 (if PSI2* was selected in Step (2)).	A*GASS	Star Gate A - No change in TM Reading.
		A*GBSS	Star Gate B = 1 (PSI*) or = 0 (PSI2*), as commanded.

TABLE 4.3.2.4-6 (Continued)

COMMAND MNEMONIC OR STEP #	COMMAND OR ACTION	REMARKS & VERIFICATION		
		TELEMETRY MNEMONIC	REMARKS & TM DATA	
(4)	• Locate Star Gate B over the measurement star:			
ATQ#5 (for ADP1 Only) (ACSANG = XXX.XX)	ADP 1 Configure - ACS Angle Delay Magnitude. (Calculate $\theta = SEP_{STARS} - BIAS_{STARS} - 0.352^{o} - 5.625^{o}$ (to locate measurement star in the center of Star Gate B), where SEP_{STARS} = Estimated Azimuthal separation angle measured in plane normal to spin axis from the SRR Star l.o.s. to the measurement star l.o.s. (measured in the direction of S/C spin); and $BIAS_{STARS}$ = value obtained from Table in "Remarks" column). • If θ is positive, then the commanded magnitude = θ. • If θ is negative, then the commanded magnitude = $360^{o} + \theta$.	AACSAD	ACS Angle Delay Magnitude = (same magnitude as that commanded).	
			SEP_{STARS} = Zero if the SRR Star and the measurement star are the same.	
			• Actual delay is 0.352^{o} larger than the commanded value.	
			SRR	Corresponding value for $BIAS_{STARS}$:
			PSI*	$BIAS_{STARS} = \delta$, where δ is the PSI* to PSI2* spin angle delay in degrees, from Figure 3.3.2.2-4A or -4B, that corresponds to the predicted relative elevation angle of the measurement star w.r.t. the SRR star. NOTE: Use of these figures for bias estimation is not exact when stars of <u>different</u> elevations are involved, as the delay is dependent on SRR star elevation in such a circumstance. However, they are sufficiently accurate for use in positioning Gate B.
			PSI2*	$BIAS_{STARS} = -\delta$, where δ is defined above.

TABLE 4.3.2.4-6 (Continued)

COMMAND MNEMONIC OR STEP #	COMMAND OR ACTION	REMARKS & VERIFICATION	
		TELEMETRY MNEMONIC	REMARKS & TM DATA
(5)	Configure ATTM for Star Data:		
ATQ#1 (for ADP1 Only)	ADP 1 Configure - Measurement Select. (SRR to SRR) and (SRR to Gated Star B).	AM1ADS	Measurement A Address: (SRR to SRR)
		AM2ADS	Measurement B Address: (SRR to Star Gate B)
		ATTM1Z OR ATTM2Z	Measurement Data - Reflects above data according to:
		ATTMSS	Data Measurement = 1(A) or = 0(B), alternating no more often than once per spin period as explained in Sections 3.3.2.3.6 & 3.3.3.6.
(6)	• Accumulate at least 30 valid samples of each measurement. Accept as a valid sample: (1) Only the first ATTM1Z or ATTM2Z sample following a change in ATTMSS. (2) Only samples accompanied by the indicated telemetry at the right:	ALOCKS	PLL Loss of Lock = 1 (No loss of lock)
		AMISSC	Missed SRR Count = No change from last sample of this parameter.
		ALOLES	PLL Loss of Lock = 1 (Enabled) } Not Necessary But Preferred
		ASRRMS	Sun/Star Select = 0 (Star) } Not Necessary But Preferred
(7)	Determine the mean of each set of 30 samples - reject samples outside of 3σ deviation from the mean; then recalculate the mean of the samples.		
(8)	• Calculate biased "Spin Angle" from SRR Star to the measurement star: AZELDIFF = [(Mean of (SRR to measurement star) samples) ÷ (Mean of (SRR to SRR) samples)].		

TABLE 4.3.2.4-6 (Continued)

COMMAND MNEMONIC OR STEP #	COMMAND OR ACTION	REMARKS & VERIFICATION	
		TELEMETRY MNEMONIC	REMARKS & TM DATA
(9)	• The actual aspect (elevation) angle of the measurement star, measured w.r.t. the S/C spin axis, may be determined as follows: (a) Retain the star slit in use as the SRR slit, but measure spin angle delay to the measurement star when that same star slit is also assigned to Gate B (i.e., ψ^* ψ^* or ψ_2^* ψ_2^*), by performing Steps (1) through (8) of the sequence in Table 4.3.2.4-5 (if not already performed). (b) Calculate $EL_{STAR} = (\Delta)$ [(AZELSAME) - (AZELDIFF)]. • If the SRR slit was the PSI* slit, then $\Delta = +1$. • If the SRR slit was the PSI2* slit, then $\Delta = -1$. (c) Use Figure 3.3.2.2-4A or Figure 3.3.2.2-4B, where $EL_{STAR} \equiv$ PSI* to PSI2* spin angle delay in degrees, to obtain the elevation angle of the measurement star w.r.t. the spacecraft spin axis.		
(10)	The actual roll (azimuth) angle of the measurement star, measured w.r.t. the SRR star in the spacecraft spin plane and in the direction of positive spacecraft spin, may be determined as follows: (a) Use the known elevation angle of the SRR star to find, from (Figure 3.3.2.2-3A or -3B, if the PSI* slit was used as the SRR slit) or (Figure 3.3.2.2-3C or -3D, if the PSI2* slit was used as the SRR slit), the corresponding position pulse delay angle. Declare this value = AZSRR. (b) Calculate: Roll (Azimuth) Angle of the Measurement Star = AZ_{STAR} = AZELDIFF + AZSRR + AZBIAS + EL_{STAR} ° where AZBIAS = value obtained from Remarks column.		SRR: Corresponding value for AZBIAS (from Table 3.3.2.4-3): PSI*: -(B) PSI2*: (B)

TABLE 4.3.2.4-6 (Continued)

COMMAND MNEMONIC OR STEP #	COMMAND OR ACTION	REMARKS & VERIFICATION	
		TELEMETRY MNEMONIC	REMARKS & TM DATA
(10) (Continued)	(c) Optionally, the calculation may be made as follows: AZ_{STAR} = AZELSAME + AZSRR + ELMEAS, where ELMEAS = the position pulse delay angle in degrees from the appropriate one of Figures 3.3.2.2-3A through -3D that corresponds to the measurement star elevation angle found in Step (9) (c), above.		

Revision

4.3.3 Roll References. The feasible combinations for establishment or transfer of Selected Roll Reference are shown in Matrix format in Table 4.3.3-1. A specific step-by-step command sequence, that includes corresponding telemetry verification, for each SRR transfer combination is identified by Table number in each cell of this matrix. All of these tables are included in this Section 4.3.3.

TABLE 4.3.3-1

ESTABLISHMENT OR TRANSFER OF SELECTED ROLI REFERENCE (SRR)

To SRR ▶ / From SRR ▼	PSI	PSI*	PSI2*	Simulated SRR
Nothing (No SRR established, or Loss of SRR)	Sun Acquisition: Perform the sequence in Table 4.3.3-3.	Star Acquisition (PSI*): Perform the sequence in Table 4.3.3-4, selecting PSI* where pertinent.	Star Acquisition (PSI2*): Perform the sequence in Table 4.3.3-4, selecting PSI2* where pertinent.	Refer to Table 4.3.3-5.
PSI	Transfer to a Different Sensor Input Perform the sequence in Table 4.3.3-6.	① Direct transfer from Sun to Star: Perform the sequence in Table 4.3.3-7, selecting PSI* where pertinent. ② Two-Step Transfer, using Simulated SRR temporarily: Perform the sequence in Table 4.3.3-8, selecting PSI* where pertinent.	① General: Perform the sequence in Table 4.3.3-7, selecting PSI2* where pertinent. ② Two-Step Transfer, Using Simulated SRR Temporarily: Perform the sequence in Table 4.3.3-8, selecting PSI2* where pertinent.	Perform the sequence in Table 4.3.3-9.
PSI*	Perform the sequence in Table 4.3.3-10.	Transfer to a Different Star: Perform the sequence in Table 4.3.3-11, selecting PSI* where pertinent.	Perform the sequence in Table 4.3.3-11, selecting PSI2* where pertinent.	Perform the sequence in Table 4.3.3-9.
PSI2*	Perform the sequence in Table 4.3.3-10.	Perform the sequence in Table 4.3.3-11, selecting PSI* where pertinent.	Transfer to a Different Star: Perform the sequence in Table 4.3.3-11, selecting PSI2* where pertinent.	Perform the sequence in Table 4.3.3-9.
Simulated SRR	Perform the sequence in Table 4.3.3-12.	Perform the sequence in Table 4.3.3-13, selecting PSI* where pertinent.	Perform the sequence in Table 4.3.3-13, selecting PSI2* where pertinent.	

TABLE 4.3.3-2

ASSUMED PERTINENT SPACECRAFT INITIAL CONFIGURATION

JUST PRIOR TO TRANSFER OF SELECTED ROLL REFERENCE (SRR)

Equipment		ψ
Earth Station(s) in Use:		26 M.
Communications S/S:		Same states as listed in first column of Table 2.3.1-1.
Controls S/S: Star Sensor Channels.		ψ^* & ψ_2^* both ON.
Sun Sensors		Mid-Range selected.
Attitude Data Processor	ON/OFF State	ADP 1 ON
	Measurement Select	A & B Channels in any configuration.
	Jet Control	(1) All Jets Disabled. (2) Spin Rate Detector Enabled. (3) All other choices: DNA.
	ADP Mode Select	(1) SRR Adv. Inhibited. (2) Sun Gate Enabled. (3) Normal selected; SRR or simulated SRR selected, as applicable. Sun or Star selected, as applicable. (4) Star Gates A & B: Channel 1 (ψ^*) or Channel 2 (ψ_2^*) - as applicable, is selected. (5) PLL Loss of Lock enabled. (6) PLL Spin Range of 4.0 to 8.85 rpm selected.
	ADP Configure Command 3 Through 11.	(1) Jet Countdown: All zeros. (2) ACS Angle: Delay magnitude DNA. (3) Roll Index Delay magnitude as required. (4) PLL Spin Period magnitude - Same as prevailing spin period. (5) JCE Buffer Output disabled.
	Spin Rate	5 rpm.
Despin Control Electronics (DCE)	ON/OFF State	1 ON; 2 OFF.
	Motor Drive	1 ON/2 OFF.
	DCE Configure Commands 0 Through 5.	(1) AZ Pointing Slew: All zeros. (2) HGA Elevation Positioner Slew: All zeros (3) Azimuth Position: As required. (4) HGA electrical boresight despun and pointed to Earth Station. (5) HGA elevation positioner OFF. (6) RAM & NADIR Reset (Not used).

NOTE: ψ_2 is not available as an SRR

TABLE 4.3.3-3
SUN ACQUISITION

COMMAND MNEMONIC OR STEP #	COMMAND OR ACTION	REMARKS & VERIFICATION	
		TELEMETRY MNEMONIC	REMARKS & TM DATA
	• Initial conditions are essentially: Both ADPs OFF, both DCEs OFF, both star sensor channels ON, spin rate at 15 rpm, and sun L.O.S. within the field-of-view of the mid-range sun sensor.		
(1)	• Turn ON ADP #1:		
ADP19 or ADPA9	ADP1 ON	AADP1S	ADP1 ON/OFF = 1 (ON)
		PBUSLI	S/C Loads current increases 250 ma.
		PLIMTI	Bus Voltage Limiter Current likely decreases same amount that PBUSLI has increased.
			• The ADP will initialize in a random manner except for thruster control functions. (Reference Para. 3.3.2.3.2). These will indicate as:
		AJMAGC	JCE Countdown (12 bits) = all zeros.
		AJCEFS	JCE Fire Enable Status = 0. (Disabled).
		AJCE1S	JCE 1 Buffer Output Status = 0 (Disabled).
			Jet Interlock Command Latch (NOT Telemetered) is in the reset (disabled) state.
(2)	• Configure the ADP:		
(2A) ATQ02 (for ADP1 Only) (R1 thru A7 = D; SRD = E)	ADP Configure - Jet Control (All jets disabled, Spin rate detector enabled. Remainder are "don't cares", but are set to: Pulse Fire, Pulse Count, 128 ms, normal fire all selected).	AJETSS	Jet Select Bits = All zero (All seven).
		AJETMS (word 2, Bit 6)	Continuous/pulse Fire select = 0 (Pulse)

TABLE 4.3.3-3. (Continued)

COMMAND MNEMONIC OR STEP #	COMMAND OR ACTION	REMARKS & VERIFICATION	
		TELEMETRY MNEMONIC	REMARKS & TM DATA
2A (Cont'd)		AMAGCS	Pulse/Time Count Select = 1 (Pulse)
		APULLS	Pulse Width Select 512/128 ms = 0 (128 ms).
		ASDETS	Spin Rate Detector Enable/Inhibit = 1 (Enabled).
		AJETMS (Word 3, Bit 0)	Normal/Alternate Fire Mode = 1 (Normal).
2B ATQ03 (for ADP1 Only) (SR8;PLD; SGA1; SGB1; SSRM; REFSUN; STN;SUGD; SRANRM).	ADP Configure - ADP Mode Select (PLL Spin Range of 8.0 to 17.7 rpm selected; PLL Loss of Lock inhibited; Star Gate A and Star Gate B are "don't cares," but Channel 1 is arbitrarily selected; Mid-range sun sensor selected; Sun selected; SRR selected; (Star) Normal selected; Sun Gate disabled; SRR advance inhibited.)	A*GASS	Star Gate A = 1 (PSI*)
		A*GBSS	Star Gate B = 1 (PSI*)
		ASUNSS	Sun Sensor Select = 00 (Midrange Selected).
		ASRRMS	Sun/Star Select = 1 (Sun)
			SRR/Simulated SRR Select = 1 (SRR)
		A*ACQS	Star Acquisition/Normal = 0 (normal)
		ASPINS	PLL Spin Range = 10 (8.0 - 17.7 rpm)
		ALOLES	PLL Loss of Lock = 0 (Inhibited)
		ASUNGS	Sun Gate = 0 (Disabled)
		ADVANS	SRR Advance = 0 (Not Advanced)
		ALOCKS	PLL Loss of Lock = 1 (In-Lock) - This is not a true indicator of lock status when PLL Loss of Lock has been inhibited.

TABLE 4. 3. 3-3. (Continued)

COMMAND MNEMONIC OR STEP #	COMMAND OR ACTION	REMARKS & VERIFICATION	
		TELEMETRY MNEMONIC	REMARKS & TM DATA
(2C) ATQ05 (For ADP1 Only) ACSANG	ADP Configure - Roll Index Delay Magnitude (Zero Delay).	ARIPAD	Roll Index Delay Magnitude (10 bits) = All Zeros.
(2D) ATQ01 (For ADP1 Only) (MEA1 = RIP, MEA2 = RIP)	ADP Configure - Measurement Select (SRR to RIP for both measurements A & B).	AM1ADS	Measurement A Address = 010 (SRR to RIP).
		AM2ADS	Measurement B Address = 010 (SRR to RIP).
		ATTM1Z or ATTM2Z (depending on TM format selected).	Attitude Measurement: During Sun Acquisition, this measurement of SRR to RIP (actually Sun to F_s) will behave like a damped sine wave (see Figure 3. 3. 3. 1-2). Once the PLL is in Lock, this measurement provides the steady state phase bias between the SRR and F_s. Average at least 10 samples of SRR to RIP readings and record for later use whenever at the prevailing spin rate (This bias is spin rate dependent.)
(3)	• After the SRR to RIP readings have stabilized, indicating sun acquisition has occurred (F_s pulse is centered azimuthally in Gate A), proceed to establish the normal operating mode:		
(3A) ATQ01 (For ADP1 Only) (MEA1 = SRR, MEA2 = S2S)	ADP Configure - Measurement Select (SRR to SRR, and SRR to PSI2).	AM1ADS	Measurement A Address = 000 (SRR to SRR)
		AM2ADS	Measurement B Address = 101 (SRR to PSI2).
		ATTM1Z or ATTM2Z	Actual spin rate and quick look S/C attitude w. r. t. the sun l. o. s. can be determined with these measurements.

TABLE 4.3.3-3. (Continued)

COMMAND MNEMONIC OR STEP #	COMMAND OR ACTION	REMARKS & VERIFICATION: TELEMETRY MNEMONIC	REMARKS & VERIFICATION: REMARKS & TM DATA
(3E)	• Set simulated SRR period to match actual spin period:		• Simulated SRR period should always be kept within ±1 bit (±0.25 msec) of the actual spin period in order for the SRR to stay within Gate A for 16 minutes at S/C spin rate of 15 rpm (see Para. 3.3.2.3.4 and 3.3.3.2).
ATQ07 & ATQ08, OR ATQ78 (For ADP1 Only) SIMSPNPD	ADP Configure - PLL Spin Period Magnitude (Set to actual spin period magnitude as seen via ATTM1Z).	ASIMSZ	PLL Spin Period Magnitude (16 bits) = Same as telemetered via ATTM1Z for SRR to SRR measurement.
ATQ03 (For ADP1 Only) (PL = E, SUG = E).	ADP Configure - ADP Mode Select (PLL Loss of Lock Enabled, Sun Gate Enabled).	ALOLES	PLL Loss of Lock = 1 (Enabled)
		ASUNGC	Sun Gate = 1 (Enabled)
(4)	• A precise attitude determination should be made before attempting to use the despun HGA for two-way communications (See Para. 4.3.2 & 4.3.4).		
(5)	• The science reference signals (RIP, RAMVPD, & NADIR VPD) would normally be used after a precise attitude determination has been made, and the RAM and NADIR angles have been calculated.		

TABLE 4.3.3-4
STAR ACQUISITION
(SUN REFERENCE NOT ATTAINABLE)

COMMAND MNEMONIC OR STEP #	COMMAND OR ACTION	REMARKS & VERIFICATION	
		TELEMETRY MNEMONIC	REMARKS & TM DATA
	• This sequence is the same as for Table 4.3.3-3, Sun Acquisition, with the following changes:		
	• The preferred sequence for establishing a star as the SRR is to first select the sun as the SRR and, after a star mapping sequence, transfer to a desired star as the SRR. This sequence assumes that due to S/C attitude or solar eclipse or equipment malfunction, that the sun cannot be used as the intermediate SRR. Regardless of its probability of occurrence, this sequence is included in the matrix of Roll Reference sequences for completeness.		
(1) & (2A)	• Perform Steps (1) and (2A) of Table 4.3.3-3; then configure to acquire the brightest available star (a star map cannot be made in advance because it requires an established SRR. An SRR has not yet been established at this point in the sequence by definition):		
(2B1)	• Configure PSI* (or PSI2*) channel for acquisition of the brightest available star:		
STRQ1 OR STRQA (S2G=4 OR 5, S1G=4 OR 5, S2BW=H, S1BW=H).	Star Sensor Threshold and Bandpass Select (PSI* or PSI2*) selected, Threshold setting 4 or 5 generally selected (see "Remarks" column), and 19 Hz Bandpass selected).		• The threshold setting should be selected to minimize false alarm rate and maximize the probability of detection. Knowing the approximate S/C attitude & ephemeris, the brightest available star can be determined. The silicon magnitude of this star can be determined from Table 3.3.3.3-3. The corresponding desired threshold setting can be determined from Figure 3.3.3.3-1 (Desired threshold setting vs. silicon magnitude). (Continued)

TABLE 4.3.3-4 (Continued)

COMMAND MNEMONIC OR STEP #	COMMAND OR ACTION	REMARKS & VERIFICATION	
		TELEMETRY MNEMONIC	REMARKS & TM DATA
(2B1) (Continued)			• If PSI* has been selected:
		A*1THS	PSI* Threshold Setting (Analog) = State 4 or 5
		A*1BPS	PSI* Bandpass State = 1 (19 Hz).
			• If PSI2* has been selected:
		A*2THS	PSI2* Threshold Setting (Analog) = State 4 or 5.
		A*2BPS	PSI2* Bandpass State = 1 (19 Hz)
(2B2) ATQ#3 (For ADP1 Only) (SR8;PLD; (SGA1 & SGB1) OR (SGA2 & SGB2); SSRM; REFSTR; STA;SUGD; SRAADV).	ADP Configure - ADP Mode Select. (PLL Spin range of 8.0 to 17.7 rpm selected; PLL Loss of Lock inhibited; Star Gate A and Star Gate B - both Channel 1 (for PSI* channel usage) OR both Channel 2 (for PSI2* channel usage); Sun sensor select is a "don't care," but midrange is arbitrarily selected; Star selected; Star Acquisition selected; Sun Gate disabled; SRR Advance).	A*GASS	Star Gate A = 1 (PSI*) OR = 0 (PSI2*), as selected.
		A*GBSS	Star Gate B = 1 OR 0 - Same choice as made for A*GASS.
		ASUNSS	Sun Sensor Select = 00 (Mid-range selected).
		ASRRMS	Sun/Star Select = 0 (Star)
			SRR/Simulated SRR = 1 (SRR)
		A*ACQS	Star Acquisition/Normal = 1 (Star Acquisition) • This causes Gate A to be 45° wide.
		ASPINS	PLL Spin Range = 10 (8.0 - 17.7 rpm).
		ALOLES	PLL Loss of Lock = 0 (Inhibited)
		ASUNGS	Sun Gate = 0 (Disabled).
		ADVANS	SRR Advance = 1 (Advanced)
		ALOCKS	PLL Loss of Lock = 1 (In-lock) - This not a true indicator of Lock status when PLL loss of lock has been inhibited.

Table 4. 3. 3-4. (Continued)

COMMAND MNEMONIC OR STEP #	COMMAND OR ACTION	REMARKS & VERIFICATION	
		TELEMETRY MNEMONIC	REMARKS & TM DATA
(2C) & (2D)	• Perform Steps (2C) and (2D) of Table 4. 3. 3-3. Substitute the word 'Star' for the word 'Sun' in the remarks column.		
(3)	• After the telemetry listed at the right have stabilized, indicating Star acquisition has occurred (F_S pulse is centered azimuthally in Gate A), proceed to measurement selection:	ATTM1Z OR ATTM2Z (Depending on TM format selected).	SRR to RIP (Telemetered as both Measurement A and Measurement B) has stabilized.
		A*1BRM	PSI* Brightness } The selected star channel output has stabilized in brightness
		A*2BRM	PSI2* Brightness }
(3A) ATQ01 (for ADP 1 only) (MEA1 = SRR, MEA2 = SRR)	ADP Configure - Measurement Select (SRR to SRR, and SRR to SRR).	AM1ADS	Measurement A Address = 000 (SRR to SRR).
		AM2ADS	Measurement B Address = 000 (SRR to SRR).
		ATTM1Z OR ATTM2Z	Actual spin rate can be determined with Measurement A&B.
(3B)	• Perform Step (3B) of Table 4. 3. 3-3.		

Table 4. 3. 3-4. (Continued)

COMMAND MNEMONIC OR STEP #	COMMAND OR ACTION	REMARKS & VERIFICATION	
		TELEMETRY MNEMONIC	REMARKS & TM DATA
(3C)	• Configure for normal operating mode after Star Acquisition:		
ATQØ3 (PLE; STN; SUGD; SRANRM)	ADP Configure - ADP Mode Select (PLL Loss of Lock Enabled; (Star) Normal selected; Sun Gate state "don't care," but arbitrarily left in disabled state; SRR Advance inhibited).	A*ACQS	Star Acquisition/Normal - 0 (Normal) • This causes Gate A to retract to a width of 11.25°.
		ALOLES	PLL Loss of Lock = 1 (Enabled)
		ADVANS	SRR Advance = 0 (Not Advanced)
(4)	• A precise attitude determination should be made before attempting to use the despun HGA for two-way communications (See Para. 4. 3. 2 and 4. 3. 4). Since the sun is not available, and presumably the star in use as the SRR is not known, a star mapping should be done using Gate B (See Figure 3. 3. 3. 3-1 for the mapping sequence) and all available stars positions used for the attitude determination. If a star other than the one presently functioning as the SRR is the preferred SRR, then a star-to-star transfer should be made after the star mapping (See Table 4. 3. 3-11), and before the despun HGA and science Reference signals are established.		

TABLE 4.3.3-5

SIMULATED SRR "ACQUISITION"

Given no established SRR and the initial conditions listed in Table 4.3.3-2, inability to perform either a sun acquisition or a star acquisition implies multiple spacecraft failures or an abnormal spacecraft attitude with the sun l.o.s. along the spin axis and no stars within the star sensor field-of-view. Also, if one were driven to attempt this sequence, one would have to make the best estimate of the present actual spin period based on previous knowledge in order to make the simulated spin period acceptably realistic. Attitude is indeterminate except via the earth L.O.S.

The actual spin period could be approximated in real time by measuring the period in the ground received signal strength signature emanating from a spacecraft omni antenna. The measurement could be refined, by transferring the downlink to the HGA that is approximately despun and the downlink is detectable, via use of the above measured spin period as the simulated SRR period. A slow drift of the HGA transmitted signal will indicate the error between true and simulated spin periods.

HGA elevation position corresponding to maximum ground received signal strength will provide the remaining information for determining spacecraft w.r.t. the earth L.O.S.

TABLE 4.3.3-6
SUN TO SUN (DIFFERENT SENSOR INPUT) TRANSFER OF SRR

COMMAND MNEMONIC OR STEP #	COMMAND OR ACTION	REMARKS & VERIFICATION	
		TELEMETRY MNEMONIC	REMARKS & TM DATA
	• Initial conditions are as shown in Table 4.3.3-2 (essentially: ADP1 ON; DCE1 ON; HGA despun, pointed to earth l.o.s., and in use for Uplink and Downlink; PSI is the SRR; and sun sensor mid-range selected.		
	• Without speculating about the probability of occurrence of this sequence, and excluding (1) intervening precessions that may require it, and (2) the geometric limits of usage of the HGA in the face of a precession from an ecliptic normal spin axis attitude, the sequence is presented here for completeness:		
(1)	• Choose any detectable star for the purpose of bias measurement. • Measure and record the (SRR to Gated Star B) ATTM to this "bias measurement" star - converted to spin angle delay via (SRR to SRR) ATTM. Also, record the intensity of this "bias measurement" star for aid in its identification:		
(2)	• Configure either PSI* or PSI2* channel for measurement of the "bias measurement" star:		
STRQ1 or STRQA	Star sensor Threshold and Bandpass Select (PSI* or PSI2* Selected, Threshold setting and bandpass selected for acceptable probability of detection of "bias measurement" star).	A*1THS	PSI* Threshold Setting — At commanded settings, according to star channel selected
		A*1BPS	PSI* Bandpass state
		A*2THS	PSI2* Threshold Setting
		A*2BPS	PSI2* Bandpass State

TABLE 4.3.3-6. (Continued)

COMMAND MNEMONIC OR STEP #	COMMAND OR ACTION	REMARKS & VERIFICATION	
		TELEMETRY MNEMONIC	REMARKS & TM DATA
(3)	• Locate Gate B over the "bias measurement" star:		
ATQ05 (For ADP 1 Only) (ACSANG)	ADP Configure - ACS Angle Delay Magnitude Calculate $\theta = SEP_{SUNSTAR} - BIAS_{SUNSTAR} - 0.352^{o} - 5.625^{o}$, where: • $SEP_{SUNSTAR}$ = Azimuthal separation angle measured in the plane normal to the S/C spin axis from the sun l.o.s. to the "bias measurement" star (measured in the direction of S/C spin); and: • $BIAS_{SUNSTAR}$ = value obtained from table in "REMARKS" column.) • If θ is positive, commanded magnitude = θ. • If θ is negative, commanded magnitude = 360^{o} θ.	AACSAD	ACS Angle Delay Magnitude = Value Commanded.
			• This action here, plus Star Gate B selection in the next step, will position the "bias measurement" star to be in the center of Gate B.
			• Actual delay is 0.352^{o} larger than the commanded value.
			Gate B Star Channel: / Corresponding value (from Tables 3.3.2.4-2 and 3.3.2.4-3) for $BIAS_{SUNSTAR}$:
			PSI* / -(1)-(2)-(3)+(A)+(B)+(C).
			PSI2* / -(1)-(2)-(3)+(A)+(C).

TABLE 4.3.3-6. (Continued)

COMMAND MNEMONIC OR STEP #	COMMAND OR ACTION	REMARKS & VERIFICATION: TELEMETRY MNEMONIC	REMARKS & VERIFICATION: REMARKS & TM DATA
(4)	• Select Star Gate B for the star channel chosen in Step (2):		
ATQ03 (for ADP 1 Only)	ADP Configure - ADP Mode Select (Star Gate B = 1 or 2) (For PSI* channel or PSI2* channel, respectively, as selected in Step 2)	A*GBSS	Star Gate B = 1 (PSI*) or = 0 (PSI2*), as commanded.
	• If the prevailing HGA azimuthal position commanded and TM values are known and recorded, note them as Step (6) recorded values and proceed directly to Step (7). Otherwise, perform Steps (5) and (6).	A*1BRM A*2BRM	PSI* Brightness PSI2* Brightness According to star channel selected, should have stabilized to a value recognizable as that of the "bias measurement" star.
(5)	• Configure for and record present ATTM measurements for HGA Azimuthal position:		
ATQ01 (for ADP 1 Only) (MEA1 = SRR, MEA2 = MIP)	ADP Configure - Measurement select. ((SRR to SRR) and (SRR to MIP)).	ATTM1Z OR ATTM2Z	Attitude Measurements: • Record (SRR to SRR) and (SRR to MIP). • Translate this, via knowledge of last azimuthal commands to DCE, to the equivalent azimuthal angle needed for DCE configure - Azimuth position command.
(6)	• Verify correctness of equivalent azimuthal position command via ground received signal strength.		• Commanded angle should be approximately 1.41° (pre-launch value for fixed set of environmental conditions), less than SRR to MIP angle calculated from TM. (Ref. Table 3.3.2.4-2)
DCEQ5 (For DCE1 Only)	DCE Configure - Azimuth Position (Approximately 1.41° less than SRR to MIP angle calculated from TM).	ATTM1Z OR ATTM2Z	Attitude Measurements: SRR to MIP should stay constant and ground received signal strength should stay constant. If not, recalculate necessary command angle to return PLL Ref to MIP angle as measured in Step (5). Record new Δ between commanded and TM angles for post-transfer use. Then repeat this step with new Δ.

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TABLE 4.3.3-6. (Continued)

COMMAND MNEMONIC OR STEP #	COMMAND OR ACTION	REMARKS & VERIFICATION	
		TELEMETRY MNEMONIC	REMARKS & TM DATA
(7)	• Configure for and record present spin angle delay to "bias measurement" star:		
ATQ01 (For ADP1 Only) (MEA1 = SRR, MEA2 = GSB)	ADP Configure - Measurement Select (SRR to SRR) and (SRR to Gated Star B):	ATTM1Z OR ATTM2Z	Attitude Measurements • Calculate spin angle delay to "bias measurement" star. Call it "PRE-MEAS"
(8)	• Disable F_S Reference to DCE:		• To prevent antenna pointing transients during SRR transfer (Refer to Para. 3.3.2.4).
DCEQ6 (For DCE1 Only) (HGA Phase = DA)	DCE Configure - Mode Command (Phase Reset Disable)	ADSPNS	Despin Phase Reset = 0 (Disabled)
			• HGA Azimuthal on-board closed loop control via the DCE has now been disconnected from using the ADP F_S pulse as a spin reference pulse. Despin control is maintained by using the 2^{12} F_S pulses from the ADP, the 2^7 F_Z pulses from the BAPTA; and therefore, effectively, the last remembered azimutnal location of the F_S pulse.

TABLE 4.3.3-6. (Continued)

COMMAND MNEMONIC OR STEP #	COMMAND OR ACTION	REMARKS & VERIFICATION	
		TELEMETRY MNEMONIC	REMARKS & TM DATA
(9)	• Transfer the SRR to the extended range sun sensor of choice:		
ATQ03 (For ADP1 Only) (SSR = U OR L)	ADP Configure - ADP Mode Select sun sensor extended range upper or lower selected, all other selections not changed)		
		ASUNSS	Sun Sensor Select = 01 (Extended Range Lower) or = 10 (Extended Range Upper), as commanded.
		ATTM1Z OR ATTM2Z	Attitude Measurements. • Calculate spin angle delay to "bias measurement" star. Call it "POST-MEAS."
			All Other TM as before.
(10)	• Calculate required new azimuth angle for HGA azimuth position: "AZ" = (Last azimuth command value from Step (6)) + "CHANGE", where: "CHANGE" = ("PREMEAS" (from Step (7)) - ("POST-MEAS" (from Step (9)).		

TABLE 4.3.3-6. (Continued)

COMMAND MNEMONIC OR STEP *	COMMAND OR ACTION	REMARKS & VERIFICATION	
		TELEMETRY MNEMONIC	REMARKS & TM DATA
(11)	• Configure for and record present ATTM measurement for HGA azimuth position and RIP delay:		
ATQ#1 (For ADP1 Only) (MEA1 = MIP, MEA2 = RIP)	ADP Configure - Measurement Select (SRR to MIP) and (SRR to RIP).	ATTM1Z OR ATTM2Z	Attitude Measurements: • Record present readings. Convert to spin angle delays via use of last known spin period.
		ARIPAD	Roll Index Delay Magnitude: • Record present value (This is the last commanded value) for later use.
(12)	• Insert new HGA Azimuth angle into DCE:		
DCEQ5 (For DCE1 Only)	DCE Configure - Azimuth Position (Use "AZ" calculated in Step (10)).		• Cannot verify until DCE Phase Reset is enabled.
(13)	• Enable F_S Reference to DCE:		
DCEQ6 (For DCE1 Only) (HGA PHASE = EN)	DCE Configure - Mode Command (Phase Reset Enable).	ADSPNS	Despin Phase Reset = 1 (Enabled)
			• Ground received signal strength via HGA does not change, indicating HGA is not moving.
		ATTM1Z OR ATTM2Z	SRR to MIP - Step change to be equal to "change," as defined in Step (10)
			• If HGA does move slightly, it will have to be repointed to earth l.o.s. - Repeat Step (12), using this new data to correct the azimuth position command

TABLE 4.3.3-6. (Continued)

COMMAND MNEMONIC OR STEP #	COMMAND OR ACTION	REMARKS & VERIFICATION	
		TELEMETRY MNEMONIC	REMARKS & TM DATA
(14)	Readjust RIP delay w.r.t. new SRR:		
ATQ06 (For ADP1 Only) RIPANG	ADP Configure - Roll Index Delay Magnitude (Magnitude = (ARIPAD value from Step (11)) + "CHANGE" (From Step (10)).	ARIPAD	Roll Index Delay Magnitude = Commanded Value
		ATTM1Z OR ATTM2Z	SRR to RIP - Step function change by a magnitude = "CHANGE."
			• RIP, RAM VPD, & NADIR VPD should now be occurring spatially as prior to the SRR transfer.

TABLE 4.3.3-7.
SUN TO STAR TRANSFER OF SRR

Command Mnemonic Or Step #	Command or Action	Remarks & Verification	
		Telemetry Mnemonic	Remarks & TM Data
	• Initial conditions are as shown in Table 4.3.3-2. (Essentially: ADP1 ON; DCE 1 ON; HGA Despun, pointed to earth l.o.s., and in use for uplink and downlink; and PSI is the SRR).		
			• This sequence will be used normally on-orbit, prior to entering a region of eclipse of the sun by Venus. The SRR candidate star that will normally be used during this point in the mission is Rigil Kentaurus. The actual transfer will occur via execution of commands in the command memory. Since the command memory is limited in size, science commands and required commands for SRR transfer must be budgeted. There may not be sufficient space in the command memory to accommodate all the commands listed in this sun-to-star transfer sequence Table 4.3.3-7. Some of the commands in this sequence should be executed in real-time before entering the eclipse region in order to save command memory space. These are identified by "(RT)". In order to update bias values, perform the entire sequence, preferably during apoapsis. Exclude "(RT)" commands and exclude "(BIAS)" commands from the stored command sequence.

Table 4.3.3-7. (Continued)

Command Mnemonic Or Step #	Command or Action	Remarks & Verification	
		Telemetry Mnemonic	Remarks & TM Data
(1) (BIAS)	• The S/C attitude should be well known on a continuous basis. However, an updating of the star map may be required periodically; and, if needed, should be performed prior to transfer to a star SRR - preferably during apoapsis.		
	• Perform a complete or limited star map per the sequence in Figure 3.3.3.3-2, as required. As a minimum, map the immediate azimuthal vicinity (10^{o}, in 2^{o} segments) of the star that is the SRR candidate. • As a minimum, record (SRR to SRR) and concurrently, the (SRR to Gated Star B) ATTM values, (1) using PSI* channel for Star Gate B with the SRR candidate star. and (2) using PSI2* channel for Star Gate B with the SRR candidate star. • Translate above to spin angle delays, using (SRR to SRR) measurement(s).		
(2) (BIAS) (RT)	• Configure PSI* (or PSI2*) channel for use as the SRR channel for the SRR candidate star:		
STRQ1 OR STRQA	Star Sensor Threshold and Bandpass Select (PSI* (or PSI2*) selected, Threshold Setting and Bandpass Selected for acceptable probability of Detection of SRR Candidate Star).	A*1THS	PSI* Threshold Setting — At commanded settings, according to Star Channel Selected.
		A*1BPS	PSI* Bandpass State
		A*2THS	PSI2* Threshold Setting
		A*2BPS	PSI2* Bandpass State

TABLE 4.3.3-7. (Continued)

COMMAND MNEMONIC OR STEP #	COMMAND OR ACTION	REMARKS & VERIFICATION	
		TELEMETRY MNEMONIC	REMARKS & TM DATA
(3)	• Locate Gate B over the SRR candidate star:		
ATQ05 (For ADP 1 Only) (ACSANG)	ADP Configure - ACS Angle Delay Magnitude Calculate $\theta = SEP_{SUNSTAR} - BIAS_{SUNSTAR} - 0.352^{o} - 5.625^{o}$, where: • $SEP_{SUNSTAR}$ = Azimuthal separation angle measured in the plane normal to the S/C spin axis from the sun l.o.s. to the "bias measurement" star (measured in the direction of S/C spin); and: • $BIAS_{SUNSTAR}$ = value obtained from table in "REMARKS" column.) • If θ is positive, commanded magnitude = θ. • If θ is negative, commanded magnitude = 360^{o} θ.	AACSAD	ACS Angle Delay Magnitude = Value Commanded.
			• This action here, plus Star Gate B selection in the next step, will position the "bias measurement" star to be in the center of Gate B.
			• Actual delay is 0.352^{o} larger than the commanded value.
		Gate	Gate B Star Channel: / Corresponding value (from Tables 3.3.2.4-2 and 3.3.2.4-3) for $BIAS_{SUNSTAR}$.
			PSI* / -(1)-(2)-(3)+(A)+(B)+(C).
			PSI2* / -(1)-(2)-(3)+(A)+(C)

TABLE 4.3.3-7. (Continued)

Command Mnemonic Or Step #	Command or Action	Remarks & Verification	
		Telemetry Mnemonic	Remarks & TM Data
④ (RT)	• Select Star Gates A and B for the desired SRR star channel (selected in Step ②) :		
ATC03 (For ADP1 Only)	ADP Configure - ADP Mode Select (Star Gates A and B both channel 1 (for PSI* channel usage) or both channel 2 (for PSI2* channel usage)).	A*GASS	Star Gate A = 1 (PSI*) OR = 0 (PSI2*), as commanded
		A*GBSS	Star Gate B = 1 or = 0 - Same choice as made for A*GASS.
			All other ADP Mode states should be unchanged.
⑤ (BIAS)	If the prevailing HGA azimuthal position commanded & TM values are known & recorded, note them as Step ⑥ recorded values & proceed directly to Step ⑦. Otherwise, perform Steps ⑤ & ⑥		
	(Continued)		

TABLE 4.3.3-7 (Continued)

Command Mnemonic Or Step #	Command or Action	Remarks & Verification	
		Telemetry Mnemonic	Remarks & TM Data
(5) (Continued)	• Configure for and record present ATTM measurements for HGA Azimuthal position:		
ATQ01 (For ADP1 Only) (MEA1 = SRR, MEA2 = MIP)	ADP Configure - Measurement select (SRR to SRR) and (SRR to MIP)).	ATTM1Z OR ATTM2Z	Attitude Measurements: • Record (SRR to SRR) and(SRR to MIP) • Translate this via knowledge of last azimuthal commands to DCE, to the equivalent azimuthal angle needed for DCE Configure - Azimuthal Position Command.
(6) (BIAS)	• Verify correctness of equivalent azimuthal position command:		• Commanded angle should be approximately 1.41° (pre-launch value for fixed set of environmental conditions) less than SRR to MIP angle calculated from TM.
DCEQ5 (For DCE1 Only)	• DCE Configure - Azimuth Position (Approximately 1.41° less than SRR to MIP angle calculated from TM.)	ATTM1Z OR ATTM2Z	Attitude Measurements: SRR to MIP should stay constant and ground received signal strength should stay constant. If not, recalculate necessary command angle to return SRR to MIP angle as measured in Step (5). Record new Δ between commanded and TM angles for post-transfer use. • Then repeat this step with new Δ.
(7) (RT)	• Configure for and record all present remaining pertinent ATTM and associated measurements for post-transfer use; then configure finally for measurement during transfer:		

TABLE 4.3.3-7. (Continued)

Command Mnemonic Or Step #	Command or Action	Remarks & Verification	
		Telemetry Mnemonic	Remarks & TM Data
(7) (Continued) (MEA1 = SRR, MEA2 = RIP; then MEA1 = SRR, MEA2 = GSB)	ADP Configure - Measurement Select (SRR to SRR) and (SRR to RIP), terminating with (SRR to Gated Star B) and (SRR to SRR)	ATTM1Z OR ATTM2Z	Attitude Measurement: • Record, when S/C is configured, the following for post-transfer use: (SRR to SRR), and (SRR to RIP) • SRR to Gated Star B should stabilize to less than 0.3° variations.
		A*1BRM	PSI* Brightness — According to Star Channel selected - should have stabilized to value recognizable as that of the SRR candidate star.
		A*2BRM	PSI2* Brightness — (same)
		ARIPAD	RIP Magnitude — Record values for post-transfer Use.
		AACSAD	ACS Angle Delay Magnitude — (same)
		ASIMSZ	PLL Spin Period Magnitude — (same)
		AVPDRD	RAM VPD — Record values for post-transfer use.
		AVPDND	NADIR VPD — (same)
		AELEVD	HGA Elevation — (same)
(8)	o Disable F_S Reference to DCE:		• To prevent antenna positioning transients during SRR transfer (Refer to Para. 3.3.2.4.4).

Table 4.3.3-7. (Continued)

Command Mnemonic Or Step #	Command or Action	Remarks & Verification	
		Telemetry Mnemonic	Remarks & TM Data
⑧(Cont'd) DCEQ6 (For DCE1 Only) (HGA PHASE = DA)	DCE Configure - Mode Command (Phase Reset Disable)	ADSPNS	Despin Phase Reset = 0 (Disabled)
			• HGA Azimuthal onboard closed loop control via the DCE has now been disconnected from using the ADP F_S pulse as a spin reference pulse. Despin control is maintained by using the 2^{12} F_S pulses from the ADP, the 2^7 F_Z pulses from the BAPTA; and therefore, effectively, the last remembered azimuthal location of the F_S pulse.

TABLE 4.3.3-7. (Continued)

COMMAND MNEMONIC OR STEP #	COMMAND OR ACTION	REMARKS & VERIFICATION	
		TELEMETRY MNEMONIC	REMARKS & TM DATA
(9)	• Transfer the SRR to the candidate star:		
ATQ03 (For ADP1 Only) (PLD; REFSTR; SRAADV)	ADP Configure - ADP Mode Select (PLL Loss of Lock Inhibited; Star Selected; SRR Advance Selected).	ASRRMS	Sun/Star Select = 0 (Star)
		ALOLES	PLL Loss of Lock = 0 (Inhibited)
		ADVANS	SRR Advance = 1 (Advanced)
			• Phase Reference of PLL will transfer to Star located in Gate B. • Star will leave Gate B, but stay in Gate A.
		ALOCKS	PLL Loss of Lock = 1 (In-Lock) - This is not a true indicator of Lock status when PLL Loss of Lock has been inhibited.
		A*1BRM	PSI* Brightness — According to channel selected, brightness will drop to zero, indicating star has left Gate B.
		A*2BRM	PSI2* Brightness
		ATTM1Z OR ATTM2Z	Attitude Measurements: • If SRR to Gated Star B is one of the selected measurement channels (e.g., during bias measurement effort near apoapsis), as soon as the star leaves Gate B, this measurement can no longer be made, and ATTM measurements A and B will no longer cycle. The last valid measurement from the alternate selected value (in this case, SRR to SRR will be continuously telemetered for both A and B measurement slots. Refer to Para. 3.3.11.2.

Table 4.3.3-7. (Continued)

Command Mnemonic Or Step #	Command or Action	Remarks & Verification	
		Telemetry Mnemonic	Remarks & TM Data
(10) (BIAS)	• Calculate required azimuth angle for HGA Azimuth position: • D = (Last azimuth command value used in step (6) + "Δ" + TMSEP, where "Δ" & TMSEP are obtained from Table in the "Remarks" column. • If D<360°, then "AZ" = D. • If D≥360° then "AZ" = D-360°.		• "Δ" may be obtained from prelaunch tests or updated during the mission using offline statistical processing of appropriate downlink information.

Star Channel chosen in step (2) :	Corresponding value for TMSEP:
PSI*	Value obtained in (1) of step (1) .
PSI2*	Value obtained in (2) of step (1) .

TABLE 4.3.3-7. (Continued)

COMMAND MNEMONIC OR STEP #	COMMAND OR ACTION	REMARKS & VERIFICATION	
		TELEMETRY MNEMONIC	REMARKS & TM DATA
(11) (BIAS)	• Configure for and record present ATTM measurements for HGA Azimuth position and RIP Delay:		
ATQ01 (For ADP1 Only) (MEA1 = MIP, MEA2 = RIP)	ADP Configure - Measurement Select (SRR to MIP) and (SRR To RIP)	ATTM1Z OR ATTM2Z	Attitude Measurements: • Record present readings. Convert to spin angle delays via use of last known spin period.
		ARIPAD	Roll Index Delay Magnitude - Record present value (this is the last commanded value) for later use - Should be the same value as recorded in step (5).
(12)	• Insert new HGA Azimuth Angle into DCE:		
DCEQ5 (For DCE1 Only)	DCE Configure - Azimuth Position (Use "AZ" calculated in Step (10)).		• Cannot verify until DCE Phase Reset is Enabled.
(13)	• Enable F_S Reference to DCE:		
DCEQ6 (For DCE1 Only) (HGA PHASE = EN)	DCE Configure - Mode Command (Phase Reset Enable)	ADSPNS	Despin Phase Reset = 1 (Enabled)
			• Ground received signal strength via HGA does not change, indicating HGA is not moving.
		ATTM1Z OR ATTM2A	SRR to MIP - Step change to be equal to "CHANGE" as defined in Step (10).
			• If HGA does move slightly, it will have to be repointed to earth l.o.s. Repeat Step (12), using this new data to correct the azimuth position command.

TABLE 4.3.3-7. (Continued)

COMMAND MNEMONIC OR STEP #	COMMAND OR ACTION	REMARKS & VERIFICATION	
		TELEMETRY MNEMONIC	REMARKS & TM DATA
(14)	Readjust RIP Delay w.r.t. new SRR:		
ATQ66 (For ADP1 Only)	ADP Configure - Roll Index Delay Magnitude (• F = (ARIPAD value from Step (11))+ "Δ" + TMSEP (From Step (10) .) • If F <360°, then Magnitude = F. • If F ≥360°, then Magnitude = F-360°.	ARIPAD	Roll Index Delay Magnitude = Commanded Value.
		ATTM1Z OR ATTM2Z	SRR to RIP - Step function change by a magnitude = "Δ" + TMSEP.
			• RIP, RAM VPD and NADIR VPD should now be occurring spatially as prior to the SRR transfer.
(15)	• Establish normal post-transfer mode for ADP:		
ATQ63 (For ADP1 Only) (PLE; SRANRM)	ADP Configure - ADP Mode Select (PLL Loss of Lock Enabled; SRR Advance Reset),	ADVANS	SRR Advance = 0 (Not Advanced)
		ALOLES	PLL Loss of Lock = 1 (Enabled)
		ALOCKS	PLL Loss of Lock - Now a true indicator.
			All Other TM as Before.
(16)	• Configure ATTM for eclipse (or lack of sun availability):		
ATQ61 (For ADP1 Only)	ADP Configure - Measurement Select [(SRR to SRR) and (SRR to Gated Star B)] or [to any other combination that excludes (SRR to PSI2)]	ATTM1Z OR ATTM2Z	Attitude Measurements: As selected.

TABLE 4.3.3-8
SUN-TO-SIMULATED SRR-TO-STAR TRANSFER OF SRR (SEMI-AUTOMATIC)

COMMAND MNEMONIC OR STEP #	COMMAND OR ACTION	REMARKS & VERIFICATION	
		TELEMETRY MNEMONIC	REMARKS & TM DATA
	• Initial conditions are as shown in Table 4.3.3-2. (Essentially: ADPI ON; DCE1 ON; HGA Despun, pointed to earth l.o.s., and in use for uplink and downlink; and PSI is the SRR).		
	• This sequence will be used normally as a two-step transfer of the SRR from Sun to Star, with an intervening period of time when the sun will be lost (entering eclipse), but the SRR candidate star is not yet visible (refer to Para. 3.3.3.7.5). • The stored commands listed in Step (SS3) are used to transfer automatically from simulated SRR to the SRR candidate star, when it is assumed that the S/C may be occulted from view of the earth. • Steps (SS4) and (SS5) are intended to be performed in real-time, prior to occultation.		
(SS1)	• At least several hours in advance of eclipse entry, perform steps (1) through (15) of Table 4.3.3-7 (Sun to Star Transfer of SRR). This will produce up-to-date sensor bias values. Record for later use: (a) The ACS Angle delay magnitude used to locate Star Gate B over the SRR Candidate Star. (From Step (3)) (b) The value of "AZ" (From Step (10)) (c) The Roll Index Delay Magnitude (From Step (14))		
(SS2)	• Return the SRR to the Sun sensor via Steps (1) through (15) of Table 4.3.3-10, (Star to Sun Transfer of SRR). Use the values learned in Step (SS1) of this procedure where advantageous.		
(SS3)	• Load the following sequence into each of the two command memories:		

TABLE 4.3.3-8. (Continued)

COMMAND MNEMONIC OR STEP #	COMMAND OR ACTION	REMARKS & VERIFICATION	
		TELEMETRY MNEMONIC	REMARKS & TM DATA
SS3.1	Store: Time Delay Code (α seconds for CP1; α+5 seconds for CP2)		• Time delay α is chosen to start at the next transition of the TP 4096 second clock bit and terminate two minutes after the SRR candidate star is predicted to become visible following a period of obscuration. • The 5 seconds difference in the time delays loaded into the two CPs is to prevent the two SCLs from addressing the same COM at the same time (e.g., the ADP configure commands) - Refer to Para. 3.6.3.2.5.
SS3.2 DCEQ6 (For DCE1 Only) (HGA PHASE = DA)	Store: DCE Configure - Mode Command (Phase Reset disable; all other states unchanged).		• This disables the F_S reference to the DCE - HGA pointing is no longer under DCE control.
SS3.3	Store: Time Delay Code (6 seconds for CP1; 1 second for CP2).		• Allows at least one second for DCE to stabilize before transferring the SRR ahead in Step SS3.4. • If SS3.2 did not take effect in CP1, then the 6 seconds delay in CP1 allows SS3.2 in CP2 to take effect before SS3.4 in CP1 is executed.
SS3.4 ADP3 (For ADP1 Only) (PLD; REFSTR; SR(ALV)	Store: ADP Configure - ADP Mode Select (PLL loss of lock inhibited; star selected; SRR Advance selected).		• This will transfer the SRR to the Candidate Star, when executed.
SS3.5 DCEQ5 (For DCE1 Only)	Store: DCE Configure - Azimuth Position (Use "AZ" recorded in Step SS1, above).		• This will retain HGA boresight on earth l.o.s. azimuthly w.r.t. the star SRR, but after DCE phase reset is enabled below.

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TABLE 4.3.3-8. (Continued)

COMMAND MNEMONIC OR STEP *	COMMAND OR ACTION	REMARKS & VERIFICATION	
		TELEMETRY MNEMONIC	REMARKS & TM DATA
SS3.6	Store: Time Delay Code (6 Seconds for CP1; 1 second for CP2).		• If SS3.5 did not take effect in CP1, then the 6 seconds delay in CP1 allows SS3.5 in CP2 to take effect before SS3.7 in CP1 is executed.
SS3.7 DCEQ6 (For DCE1 Only)	Store: DCE Configure - Mode Command (Phase Reset Enable)		• This enables the F_S reference to the DCE. • The DCE control of HGA pointing is now restored.
SS3.8 ATQ06 (For ADP1 Only)	Store: ADP Configure - Roll Index Delay Magnitude (Use the magnitude recorded in Step SS1, above).		• This readjusts the RIP delay w.r.t. the new SRR.
SS4	• Before transferring to the simulated SRR, configure the S/C for the later automatic transfer to Star SRR:		
SS4.1	• Configure PSI* (or PSI2*) channel for use as the SRR channel for the SRR candidate star:		
STRQ1 OR STRQA	Star Sensor Threshold and Bandpass Select (PSI* (or PSI2*) selected, Threshold Setting and Bandpass selected for acceptable probability of detection of SRR candidate star).	A*1THS	PSI* Threshold Setting — At commanded settings, according to Star channel selected.
		A*1BPS	PSI* Bandpass State
		A*2THS	PSI2* Thresnold Sett.
		A*2BPS	PSI2* Bandpass State
SS4.2	• Locate Gate B over the SRR Candidate Star:		

TABLE 4.3.3-8. (Continued)

COMMAND MNEMONIC OR STEP *	COMMAND OR ACTION	REMARKS & VERIFICATION	
		TELEMETRY MNEMONIC	REMARKS & TM DATA
SS4.2 (Continued) ATQ05 (For ADP1 Only)	ADP Configure - ACS Angle Delay Magnitude (Magnitude = (From Step SS1 above): (a) - (b) - 4.62°)	AACSAD	ACS Angle Delay Magnitude = value commanded.
			• This action here, plus Star Gate B selection in the next step, will position the leading edge of Gate B to be 5.625° (½ the width of Gate B) in advance of the SRR candidate star azimuthal location.
SS4.3	• Select Star Gates A and B for the desired SRR star channel (Selected in Step SS4.1).		
ATQ03 (For ADP1 Only)	ADP Configure - ADP Mode Select (Star Gates A and B both Channel 1 (for PSI* channel usage) or both Channel 2 (for PSI2* usage)).	A*GASS	Star Gate A = 1 (PSi*) or = 0 (PSI2*, as commanded)
		A*GBSS	Star Gate B = 1 or 0 - same choice as made for A*GASS.
			All other ADP Mode states should be unchanged.
SS5	• When it is required to transfer SRR from the Sun to the simulated SRR, perform Steps 1 through 4 of Table 4.3.3-9, (Sun or Star to Simulated SRR Transfer of SRR). • Automatic execution of the stored sequence, that was loaded into the command memory in Step SS3, will occur later in time.		

TABLE 4.3.3-9
SUN OR STAR TO SIMULATED SRR TRANSFER OF SRR

COMMAND MNEMONIC OR STEP #	COMMAND OR ACTION	REMARKS & VERIFICATION	
		TELEMETRY MNEMONIC	REMARKS & TM DATA
	• Initial conditions are as shown in Table 4.3.3-2. (Essentially: ADP1 ON; DCE1 ON; HGA Despun, pointed to earth l.o.s., and in use for uplink and downlink; and PSI or PSI* or PSI2* is the SRR.)		
	• This sequence will be used normally only if the sun and all reliably detectable stars will not be available imminently as an SRR. • This is a general purpose sequence, and not the specific sequence, to be used as the first step in a two-step transfer from sun to star SRR during on-orbit eclipse entry (refer to Para. 3.3.3.7.5 and to Table 4.3.3-8 (Sun-to-Simulated SRR-to-Star Transfer of SRR (Semi-automatic) for further description.)		
(1)	• Configure for and record present ATTM measurements for HGA Azimuthal position, the spin period, and simulated spin period.		
ATQ#1 (For ADP1 Only) (MEA1 = SRR, MEA2 = MIP)	ADP Configure - Measurement Select. ((SRR to SRR) and (SRR to MIP)).	ATTM1Z OR ATTM2Z	Attitude Measurements: • Record (SRR to SRR) and (SRR to MIP).
		ASIMSZ	PLL Spin Period Magnitude (16 Bits) } Record this simulated spin period magnitude.
(2)	• The S/C spin rate is expected to increase as the S/C passes thru an eclipse and contracts. • Estimate the mean of the present spin period and the expected spin period at eclipse exit.		
(3)	• Change the simulated spin period magnitude to be the mean value calculated in Step (2).		

Table 4.3.3-9. (Continued)

Command Mnemonic or Step #	Command or Action	Remarks & Verification	
		Telemetry Mnemonic	Remarks & TM Data
(3) (Cont'd) ATQ97 (For ADP1 Only)	If the 8 MSBs have to be changed: ADP Configure - PLL Spin Period Magnitude (MSBs)	ASIMSZ	PLL Spin period Magnitude (8 MSB of 16 Bits) } Same as that Commanded.
ATQ98 (For ADP1 Only)	If the 8 LSBs have to be changed: ADP Configure - PLL Spin Period Magnitude (LSBs)	ASIMSZ	PLL Spin Period Magnitude (8 LSB of 16 Bits) } Same as that Commanded.
(4)	Transfer to the simulated SRR:		
ATQ93 (For ADP1 Only) (PLD; REFSIM)	ADP Configure - ADP Mode Select (PLL loss of lock inhibited; simulated SRR selected).	ALOLES	PLL Loss of Lock = 0 (Inhibited)
		ALOCKS	PLL Loss of Lock = 1 (In-Lock) - This is not a true indicator of lock status when PLL Loss of Lock has been inhibited.
		ASRRMS (Word 4, Bit 6)	SRR/Simulated SRR = 0 (Simulated SRR)
		ATTM1Z OR ATTM2Z	Attitude Measurements - (SRR to SRR) now indicates last commanded simulated SRR spin period.
			• Ground received signal strength via HGA will not change, indicating HGA is moving azimuthally. This is due to the difference between the commanded spin rate and the actual spin rate.

TABLE 4.3.3-10
STAR TO SUN TRANSFER OF SRR

COMMAND MNEMONIC OR STEP #	COMMAND OR ACTION	REMARKS & VERIFICATION	
		TELEMETRY MNEMONIC	REMARKS & TM DATA
	• Initial conditions are as shown in Table 4.3.3-2 (Essentially: ADP1 ON; DCE1 ON; HGA Despun, pointed to earth l.o.s., and in use for uplink and downlink; and either PSI* channel or PSI2* channel is in use for the SRR).		
(1) Through (8)	• Perform Steps (1) through (8) of Table 4.3.3-6 (Sun to Sun (Different Sensor Input) Transfer of SRR), with one exception: In Step (2), configure the star channel (PSI* or PSI2*) that is not being used as the pre-transfer SRR Star Channel.		
(9)	Transfer the SRR to any one of the three sun sensors of choice:		
ATQ03 (For ADP1 Only) (PLD; SSRX; REFSUN; SUGD; SRAADV)	ADP Configure - ADP Mode Select (PLL loss of lock inhibited; sun sensor midrange or extended range upper or extended range lower; sun selected; Sun Gate Disabled; SRR Advance selected).	ASUNSS	Sun Sensor Select = 00 (Mid-Range) or =01 (Extended Range Lower) or = 10 (Extended Range Upper), as commanded.
		ASRRMS (Word 4, Bit 5)	Sun/Star Select = 1 (Sun)
		ALOLES	PLL Loss of Lock = 0 (Inhibited)
		ALOCKS	PLL Loss of Lock = 1 (In-Lock). This is not a true indicator of Lock status when PLL Loss of Lock has been inhibited.
		ADVANS	SRR Advance = 1 (Advanced)
		ATTM1Z OR ATTM2Z	Attitude Measurements— • Calculate spin angle delay to "bias measurement" star. Call it "POST-MEAS"
			• All other TM as before.

TABLE 4.3.3-10. (Continued)

COMMAND MNEMONIC OR STEP #	COMMAND OR ACTION	REMARKS & VERIFICATION	
		TELEMETRY MNEMONIC	REMARKS & TM DATA
(10) Through (14)	• Perform Steps (10) through (14) of Table 4.3.3-6.		
(15)	• Establish normal post-transfer mode for ADP:		
ATQ03 (For ADP1 Only) (PLE; SUGE; SRANRM)	ADP Configure - ADP Mode Select (PLL loss of lock enabled; Sun Gate enabled; SRR Advance inhibited).	ASUNGS	Sun Gate = 1 (Enabled)
		ALOLES	PLL Loss of Lock = 1 (Enabled)
		ALOCKS	PLL Loss of Lock - Now a true indicator.
		ADVANS	SRR Advance = 0 (Not Advanced)
			All Other TM as Before.

TABLE 4.3.3-11
STAR TO STAR TRANSFER OF SRR

COMMAND MNEMONIC OR STEP #	COMMAND OR ACTION	REMARKS & VERIFICATION	
		TELEMETRY MNEMONIC	REMARKS & TM DATA
	• Initial Conditions are as shown in Table 4.3.3-2. (Essentially: ADP1 ON; DCE1 ON; HGA despun, pointed to Earth l.o.s. and in use for uplink and downlink; and either PSI* channel or PSI2* channel is in use for the SRR).		
	• This sequence assumes the sun is not available either as an SRR or for AT&M measurements to the PSI2 channel. • The S/C Attitude should be well known on a continuous basis. However, an updating of the Star Map may be required periodically; and, if needed, should be performed prior to transfer to a different star as the SRR.		
(1) Through (3)	• Perform Steps (1) through (3) of Table 4.3.3-7, (Sun to Star Transfer of SRR), with one exception: For the star channel that is the present SRR channel, it is undesirable to change the threshold selection and the bandpass selection. This is to prevent loss of the present SRR star and subsequence loss of DCE control.		
(4)	• Select Star Gate B for the desired SRR Star channel (selected in Step (2)), but retain Star Gate A selection for the present SRR:		

Table 4.3.3-11. (Continued)

COMMAND MNEMONIC OR STEP #	COMMAND OR ACTION	REMARKS & VERIFICATION	
		TELEMETRY MNEMONIC	REMARKS & TM DATA
(4) (Contd) ATQ03 (For ADP1 Only)	ADP 1 Configure - ADP Mode Select (Star Gate A - Retain present channel selection. Star Gate B - Channel 1 (if PSI* was selected in Step (2)) or Channel 2 (if PSI2* was selected in Step (2))).	A*GASS	Star Gate A - No change in TM reading.
		A*GBSS	Star Gate B = 1 (PSI*) or = 0 (PSI2*), as commanded.
(5) Through (9)	• Perform Steps (5) through (9) of Table 4.3.3-7, with the following exception: In Step (9), include Star Gate A channel selection to be the same as the prevailing Star Gate B channel selection. (If it is not already the same).	A*GASS	Star Gate A = 1 or = 0 - Same as for A*GBSS (Star Gate B).
(10) Through (16)	• Perform Steps (10) through (16) of Table 4.3.3-7.		

TABLE 4.3.3-12
SIMULATE SRR TO SUN TRANSFER OF SRR

COMMAND MNEMONIC OR STEP #	COMMAND OR ACTION	REMARKS & VERIFICATION	
		TELEMETRY MNEMONIC	REMARKS & TM DATA
	• Initial conditions are as shown in Table 4.3.3-2 (Essentially: ADP1 ON; DCE1 ON; HGA Despun, pointed to earth l.o.s., and in use for uplink and downlink; and simulated SRR is the SRR).		
	• Without speculating about the probability of occurrence of this sequence, the sequence is presented here for completeness.		
(1) Through (8)	• Perform Steps (1) through (8) of Table 4.3.3-6 (Sun to Sun (Different Sensor Input) Transfer of SRR)		
(9)	Transfer the SRR to anyone of three sun sensors of choice:		
ATC#3 (For ADP1 Only) (PLD; SSRX; REFSUN; SUGD; SRAADV).	ADP Configure - ADP Mode Select (PLL loss of lock inhibited; sun sensor midrange or extended range upper or extended range lower; sun selected; Sun Gate disabled; SRR Advance selected).	ASUNSS	Sun Sensor Select = 00 (Mid-Range) or = 01 (Extended Range Lower) or = 10 (Extended Range Upper), as commanded.
		ASRRMS (Word 4 Bit 5)	Sun/Star Select = 1 (Sun)
		ALOLES	PLL Loss of Lock = 0 (Inhibited)
		ALOCKS	PLL Loss of Lock = 1 (In-Lock). This is not a true indicator of lock status when PLL Loss of Lock has been inhibited.
		ADVANS	SRR Advance = 1 (Advanced)
		ATTM1Z OR ATTM2Z	Attitude Measurements: • Calculate spin angle delay to "bias measurement" star. Call it "POST-MEAS".
			All other TM as before.

Table 4.3.3-12. (Continued)

COMMAND MNEMONIC OR STEP #	COMMAND OR ACTION	REMARKS & VERIFICATION	
		TELEMETRY MNEMONIC	REMARKS & TM DATA
(10) Through (14)	• Perform Steps (10) through (14) of Table 4.3.3-6.		
(15)	• Establish normal post-transfer mode for ADP:		
ATQ03 (For ADP1 Only) (PLE; SUGE)	ADP Configure - ADP Mode Select (PLL Loss of Lock Enabled; sun gate enabled).	ASUNGS	Sun Gate = 1 (Enabled)
		ALOLES	PLL Loss of Lock = 1 (Enabled)
		ALOCKS	PLL Loss of Lock - Now a true indicator
			• All Other TM as Before.

TABLE 4.3.3-13
SIMULATE SRR TO STAR TRANSFER OF SRR

COMMAND MNEMONIC OR STEP #	COMMAND OR ACTION	REMARKS & VERIFICATION	
		TELEMETRY MNEMONIC	REMARKS & TM DATA
	• Initial conditions are as shown in Table 4.3.3-2. (Essentially: ADP1 ON; DCE1 ON; HGA despun, pointed to earth l.c.s., and in use for uplink and downlink; and simulated SRR is the SRR).		
	• This is a general purpose sequence and <u>not</u> the specific sequence to be used as the 2nd step in a two-step transfer from sun to star SRR during on-orbit eclipse entry (Refer to Para. 3.3.3.7.5 and to Table 4.3.3-8 (Sun-to-Simulated SRR-to-Star Transfer of SRR-(Semi-Automatic)) for further description).		
(1) Through (15)	• Perform Steps (1) through (15) of Table 4.3.3-7, Sun to Star Transfer of SRR, except do only the limited star mapping in Step (1) <u>at the most</u>. The true spin period of the S/C can easily deviate ±1 bit (0.325 milliseconds) due to various causes, so that this transfer from the simulated SRR should be done expediently, as the SRR Candidate star can drift out of star Gate B in a matter of tens of minutes (Refer to Para. 3.3.3.7.5 for further discussion of drift Rate). It is also necessary to include SRR Select in Step (9).	ASRRMS (Word 4, Bit 6)	SRR/Simulated SRR = 1 (SRR)

4.3.4 High Gain Antenna Pointing. There are two distinct operational modes utilized in pointing the orbiter high gain antenna (HGA) during the Pioneer Venus mission. The first mode involves fixed inertial pointing. An azimuth position is established by adjusting the despun platform phase angle measured from an inertial reference (sun or star) to the desired azimuth orientation. An elevation position is separately established by command of the HGA elevation positioner. The second mode involves open loop inertial tracking using commanded rates. Slewing rates are commanded separately for azimuth and elevation, and the antenna moves at these rates in both directions simultaneously. The motion is a series of small steps: the azimuth step size is 0.088 deg (1 part in 2^{12}); the elevation step size is 0.041 deg (beam travel). Several rate commands are issued in a timed series to provide a piece-wise linear approximation to a desired nonlinear tracking profile.

The fixed pointing mode is utilized during the entire mission to keep the HGA pointed at the earth for communications. The pointing direction is periodically updated to account for change in earth direction. The tracking mode is utilized during the first 80 days on orbit near periapsis as the spacecraft is occulted from earth view. The apparent earth direction varies during occultation due to refraction of the RF signal as it passes through the Venus atmosphere. The virtual earth track is approximated by a series of azimuth and elevation rate commands stored in the spacecraft command memory. Changes in the HGA pointing are also commanded during initial checkout and later during RF calibration. A calibration profile is established from a series of fixed pointing positions. The antenna dwells at each position long enough to average-out signal fluctuations.

The following Section, 4.3.4.1, describes the operational sequences required to despin or spin the HGA and to switch the communications link between the omni antennas and the HGA. Fixed Pointing of the HGA is discussed in Section

4.3.4.2, slewing is discussed in Section 4.3.4.3, RF calibration is discussed in Section 4.3.4.4, and Failure Modes are discussed in Section 4.3.4.5.

4.3.4.1 General Operational Characteristics. Before the high gain antenna can be used for communications, for virtual earth tracking during occultation, or for RF calibration as described in Sections 4.3.4.2 through 4.3.4.4, the antenna platform must be despun and communication must be routed to the HGA. The despinning and communications switching functions are discussed in Sections 4.3.4.1.1 and 4.3.4.1.2. The reciprocal operations of allowing the high gain antenna to spin up and of switching communications back to the omni antennae are discussed in Sections 4.3.4.1.3 and 4.3.4.1.4.

4.3.4.1.1 Despinning the High Gain Antenna. The high gain antenna will be despun following spacecraft separation from the launch vehicle, spin-up, and magnetometer boom deployment as discussed in Sections 4.2.1 and 4.2.2. The HGA will also be despun following orbit insertion as discussed in Section 4.2.3. The HGA may be spun up and despun at other times during cruise or on orbit.

The sequence of commands required to despin the antenna is given in Table 4.3.4.1.1-1. The despin control electronics is turned on, a phase angle is selected, and the motor drive is initiated. The attitude data processor, ADP, must be locked onto the roll reference before the motor drive is turned on.

4.3.4.1.2 Switching Communications to the HGA. One command (ANT21 or ANTB1) switches the uplink and downlink communications path from the forward omni antenna (or backup antenna) to the HGA. Telemetry channel RANT2S indicates the state of the associated switch. The HGA must be despun as described in Section 4.3.4.1.1 and pointed at the earth before a communications link is established.

TABLE 4.3.4.1.1-1

DESPINNING THE HGA

STEP NUMBER	CMD MNEMONIC	COMMAND TITLE OR EVENT	TELEMETRY VERIFICATION	
			TLM MNEMONIC	TLM TITLE
		(Starting configuration: DCE off, BAPTA Motor Drive off, PLL locked onto SRR)		
1		Turn on either DCE 1 or 2:		
1A	DCE19 or DCEA9	Despin Control Electronics #1 ON/ #2 OFF	ADCE1V	DCE #1 Power Supply Voltage - rises from zero volts to a nominal +5 volts.
		•DCE will initialize in a random manner. This command should be followed by DCE configure quantitative command before sending MOTOR DRIVE 1 ON command.	ADCE2V	DCE #2 power supply voltage - reads zero volts.
			ADCE1T	DCE 1 Temperature - Rises and settles at ____ °F.
			PEUSLI	S/C load current increases 275 milliamps
OR:		OR:		
1B	DCE29 or DCEb9	Despin Control Electronics #2 ON/ #1 OFF	ADCE1V	DCE #1 power supply voltage - reads zero volts.
		•DCE will initialize in a random manner. This command should be followed by DCE configure quantitative command before sending MOTOR DRIVE 2 ON command.	ADCE2V	DCE #2 power supply voltage - Rises from zero volts to nominal +5 volts.

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TABLE 4.3.4.1.1-1 (Continued)

STEP NUMBER	CMD MNEMONIC	COMMAND TITLE OR EVENT	TELEMETRY VERIFICATION	
			TLM MNEMONIC	TLM TITLE
1B (Continued)			ADCE2T	DCE 2 Temperature - Rises and settles at _____ °F.
			PBUSLI	S/C load current increases 275 milliamps
		Initialize Azimuth Pointing instructions to the HGA:		
2A	DCEQ1 (For DCE #1 ON)	Azimuth Pointing Slew = Zero degrees/minute. (CMD #0) (Quantitative Command)	ATTM1Z ATTM2Z	The time interval measurement of PLL to MIP (master index pulse) provides the despun antenna position profile versus GMT time after Motor Drive is turned ON ahead.
OR:				
2B	DCEQA (For DCE #2 ON)			
3A	DCEQ5 (For DCE #1 ON)	Azimuth Position (Select Desired SRR to MIP angle) - In 2's complement notation. (CMD #4) (Quantitative Command)	ATTM1Z ATTM2Z	The time interval measurement of PLL to MIP (master index pulse) provides the despun antenna position profile versus GMT time after Motor Drive is turned ON ahead.
OR:				
3B	DCEQE (For DCE #2 ON)			

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TABLE 4.3.4.1.1-1 (Continued)

STEP NUMBER	CMD MNEMONIC	COMMAND TITLE OR EVENT	TELEMETRY VERIFICATION	
			TLM MNEMONIC	TLM TITLE
4A	DCEQ6 (For DCE No. 1 ON)	Mode Command (Phase Reset enabled; HGA Position OFF) (CMD No. 5) (Quantitative Command)	ADSPNS	Despin phase reset enable/disable = 1.
OR:			APOS1S	HGA Positioner Electronics 1 ON/OFF = 0 (OFF)
4B	DCEQF (For DCE No. 2 ON)		APOS2S	HGA Positioner Electronics 2 ON/OFF = 0 (OFF)
5		Turn on either motor drive:		
5A	MTR19 or MTRA9 (For DCE No. 1 ON)	Motor Drive No. 1 ON • Antenna will despin and assume a position in azimuth based upon what had been commanded in the DCE configure commands.	AMTR1S	Despin motor drive 1 ON/OFF=1 - identifies DCE No. 1 ON.
			AMTR2S	Despin motor Drive 2 ON/OFF=0 - identifies DCE No. 2 OFF.
			PBUSLI	S/C load current increases 150 milliamps if
			ARATEQ	Rate Torque - Assumes a nominal 2.75V (precalibrated) with no torque commands present.
			APOSTQ	Position Torque - Assumes a nominal 2.5V (precalibrated) with no torque commands present.
			AMOTRI	Despin Motor Drive Current - Assumes a nominal zero amps with no torque cmds present.

TABLE 4.3.4.1.1-1 (Continued)

STEP NUMBER	CMD MNEMONIC	COMMAND TITLE OR EVENT	TELEMETRY VERIFICATION	
			TLM MNEMONIC	TLM TITLE
OR:				
5B	MTR29 or MTRB9 (For DCE No. 2 ON)	Motor Drive No. 2 ON •Antenna will despin and assume a position in azimuth based upon what had been commanded in the DCE configure commands.	AMTR1S	Despin motor drive 1 ON/OFF=0, Identifies Unit No. 1 OFF.
			AMTR2S	Despin motor drive 2 ON/OFF=1, Identifies Unit No. 2 ON.
			PBUSLI	S/C Load current increases 150 milliamps.
			ARATEQ	Rate torque - Assumes a nominal 2.75V (pre-calibrated) with no torque commands present.
			APOSTQ	Rate torque - Assumes a nominal 2.5V (pre-calibrated) with no torque commands present.
			AMOTRI	Despin motor drive current - assumes a nominal zero amps with no torque commands present.
		Position HGA in elevation, if desired:		

TABLE 4.3.4.1.1-1 (Continued)

STEP NUMBER	CMD MNEMONIC	COMMAND TITLE OR EVENT	TELEMETRY VERIFICATION	
			TLM MNEMONIC	TLM TITLE
6A	DCEQ4 (For DCE No. 1 ON)	HGA Elevation Positioner Slew = Desired Slew Rate (CMD No. 3) (Quantitative Command)	AELEVD	HGA elevation position No change until next step.
OR:				
6B	DCEQD (For DCE No. 2 ON)			
7A	DCEQ6 (For DCE No. 1ON)	Mode Command. (Phase Reset enabled; HGA Position ON.) (CMD No. 5) (Quantitative Command)	APOS1S	HGA Positioner Electronic 1 ON/OFF= 1 (ON) if DCE No. 1 is ON; =0 (OFF) if DCE No. 1 is OFF.
OR:				
7B	DCEQF (For DCE No. 2 ON)		APOS2S	HGA Positioner Electronics 2 ON/OFF =1 (ON if DCE No. 2 is ON;=0 (OFF) if DCE No. 2 is OFF.
			AELEVD	HGA elevation position changes at slew rate selected in Step 6.
		When desired elevation position is reached:		
8A	DCEQ4 (For DCE No. 1 ON)	HGA Elevation Positioner Slew = Zero degrees/minute. (CMD No. 3) (Quantitative Command)		
OR:				
8B	DCEQD (For DCE No. 2 ON)			

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TABLE 4.3.4.1.1-1 (Continued)

STEP NUMBER	CMD MNEMONIC	COMMAND TITLE OR EVENT	TELEMETRY VERIFICATION	
			TLM MNEMONIC	TLM TITLE
9A	DCEQ6 (For DCE No. 1 ON)	Mode Command (Phase Reset enabled; HGA Positioner OFF (CMD No. 5) (Quantitative Command)	APOS1S	HGA Positioner Electronics 1 ON/OFF = 0 (OFF)
OR:				
9B	DCEQF (For DCE No. 2 ON)		APOS2S	HGA Positioner Electronics 2 ON/OFF = 0 (OFF)
		HGA is now positioned for transfer of communications links to it, if desired.		Note: Turning Positioner Electronics OFF introduce 2 step pointing uncertainty upon re-application of Positioner power.

It may be desirable to avoid any possibility of signal interruption when switching from the forward omni antenna to the HGA. Uplink interruption results in a downlink frequency shift when the auxiliary oscillator replaces the coherent reference. Downlink interruption results in a temporary loss of telemetry. In addition, the antenna gain is increased by 25 dB. When exercising the forward omni to HGA switch, uplink and downlink interruption is avoided if the aft omni antenna is utilized for both links. This requires selection of the appropriate uplink frequency and appropriate internal switching of the downlink signal path. A brief signal interruption may still occur when the uplink and/or downlink is switched from the aft omni antenna to the HGA. The downlink frequency shift attendent to an uplink interruption can be minimized by first tuning the uplink frequency to produce the expected downlink frequency via the auxiliary oscillator. Additional details of communication subsystem operation are given in Section 3.7.3.

4.3.4.1.3 Spinning Up the High Gain Antenna. The high gain antenna is spun up to the spacecraft spin rate by turning off the motor drive (MTR1Ø or MTRAØ). The despin control electronics may also be turned off (DCE1Ø or DCEAØ), if desired, to reduce power consumption. However, subsequent despin of the HGA then requires initialization of Azimuth and Elevation rate before the motor drive is turned on. As a minimum, the HGA will be spun up prior to orbit insertion as described in Section 4.2.3.

4.3.4.1.4 Switching Communications from the HGA. Before spinning up the HGA, as described in Section 4.3.4.1.3, or whenever communications through the forward omni (or backup) antenna is desired the link must be switched from the HGA. One command (ANT22 or ANTB2) accomplishes this. Selection between the forward omni and backup antenna is accomplished by command ANT32 or ANTC2. As described in Section 4.3.4.1.2, RF signal interruption will be avoided if the aft omni is selected for uplink and downlink before the switching command is sent.

4.3.4.2 Fixed Pointing. The HGA azimuth pointing direction is changed by revising the phase angle to the inertial roll reference (sun or star) which is accomplished by sending a quantitative command to the despin control electronics. DCEQ1 or DCEQA produce a discrete azimuth rate; DCEQ5 or DCEQE produce a discrete azimuth position. The HGA elevation pointing direction is changed by commanding a discrete rate using the quantitative rate command, DCEQ4 or DCEQD, followed by a zero rate command after a controlled time interval. The time interval between initial rate command and the zero rate command is selected to achieve the desired position change. (NOTE: 1/8 sec time between cmds from memory must be accounted for). The elevation positioner electronics within the DCE must be turned on (DCEQ6 or DCEQF) before an elevation rate can be implemented. Details of utilization of these commands are discussed in Sections 3.3.2 and 3.3.4.

The HGA pointing direction is periodically updated throughout the entire mission from launch by realtime command. During the occultation season, the pointing function is achieved by appropriate biasing of the tracking commands as explained in Section 4.3.4.3.2.3.

Because the S-band HGA beamwidth is broad (±3.8 degrees at -3dB) relative to pointing uncertainties and variation in the earth pointing direction, daily command updates are not necessary. The frequency of position update is a function of required link margin and available operations time. Open loop control, based on ground calculations of the change in earth pointing direction, provides acceptable performance. However, the received RF signal strength can also be used as a reference to achieve optimum antenna positioning. In either case, the antenna position should be adjusted to be the value optimized for midway between adjustments.

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The desired change in HGA pointing direction is the vector sum of components due to orbital motion of the earth with respect to the spacecraft and due to attitude precession of the spacecraft. Relative earth motion is predominantly an azimuth motion; however, there is an elevation component during cruise because the spacecraft trajectory is not in the ecliptic plane and a similar elevation component on-orbit because of the inclination of the Venus orbital plane. The elevation and azimuth history of the earth pointing direction, with respect to inertial coordinates centered at the spacecraft, is shown in Figures 4.3.4.2-1 through 4.3.4.2-4. The azimuth direction varies from 25 to 70 degrees (w.r.t. Aries). The maximum azimuth rate is about 1.0 degree/day. Commanded azimuth angles are measured from the roll reference F_S pulse. The relationship to inertial coordinates depends on the reference being utilized as explained in Section 3.3.2.4. Ignoring the several-day transient immediately after launch, the greatest beam elevation offset from zero is about -18 degrees and the maximum rate of change in elevation is 0.45 degrees/day. A positive elevation angle indicates rotation toward the -Z axis.

These maximum elevation and azimuth rates result in an accumulated pointing error of 1.6 and 3.5 degrees, respectively, for position update once every week (assuming position is optimized for the subsequent period before next update). This corresponds to about a 2.6 dB drop in link performance from the peak.

Spacecraft attitude precession primarily affects elevation pointing. The shift during cruise is due to solar forces acting on the asymmetric spacecraft. Additional perturbations occur while on-orbit due to aerodynamic forces as the spacecraft passes through the upper atmosphere. The magnitude of attitude precession for the entire mission is given in Section 4.1.2. There may also be attitude shifts due to jet misalignment which can be estimated by attitude

determination following maneuvers. Subsequent RF calibration may be desired.

4.3.4.3 Slewing. Slewing the HGA in azimuth or elevation is achieved by use of the quantitative commands identified in Section 4.3.4.2. A zero rate command terminates a slew. It is operationally convenient to accomplish a tracking profile by use of a series of commands stored in the command memory. Appropriate stored time delay codes result in known positions at the start and finish of each rate. The despin control electronics must be ON (DCE19, DECA9, DCE29, or DCEB9) to implement azimuth or elevation rate commands. In addition, the despin motor drive must be ON (MTR19, MTRA9, MTR29, or MTRB9) for azimuth slewing and the elevation positioner electronics must be ON (DECQ6 or DCEQF) for elevation slewing. Details of command utilization are discussed in Sections 3.3.2 and 3.3.4.

HGA slewing is utilized during initial antenna checkout and during the occultation season. Details of this usage are given in Sections 4.3.4.3.1 and 4.3.4.3.2, respectively. Additional HGA positioning during calibration is discussed in Section 4.3.4.4.

4.3.4.3.1 Initial Checkout of Despun Antenna. After completion of the post-launch spinup, magnetometer boom deployment and antenna platform despinning sequences, described in Sections 4.2.1, 4.2.2 and 4.3.4.1, the antenna will be slewed in elevation from its launch position, beam angle = +17.3 degrees, to the expected earth pointing direction, beam angle = -20 degrees. In this manner, elevation slewing will be qualitatively verified. Quantitative performance may be determined at this time by sending position and rate commands and using position telemetry and/or received signal strength for verification. Telemetry of azimuth position is through channels ATTM1Z and ATTM2Z which indicate the time delay between the selected roll reference and the MIP when AM1ADS or AM2ADS indicates bits 110 (one of seven different selectable telemetry measurements). The

elevation reading is telemetered in channel ABLRVD from a calibrated potentiometer contained within the elevation positioner.

4.3.4.3.2 Occultation. Occultation of the earth-spacecraft line of sight by Venus occurs near periapsis during the first 80 days after orbit insertion. Just before and just after the spacecraft disappears behind Venus as viewed from the earth, the signal strength diminishes or increases in a manner characteristic of the Venus atmosphere through which it is passing. Interruption of the communication link can be delayed by biasing the orientation of the transmitted signal to account for refraction, i.e., by tracking the virtual earth.

The longer the virtual earth is tracked, the deeper the penetration of the refracted signal and the larger the altitude range of measured atmosphere. Tracking performance is specified to a total refracted angle of 17 degrees (X-band) which is reached in all early orbits, and at a slightly later time for consecutive periapsis passes according to the relative geometry of the spacecraft, Earth, and Venus. Near the end of the 80-day occultation season, the 17 degree angle is not reached at all and the virtual earth can be tracked continuously during the entire periapsis pass.

A plot of azimuth and elevation position of the virtual earth for the 80-day occultation season is given in Figures 4.3.4.3.2-1 and -2, respectively. The angle is plotted as a function of time from periapsis and parametrically with number of days on-orbit. The plots terminate when the refracted angle reaches 17 degrees.

Unlike operating during cruise and after the occultation season where only S-band is required, occultation measurements are enhanced by transmitting at both X- and S-band. Because the antenna pattern is much sharper at X-band than at S-band, pointing is referenced to the peak of the X-band pattern. The divergence loss, the

relative pointing offset between the X-band and S-band beams, is therefore minimized.

The occultation season consists of three distinct phases. The first phase, primarily the first day after orbit insertion, offers limited performance since maneuvers to establish a precise attitude and orbit are being executed and pointing is therefore uncertain. The second phase begins after these maneuvers are over and ends when the first periapsis altitude correction is performed. Performance is best during this phase. During the third and final phase the altitude correction maneuvers, increased aerodynamic torques, and increased nonlinearity of the virtual earth track reduce performance.

4.3.4.3.2.1 Phase 1. Occultation data during the first periapsis pass after orbit insertion will be less accurate than data from the following passes. This results from the orbit and attitude maneuvers performed at this time as indicated in 4.3.1.2.2. There will be no RF calibration prior to the first periapsis pass for use in nulling the pointing errors. RF calibration must await spacecraft attitude stabilization. In addition, there will not be sufficient attitude determination data for use as an alternate calibration source. Performance may still be close to 1.0 degree total inertial position tracking error based on calculated pointing.

4.3.4.3.2.2 Phase 2. The second phase of the occultation season follows the orbit and attitude corrections discussed above and terminates when the first periapsis altitude correction maneuver is performed. Accurate pointing is premised on well-known initial spacecraft attitude and well-known disturbances to this attitude. Initial RF calibration further enhances pointing accuracy.

During occultation, the spacecraft command memory is programmed to slew the antenna along the virtual earth track. A tracking error results from several sources. Principal errors include spacecraft dynamics, mechanism and platform dynamics, attitude determination, orbit

determination, thermal distortion, command execution timing, mechanical alignment, and RF alignment. One means of calibrating several of these errors is to perform an initial in-flight RF calibration (ground received RF signal strength versus commanded step position) as described in Section 4.3.4.4. With such a calibration, three methods of operating the spacecraft and reducing error buildup include relying on additional RF calibrations, bookkeeping command magnitude and times, and relying on position telemetry. If RF calibration is never performed, then two additional cases with a common reliance on attitude determination include the bookkeeping approach (which requires driving the antenna against a stop for initial calibration) and utilization of position telemetry. These five options which span a range of achievable performance and a range of operational requirements are presented in Reference: Paragraph 1.5.2.6.

A comparison of the five operational options is summarized in the following table and described below.

OCCULTATION OPERATIONAL OPTIONS

Calibration Technique	Advantages	Disadvantages
1) Initial BF Calibration: BF Calibration Update once every three days	•Least sensitive to attitude disturbance model. •Good performance, pointing error = 0.92°	•Greatest operations requirements
2) Initial BF Calibration: Bookkeeping Update	•Best performance pointing error = 0.92°	•Significant operations requirements •Sensitive to attitude disturbance model
3) Initial BF Calibration: Position Telemetry Update	•Pointing error = 0.95°	•Sensitive to attitude disturbance model
4) No initial BF Calibration: Bookkeeping Update	•Moderate Operations Requirements	•Pointing error = 1.03°*
5) No initial BF Calibration: Position Telemetry Update	•Minimum Operations Requirements	•Pointing error = 1.05°*

*Design goal pointing error = 1.0 degree.

The indicated performance values are based on the operating sequence presented in Section 4.3.4.3.2.6 for the 40th day of the occultation season. This bounds the performance for the first 40 days assuming there is no altitude correction during this time. Any altitude correction must be followed by a new RF calibration (options 1, 2 and 3) to maintain this performance.

A design goal has been established to track the virtual earth to less than 1.0 degree. The corresponding range in X-band signal strength is 2.4 dB. (0.2 dB at S-Band). As indicated in the previous table, an initial calibration is required to achieve this design goal pointing error. RF calibration is preferred as a direct measure of initial pointing error.

The command memory is updated periodically to reflect changes in orbital geometry and spacecraft attitude. It is desirable to have a direct measure of these changes when updating the command sequence rather than relying on calculations. Operating Option 1 provides this update based on additional RF calibrations. The error buildup due to attitude disturbance uncertainty is thereby determined. An altitude disturbance modeling uncertainty of ±10% results in a maximum error of 0.04 degree over a three-day period. The operations required for periodic RF calibration are discussed in Section 4.3.4.4.2.

The 0.925 degree pointing error performance presented above for Option 1 is based on update every three days. (Basis for this pointing error is described in Reference: Paragraph 1.5.26). Increasing the time results in a small performance loss. Similarly, decreasing the time provides only a small improvement.

If periodic RF calibration is not done, then predicted precession due to attitude disturbance must be included when calculating pointing updates. (The predicted information is contained in Reference: Paragraph 1.5.34). Options 2 and 3

illustrate two techniques of determining the remaining pointing position errors prior to memory update. Option 2 relies on bookkeeping of all the previous command rates and times. From this information, the present position is accurately defined compared to the initial position. Option 3 utilizes the position telemetry. This introduces the telemetry quantization error which cannot be calibrated. However, it reduces the data reduction and ground operations task. Because temperature variation during the occultation season at periapsis is not great, telemetry uncertainty due to temperature dependence is not significant.

The apparent paradox that bookkeeping, Option 2, results in better performance than periodic calibration, Option 1, is due to the assumed time constraints for the calibration (see Reference: Paragraph 1.5.27). If more data points are taken during each periodic calibration, the performance can be improved for Option 1 to a limiting value (periodic RF calibration to same accuracy as initial RF calibration) of 0.910 degrees.

Predicted performance without initial RF calibration, Options 4 and 5, does not meet the 1.0 degree tracking error goal. However, because Options 4 and 5 do not utilize RF calibration they do depend on accurate modelling of precession due to attitude disturbance. Attitude determination is the basis of these cases; precession is continually monitored. The disturbances still must be predicted during the time to elapse until the next command memory update.

4.3.4.3.2.3 Phase 3. Once a periapsis altitude reduction maneuver is performed, the initial RF calibration is no longer valid. Either another calibration is required or operating Options 4 or 5 must be implemented. Later in the mission, the virtual earth track is more nonlinear. More commands and more linear approximating segments are required to provide the same performance or else degraded performance results. In addition, aerodynamic torques become substantial as the

periapsis altitude is lowered to regions of greater atmospheric density (near 150 km altitude). The magnitude of the additional attitude precession is discussed in Section 4.1.2. Additional attitude uncertainties up to 1.0 degree may be expected.

4.3.4.3.2.4 Second Season. A second occultation period occurs near apoapsis from day 156 to 166. The HGA may be slewed as desired during these passes. However, the great altitude of the spacecraft above Venus precludes any significant advantage of antenna slewing.

4.3.4.3.2.5 Occultation Operating Sequence. The occultation operating sequence consists of a series of azimuth and elevation rate commands contained in the spacecraft command memory to provide a piecewise linear approximation to the virtual earth track shown in Figures 4.3.4.3.2-1 and 4.3.4.3.2-2. The sequence is programmed to account for variations during the subsequent interim period until the next program update. The timing is selected to correspond to the average requirements of this period. At the end of each daily sequence, a bias determined via offline ground software can be incorporated which accounts for daily earth motion and predicted daily precession due to environmental forces. The final azimuth and elevation pointing direction after completion of a periapsis passage is therefore different than the position at the start of periapsis passage but corresponds to the value desired for the start of the next passage.

A detailed occultation sequence is given in Table 4.3.4.3.2.5-1 for the 40th day of the occultation season. This sequence is representative of that for any day of Phase 2 described in Section 4.3.4.3.2.2 and bounds the number of commands required to achieve 1.0 degree inertial tracking error for any occultation profile of Phase 2 up to the first 40 days.

TABLE 4.3.4.3.2.5-1

OCCULTATION SEQUENCE (40TH PERIAPSIS PASS)

Relative Time	CMD Mnemonic	Command Title or Event	Telemetry Verification/Remarks
P = Periapsis		Comments (1) Azimuth beam rate = Azimuth Mechanical Rate (2) Elevation beam rate is given with mechanical rate in parentheses. (3) Azimuth rate quantization = 0.05 deg/min Elevation Rate quantization = TBD deg/min (Mech. Rate Quant. = 0.02 Deg/min.)	The following channels indicate antenna position. (There is no slewing rate telemetry). Azimuth: Attitude measurement - ATTM1Z, ATTM2Z, and AM1ADS (or AM2ADS). (Select SRR - MIP measurement to determine azimuth position). Elevation: HGA Elevation - AELEVD
Before P-9.2 min.	DCEQ6 (or DCEQF)	HGA Positioner ON	HGA Positioner Elect. ON/OFF - APOS1S or APOS2S
	TDC	Time Delay as Required	
P-9.2 Min.	DCEQ4 (or DCEQD)	Elevation Rate = +5.88 deg/min. (Mech. Rate = TBD)	
	DCEQ1 (or DCEQA)	Azimuth Rate = -0.50 deg/min.	
	TDC	Time Delay = 0.8 min.	

TABLE 4.3.4.3.2.5-1 (Continued)

Relative Time	CMD Mremonic	Command Title or Event	Telemetry Verification/Remarks
P -8.4 Min.	DCEQ4	Elevation Rate = +5.23 deg/ min. (Mech. Rate = TBD)	
	TDC	Time Delay = 0.8 min.	
P -8.2 Min.	DCEQ1	Azimuth = -0.80 deg/ min.	
	TDC	Time Delay = 1.0 min.	
P -7.2 Min.	DCEQ1	Azimuth Rate = -1.36 deg/ min.	
	TDC	Time Delay = 1.0 min.	
P -6.2 Min.	DCEQ1	Elevation Rate = -1.68 deg/ min. (Mech. Rate = TBD)	Timing of rate commands at P -6.2 min. and P -6.1 min. selected to null out error due to rate quantization (See Ref. par. 1.5.27).
	TDC	Time Delay = 0.1 min.	
P -6.1 Min.	DCEQ4	Azimuth Rate = -0.38 deg/ min.	
	TDC	Time Delay = 0.8 min.	

TABLE 4.3.4.3.2.5-1 (Continued)

Relative Time	CMD Mnemonic	Command Title or Event	Telemetry Verification/Remarks
P +6.9 Min.	DCEQ1	Azimuth Rate = 2.77 deg/ Min.	
	TDC	Time Delay = 0.3 min.	
P +7.2 Min.	DCEQ4	Elevation Rate = +1.25 deg/ min. (Mech. Rate = TBD)	
	TDC	Time Delay = 0.8 min.	
P +8.0 Min.	DCEQ4	Elevation Rate = +1.88 deg/ min. (Mech. Rate TBD)	
	TDC	Time Delay = 0.2 min.	
P +8.2 Min.	DCEQ1	Azimuth Rate = 2.20 deg/ min.	
	TDC	Time Delay = 0.6 min.	
P +8.8 Min.	DCEQ4	Elevation Rate = +2.50 deg/ min. (Mech. Rate = TBD)	
	TDC	Time Delay = As Required.	

TABLE 4.3.4.3.2.5-1(Continued)

Relative Time	CMD Mnemonic	Command Title or Event	Telemetry Verification/Remarks
P +9.4 Min.	DCEQ4	Elevation Rate = +2.88 deg. (Mech. Rate = TBD)	
	TDC	Time Delay = 0.8 min.	Timing selected so final position is biased for best pointing at start of next periapsis pass.
P +10.2 Min.	DCEQ4	Elevation Rate = 0.0 (Mech. Rate = TBD)	
P +10.2 Min.	DCEQ1	Azimuth Rate = 0.0	
	DCEQ6	HGA Positioner OFF	
	TDC	Time Delay = As Required	

The given sequence only indicates commands directly associated with antenna pointing. The total periapsis sequence includes other engineering and scientific instrument commands. To provide maximum flexibility, an individual time delay is assigned to every rate command. In the event that these times can be modified to correspond to existing time slots or if elevation and azimuth rates can be initiated simultaneously, then the number of command executions is reduced without degrading performance.

A second example for an operating sequence to slew the HGA in both azimuth and elevation is shown in Figures 4.3.4.3.2.5-1 and 4.3.4.3.2.5-2 respectively, as it would apply for a nominal orbit day 50. This is an example of the second operational mode defined at the start of Section 4.3.4: Open loop inertial tracking using commanded rates.

The Azimuth control begins with the Azimuth Delay being commanded to 285.91° before Occultation start, i.e., before (PERIAPSIS minus 8.88 MINUTES). Throughout the occultation, the Azimuth rate will be changed at the start of each of five time segments, and finally at the end of the fifth time segment, as shown in the following table:

TIME SEGMENT NUMBER	SEGMENT START TIME IN MINUTES, RELATIVE TO PERIAPSIS	COMMANDED AZIMUTH RATE (DEGREES/MINUTE)
1	-8.709	-1.932
2	-6.754	-3.300
3	+0.328	+4.064
3	+3.520	+4.587
4	+6.511	+2.012
5	+6.736	0.000

The linear segmentation of Azimuthal position of the HGA is compared to the ideal dish azimuth in Figure 4.3.4.3.2.5-1, and the resulting RMS error

between commanded and ideal positions is also illustrated.

The Elevation Control is analogous to the Azimuth Control. Throughout the occultation, the elevation rate will be changed at the start of each of five time segments (that begin at different times than those for azimuth control), and finally at the end of the fifth time segment, as shown in the following table:

TIME SEGMENT NUMBER	SEGMENT START TIME IN MINUTES, RELATIVE TO PERIAPSIS	COMMANDED ELEVATION DISH RATE (DEGREES/MINUTE)
1	-9.079	+1.622
2	-6.224	+0.977
3	-4.337	-0.870
4	+3.546	-0.059
5	+5.030	+0.557
---	+6.788	0.000

Dish elevation angle at the start of occultation (Periapsis -8.88 MINUTES) is assumed to be -1.49°.

The linear segmentation DISH elevation position of the HGA is compared to the ideal dish elevation in Figure 4.3.4.3.2.5-2, and the resulting RMS error between commanded and ideal BEAM (not DISH) elevation positions is also illustrated.

4.3.4.4 RF Calibration. As described in Section 4.3.4.3.2.2, an initial RF calibration is desirable to establish a well-known initial pointing position from which subsequent pointing commands can be based. A representative initial RF calibration sequence is presented in Table 4.3.4.4-1. Another is contained in Reference: Paragraph 1.5.28. As also described in Section 4.3.4.2.2.2, a periodic RF calibration is desirable to re-establish a known pointing position without relying on predictions, in particular, without relying on accurate

prediction of environmental forces and resultant attitude precession. A representative periodic RF calibration sequence is presented in Table 4.3.4.4-2.

All RF calibrations are nominally performed at X-band. Because the antenna pattern is much sharper at X-band than at S-band, the losses due to pointing error are more severe at X-band. Calibration at X-band removes the error due to divergence of the S- and X-band patterns (approximately 0.1 degree). Furthermore, less calibration data are required to achieve a desired pointing accuracy for the sharper X-band pattern. (3 dB half beamwidth = 1.1 degree X-band and 3.8 degrees S-band).

4.3.4.4.1 Initial Calibration. If an antenna positioning strategy is selected based on an initial RF calibration, this calibration is best performed as early as possible after orbit and attitude maneuvers are completed during the first day on orbit. Operations constraints must be considered in selecting the exact timing of an initial calibration. Time must be allowed between the calibration and the subsequent periapsis passage to allow for data analysis and subsequent realtime antenna positioning to the calculated zero point.

Prior to calibration, the antenna is pointed directly at the calculated position of the earth based on attitude determination, ground alignment measurements, and position telemetry. The initial position need only be accurate to about 1.0 degrees since the pattern calibration error away from the boresight (up to -3 dB) is negligible. The calibration sequence then begins with the antenna slewed in azimuth while held at constant elevation. The antenna is stopped at many points extending beyond the -3 dB points on both sides of the peak. A similar elevation calibration follows. A sufficient number of points and a sufficient dwell time at each point establishes the antenna pattern to the desired accuracy. See Reference: Paragraph 1.5.27. The pointing performance presented in Section

4.3.4.3.2.2 is based on an RF calibration accuracy of 0.1 degree azimuth and 0.1 degree elevation (beam). This corresponds to <0.1 dB/degree (S-band) and ≈2 dB/degree (X-band) in the vicinity of the peak of the gain pattern.

Additional azimuth and/or elevation plots can be performed to statistically reduce any error and assure the 0.1 degree pointing error is achieved in azimuth and elevation. After calibration is completed, the antenna is repositioned in elevation and azimuth to the new calculated zero position by real time command using recorded step history rather than position telemetry as a reference.

A detailed representative initial RF calibration command sequence is given in Table 4.3.4.4.1-1. Azimuth and elevation position telemetry provides verification as indicated. The received downlink will also qualitatively verify antenna motion. The calibration sequence may be performed by stored or real time commands. Although slow slew rates are indicated in the table to provide sufficient dwell at each calibration point, individually programmed steps may also be employed. The dwell time must be long enough (at least 12 seconds) to provide averaging of periodic variation due to spacecraft wobble mechanism backlash, and azimuth jitter and to average statistical fluctuations in transmitter and receiver performance.

It is assumed that variation in earth atmosphere loss as a function of ground elevation angle and local time varying phenomena is known. A check of this may be made by performing consecutive calibrations in opposite directions and comparing the final data point with the initial one.

4.3.4.4.2 Periodic Calibration. A representative periodic RF calibration sequence is given in Table 4.3.4.4.2-1. The procedure is greatly abbreviated compared to that discussed in Section 4.3.4.4.1 for initial RF calibration. The local gain slope is determined at a point on each side of the peak and compared to the gain profile

TABLE 4.3.4.4.1-1

INITIAL RF CALIBRATION (REPRESENTATIVE)

Relative Time	CMD Mnemonic	Command Title or Event	Telemetry Verification/Remarks
T1 = Start of Azimuth Calibration Sequence T2 = Start of Elevation Calibration Sequence		Comments: (1) Prior to initiation of calibration sequence, antenna must be pointed towards earth (DCE and motor drive must be ON). (2) Calibration is performed at X-Band. (3) Azimuth Beam Rate = Azimuth Mechanical Rate. (4) Elevation Beam Rate is given with mechanical rate in parentheses. (5) Azimuth Rate quantization = 0.05 deg/min. Elevation Rate quantization = TBD. (Mech. Rate Quant. = 0.02 deg/min).	The following channels indicate antenna position. (There is no slewing rate telemetry). Azimuth: Attitude Measurement - ATTM1Z, ATTM2Z, and AM1ADS (or AM2ADS) (Select SRR - MIP Measurement to determine azimuth position) Elevation: HGA Elevation - AELEVD.
		<u>Begin Azimuth Calibration</u>	
T1 = 0	DCEQ1 (or DCEQA)	Azimuth Rate = -5.0 deg/min.	Arbitrary rate (if commands are real time, a slower rate may be desired to allow more time before next command).
T1 +0.6 Min.	DCEQ1	Azimuth Rate = 0.0 deg/min.	Beam motion stopped after -3.dB point is passed (±1.1 deg X-Band). Command only necessary if additional ground preparation time is required. Otherwise, omit.

TABLE 4.3.4.4.1-1 (Continued)

Relative Time	CMD Mnemonic	Command Title or Event	Telemetry Verification/Remarks
T1 +0.6 Min. (Plus time delay for any additional ground preparation	DCEQ1	Azimuth Rate = +0.425 deg/min. Alternate: Antenna discretely commanded to any set of calibration points using approximately timed rate commands.	≤ One step (0.088 degrees) per spin period (12 seconds) allows averaging of S/C wobble. 68 calibration points (Steps).
T1 +20.118 Min.	DCEQ1	Azimuth Rate = -5.0 deg/min.	Arbitrary Rate.
T1 +20.718 Min.	DCEQ1	Azimuth Rate = 0.0 deg/min.	Antenna repositioned to original pointing direction before calibration.
		End Azimuth Calibration	There may be a pause here to await analysis of calibration data and subsequent repositioning of antenna before proceeding (alternately, repositioning can be done at a later time without degrading additional calibration activities).
		<u>Begin Elevation Calibration</u>	
T2 -TBD Sec.	DCEQ6 (or DCEQF)	HGA Positioner ON	HGA Positioner Elec. ON/OFF - APOS1S or APOS2S.
T2 = 0	DCEQ4 (or DCEQD)	Elevation Rate = -5.0 deg/min. (Mech. Rate = TBD)	Arbitrary rate (if commands are real time, a lower rate may be desired to allow more time before next command).
T2 +0.6 Min.	DCEQ4	Elevation Rate = 0.0 deg/min. (Mech. Rate = TBD)	Beam Slewed passed -3 dB point (±1.1 deg X-band) Command only necessary if additional ground preparation time is required. Otherwise omit.

TABLE 4.3.4.4.1-1(Continued)

Relative Time	CMD Mnemonic	Command Title or Event	Telemetry Verification/Remarks
T2 +0.6 Min. (Plus time delay for any additional ground preparation)	DCEQ4	Elevation Rate = 0.189 deg/min. (Mech. Rate = TBD)	± One step (0.040 Degrees) per spin period (12 seconds)
		Alternate: Antenna discretely commanded to any set of calibration points using appropriately timed rate commands.	Allows averaging of S/C wobble 150 calibration points (Steps).
T2 +53.510 Min.	DCEQ4	Elevation Rate = -5.0 deg/min. (Mech. Rate = TBD)	Arbitrary Rate.
T2 +54.110 Min.	DCEQ4	Elevation Rate = 0.0 deg/min. (Mech. Rate = TBD)	Antenna repositioned to original pointing direction before calibration.
	DCEQ6	HGA Positioner OFF	HGA Positioner Elect. ON/OFF - APOS1S or APOS2S.
		End Elevation Calibration	There may be a pause here to await analysis of calibration data and subsequent repositioning of antenna before proceeding (alternately, repositioning can be done at a later time without degrading additional calibration activities).

TABLE 4.3.4.4.1-1(Continued)

Relative Time	CMD Mnemonic	Command Title or Event	Telemetry Verification/Remarks
T = TBD	DCEQ1 and DCEQ4	Reposition antenna based on analysis of calibration data.	Azimuth and elevation calibrations may be repeated prior to final positioning. This verifies assumed insensitivity to small offset of original pointing direction and allows calibration of unknown time dependent system biases. Final position includes allowance for average bias during period preceeding next calibration.

established by initial calibration to determine the location of the pattern peak. The sequence can be executed by command memory with later realtime commands to position the antenna to the indicated pointing direction.

Slope determination accuracy and hence accuracy of the calibration update depends on the number of data points taken to indicate the slope. Reference: Paragraph 1.5.27 describes the performance achieved as a function of the amount of data taken. For the representative sequence of Table 4.3.4.4.2-1, based on 15 data points on each side of the peak, an accuracy of 0.1 degree is achieved. The total operations time commitment is 7.0 minutes.

The periodic RF calibration sequence is presented in Table 4.3.4.4.2-1, for elevation only. A similar sequence can be performed in azimuth. However, the only appreciable uncertainty, attitude disturbance modelling error, only affects elevation pointing.

The time elapsed between periodic RF calibrations determines the pointing performance achieved. A total inertial position error of 0.925 degree is indicated in Section 4.3.4.3.2.1 for update every three days. For the same periodic calibration sequences, without improving performance by taking more data points, this value is only 0.939 degree if periodic update is every 40 days. However, it must be emphasized that the primary motivation for periodic RF calibration is not enhanced performance but rather assurance that expected performance is achieved. Frequent calibration is therefore desirable.

After periapsis altitude reduction is performed at about day 52, the usefulness of periodic RF calibration is increased. Aerodynamic forces become important and attitude offset occurs due to the maneuver itself. As a minimum, after each maneuver (one is planned at day 52 and another at day 69) a calibration is desirable. In addition, the greater uncertainty encompassing the

TABLE 4.3.4.4.2-1

PERIODIC RF CALIBRATION (REPRESENTATIVE)

Relative Time	CMD Mnemonic	Command Title or Event	Telemetry Verification/Remarks
T = Start of Calibration Sequence		Comments: (1) Periodic RF Calibration required in elevation only. (2) Initial pointing based on previous RF calibration. (3) Calibration is performed at X-Band. (4) Elevation beam rate is given with mechanical rate in parentheses. (5) Elevation rate quantization = TBD (Mech. Rate Quant. = 0.02 deg/min.)	The following channel indicates antenna elevation (There is no slewing rate telemetry) AELEVD.
		Begin Calibration Sequence	
T - TBD Sec.	DCEQ6	HGA Positioner ON	HGA Positioner Electronics ON/OFF - APOS1S or APOS2S.
T = 0	DCEQ4 (or DCEQD)	Elevation Rate = -5.0 deg/min. (Mech. Rate = TBD)	Arbitrary rate (slower rate may be desired for real time commands to provide more time before next command).
T +0.220 min.	DCEQ4	Elevation Rate = 0.0 deg/min. (Mech. Rate = TBD)	Beam slewed to -3.0 dB point (±1.1 deg X-band) Command only necessary if additional ground preparation time is required. Otherwise, omit.

TABLE 4.3.4.4.2-1(Continued)

Relative Time	CMD Mremonic	Command Title or Event	Telemetry Verification/Remarks
T +0.220 Min. (Plus time delay for any additional ground preparation)	DCEQ4	Elevation Rate = 0.189 deg/min. (Mech. Rate = TBD)	≤ One Step (0.040 degrees) per spin period (12 seconds). 15 calibration points.
T +3.395 Min.	DCEQ4	Elevation Rate = +5.0 deg/min. (Mech. Rate = TBD)	Arbitrary Rate.
T +3.595 Min.	DCEQ4	Elevation Rate = 0.189 deg/min. (Mech. Rate = TBD)	15 Calibration points.
T +6.770 Min.	DCEQ4	Elevation Rate = -5.0 deg/min. (Mech. Rate = TBD)	Arbitrary Rate.
T +6.990 Min.	DCEQ4	Elevation Rate = 0.0 deg/min. (Mech. Rate = TBD)	Antenna repositioned to original pointing direction before calibration.
T +7.0 Min.	DCEQ6	HGA Positioner OFF End Calibration Sequence	HGA Positioner Electronics ON/OFF - APOS1S or APOS2S.
T + TBD	DCEQ4	Reposition antenna based on analysis of calibration data.	Include calculated average bias during period preceeding next calibration.

aerodynamic forces warrants additional calibrations.

4.3.4.4.3 RF Calibration(s) During Cruise. Additional RF calibration may be desirable during cruise. The advantages include early verification of calculated performance, familiarity with the operations, generation of a reference for latter calibrations, and optimum pointing for earth communications. This latter factor suggests a calibration following all maneuvers. The first sequence for an RF calibration during cruise would be similar to that presented in Section 4.3.4.4.1 for initial RF calibration. Subsequent calibrations could be reduced in scope corresponding to the periodic calibration sequence of Section 4.3.4.4.2.

4.3.4.5 Failure Modes. Single point failures which degrade operation of the high gain antenna include mechanical or electrical failure of the elevation positioner, failure of the elevation position potentiometer, and failure of the despin bearing.

4.3.4.5.1 Elevation Positioner Failure. Mechanical failure of the HGA elevation positioner mechanism or electrical failure of the positioner control electronics will result in inability to step the antenna in elevation. The severity of this failure depends on the resting elevation of the antenna boresight. If this elevation is near zero, then communications performance is not significantly degraded. If the beam is offset less than 5 degrees, the spacecraft attitude can be biased to achieve nominal communications without any significant sacrifice in scientific instrument, thermal, or power performance. If the beam offset is greater, then communications performance will be degraded and the backup sleeve-dipole antenna may be required. The corresponding worst performance loss is 17 dB.

4.3.4.5.2 Elevation Position Potentiometer Failure. Failure of the HGA elevation position potentiometer or associated telemetry electronics will preclude a direct indication of antenna

elevation. Bookkeeping of all elevation stepping commands will be required to determine elevation position. Otherwise positioning and communications performance are not affected by this failure.

4.3.4.5.3 Despin Bearing Failure. Failure of the despin bearing will preclude azimuth antenna pointing. Communications must be through the backup sleeve-dipole antenna. The corresponding 17 dB reduction in gain requires a factor of 50 reduction in data rate or utilization of a 64 m ground station and a factor of 8 reduction in data rate.

4.3.5 Orbiter Power and Thermal Operations. This section describes the power load and heater configurations, solar panel performance, and battery management for all of the various Orbiter critical mission events.

4.3.5.1 Orbiter Load Management. The Orbiter load configurations are shown in Table 4.3.5.1-1 in chronological order. The start time for each mission phase is indicated at the top; launch (L) is the zero time reference. The sun angles with reference to the spin axis and the solar distances are also listed. The latter determines the effective solar intensity, which varies from 0.850 near Earth to 1.933 near Venus.

All loads are grouped by subsystems. The electrical nature of the load is defined by a letter: P for constant power, I for constant current and R for constant resistance. The operational mode of each load for each mission phase is also indicated: on, standby, or off.

The total spacecraft load, listed near the bottom of the page, represents the power required by all spacecraft loads, including science. This amount of power must be supplied by either the battery, the solar panel, or in a shared battery solar panel mode.

The minimum available solar panel current at 28 volts and the margin, or contingency, are shown

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TABLE 4.3.5.1-1

POWER PROFILE FOR ORBITER

		Pre-Launch and Launch		Cruise - 1		Cruise - 2		Orbit Insertion		Periapsis Occult	
Start Time		-0.500 Hours		1.2 Days		1.48.0 Days		0+0.0 Days		0+0.0 Days	
Mode Duration		1.00 Hr.		0.000 Hr.		0.000 Hr.		0.000 Hr.		0.000 Hr.	
Sun Angle		0.00 Deg.		89.80 Deg.		88.90 Deg.		100.00 Deg.		90.00 Deg.	
Solar Dist.		0.00 mkm		152.2 mkm		161.50 mkm		108.00 mkm		107.60 mkm	
Subsystem/Units		Units ON/SBY/OFF	Watts/Amps 27.5 V	Units ON/SBY/OFF	Watts/Amps 28.0 V	Units ON/SBY/OFF	Watts/Amps 28.0V	Units ON/SBY/OFF	Watts/Amps 28.0 V	Units ON/SBY/OFF	Watts/Amps 28.0 V
Comm (RF)	Xpndr-Rcvr P	2/0/0	4.4/0.160	2/0/0	4.4/0.157	2/0/0	4.40/0.157	2/0/0	4.4/0.157	2/0/0	4.40/0.157
	Xpndr-Xctr P	1/0/1	3.9/0.142	1/0/1	3.9/0.139	1/0/1	3.90/0.139	1/0/1	3.9/0.139	1/0/1	3.90/0.139
	Dmdr. Amps I	1/0/3	10.31/0.375	1/0/3	13.50/1.375	1/0/3	10.5/1.375	2/0/2	21.00/0.750	2/0/2	21.0/0.750
	Pwr. Amps I	1/0/3	31.63/1.150	1/0/3	32.20/1.150	1/0/3	32.2/1.150	2/0/2	64.40/2.300	2/0/2	64.40/2.30
	Switches I	0/1/0	0.20/0.007	0/1/0	0.2/0.007	0/1/0	0.20/0.007	0/1/0	0.20/0.007	0/1/0	0.20/0.007
	X-Band Xmtr I	0/0/1	0.00/0.000	0/0/1	0.0/0.000	0/0/1	0.00/0.000	0/0/1	0.00/0.000	1/0/0	15.40/0.550
Subtotal			50.44/1.834		51.20/0.828		51.25/1.828		93.90/3.353		109.28/3.903
Data	TEL Prosr. I	1/0/1	7.78/0.283	1/0/1	7.92/0.283	1/0/1	7.92/0.283	1/0/1	7.92/0.283	1/0/1	7.92/0.283
	PCM En+8 Dtms I	1/0/1	4.03/0.146	1/0/1	4.10/0.146	1/0/1	4.10/0.146	1/0/1	4.10/0.146	1/0/1	4.10/0.146
	Data storage I	0/0/2	0.00/0.000	0/0/2	0.00/0.000	0/0/2	0.00/0.000	1/1/0	3.56/0.127	1/1/0	3.56/0.127
Subtotal			11.79/0.429		12.02/0.429		12.02/0.429		15.58/0.556		15.58/0.556
Command	Cmd PR-6 Coms I	2/0/0	9.02/0.328	2/0/0	9.18/0.328	2/0/0	9.18/0.328	2/0/0	9.18/0.328	2/0/0	9.18/0.328
	Cmd Memory I	2/0/0	3.91/0.142	0/0/2	0.00/0.000	0/0/2	0.00/0.000	2/0/0	3.98/0.142	2/0/0	3.98/0.142
	Pyro Control I	0/1/0	0.06/0.002	0/1/0	0.06/0.002	0/1/0	0.06/0.002	0/1/0	0.06/0.002	0/1/0	0.06/0.002
Subtotal			12.99/0.472		9.24/0.330		9.24/0.330		13.22/0.472		13.22/0.472
Controls	At Data Proc. I	0/0/2	0.00/0.000	1/0/1	6.90/0.246	1/0/1	6.90/0.246	1/0/1	6.90/0.246	1/0/1	6.90/0.246
	Jet Cont E I	0/0/2	0.00/0.000	0/0/2	0.00/0.000	0/0/2	0.00/0.000	0/0/2	0.00/0.000	0/0/2	0.00/0.000
	Driver E I	2/0/0	0.03/0.001	2/0/0	0.03/0.001	2/0/0	0.03/0.001	2/0/0	0.03/0.001	2/0/0	0.03/0.001
	Star Sensor P	0/0/1	0.00/0.000	1/0/0	1.29/0.046	1/0/0	1.29/0.046	1/0/0	1.29/0.046	1/0/0	1.29/0.046
	Despin Cont E I	0/0/2	0.00/0.000	1/0/1	8.24/0.294	1/0/1	8.24/0.294	0/0/2	0.00/0.000	1/0/1	8.24/0.294
	BAPTA I P	0/0/2	0.00/0.000	1/0/1	8.00/0.286	1/0/1	8.00/0.286	0/0/2	0.00/0.000	1/0/1	8.00/0.286
	HGA E I	0/0/2	0.00/0.000	0/0/2	0.00/0.000	0/0/2	0.00/0.000	0/0/2	0.00/0.000	1/0/1	1.44/0.051
	BAPTA Motor R	0/0/1	0.00/0.000	1/0/0	1.79/0.064	1/0/0	1.79/0.064	0/0/1	0.00/0.000	1/0/0	1.79/0.064
	HGA Position R	0/0/1	0.00/0.000	0/0/1	0.00/0.000	0/0/1	0.00/0.000	0/0/1	0.00/0.000	1/0/0	13.23/0.473
Subtotal			0.03/0.001		26.25/0.937		26.25/0.937		8.22/0.294		40.92/1.461
Prop	Jet Valves R	0/7/0	0.00/0.000	0/7/0	0.00/0.000	0/7/0	0.00/0.000	0/7/0	0.00/0.000	0/7/0	0.00/0.000
	Jet Htr H14 R	2/0/0	1.43/0.052	2/0/0	1.48/0.053	2/0/0	1.48/0.053	2/0/0	1.48/0.053	2/0/0	1.48/0.053
	Jet Htr R23 R	2/0/0	6.71/0.244	2/0/0	6.94/0.248	2/0/0	6.94/0.248	2/0/0	6.94/0.248	2/0/0	6.94/0.248
	Jet Htr A5 R	1/0/0	0.71/0.026	1/0/0	0.73/0.026	1/0/0	0.73/0.026	1/0/0	0.73/0.026	1/0/0	0.73/0.026
	A Jet Htr A6 R	1/0/0	1.71/0.060	1/0/0	1.71/0.061	1/0/0	1.71/0.061	1/0/0	1.71/0.061	1/0/0	1.71/0.061
	A Jet Htr A7 R	1/0/0	1.37/0.050	1/0/0	1.42/0.051	1/0/0	1.42/0.051	1/0/0	1.42/0.051	1/0/0	1.42/0.051
	Prop. Tnk Htrs. R	1/0/1	15.87/0.577	1/0/1	16.44/0.587	1/0/1	16.44/0.587	1/0/1	16.44/0.587	1/0/1	16.44/0.587
	OBM Htrs. R	1/0/1	16.83/0.612	1/0/1	17.43/0.623	1/0/1	17.43/0.623	0/0/2	0.00/0.000	0/0/2	0.00/0.000
Subtotal			44.63/1.621		46.15/1.649		46.15/1.649		28.73/1.026		28.73/1.026
Power	Batt. Charge R	0/2/0	0.00/0.000	2/0/0	9.80/0.350	2/0/0	5.60/0.200	2/0/0	9.80/0.350	0/2/0	0.00/0.000
	Bus Limiter R	0/7/0	0.96/0.035	0/7/0	1.00/0.035	0/7/0	1.00/0.315	0/7/0	1.00/0.035	0/7/0	1.00/0.035
	Under/Over Sw. R	1/0/0	0.35/0.013	1/0/0	0.36/0.013	1/0/0	0.36/0.013	1/0/0	0.36/0.013	1/0/0	0.36/0.013
	Current Sensor R	3/0/0	0.77/0.010	3/0/0	0.28/0.010	3/0/0	0.28/0.010	3/0/0	0.28/0.010	3/0/0	0.28/0.010
	Pwr. Interface R	1/0/0	0.19/0.007	1/0/0	0.20/0.007	1/0/0	0.20/0.007	1/0/0	0.26/0.007	1/0/0	0.20/0.007
	Batt. Disch. R	2/0/0	3.46/0.090	0/2/0	1.40/0.050	0/2/0	1.40/0.050	0/2/0	1.43/0.050	0/2/0	1.43/0.050
Subtotal			4.12/0.155		13.05/0.466		8.84/0.315		13.05/0.466		3.25/0.116
Distribution Loss			3.00/		4.59/		4.53/		4.00/		6.99/
S/C Total			127.01/4.512		165.27/5.639		158.28/5.488		176.68/6.167		
Science	P	1/0/0	2.90/0.091	0/0/1	40.00/1.429	1/0/0	40.00/1.429	0/0/1	00.00/0.000	1/0/0	90.00/3.214
Total S/C Loads			129.51/4.603		205.27/7.068		198.28/6.917		176.68/6.167		307.97/10.78
Solar Panel Power			0.00/0.000		233.28/8.329		199.45/7.123		333.90/11.925		337.24/12.044
Solar Panel Contingency			/0.000		/1.261		/0.206		/5.738		/1.296
Battery Amps Used			/4.603		/0.000		/0.000		/0.000		/0.000
Battery Amps Required			4.603 Amphrs		8.000 Amphrs		0.000 Amphrs		0.000 Amphrs		0.000 Amhrs
Battery Dischg. Depth			30.7%		0.0%		0.0%		0.0%		0.0%

TABLE 4.3.5.1-1. (Continued)

		Perigapsis Eclipse/Occult		Perigapsis Heat		Sunlight		Apoapsis Eclipse	
Start Time		0 40.0 Days		0 180.0 Days		0.0 Days		0 185.0 Days	
Mode Duration		0.417 Hr.		0.200 Hr.		0.600 Hr.		3.360 Hrs.	
Sun Angle		90.00 Deg.		90.00 Deg.		90.00 Deg.		90.00 Deg.	
Solar Dist.		107.50 mkm		107.60 mkm		107.60 mkm		107.60 mkm	
Subsystem/Units		Units ON/SBY/OFF	Watts/Amps 27.5V	Units ON/SBY/OFF	Watts/Amps 27.5V	Units ON/SBY/OFF	Watts/Amps 28.0V	Units ON/SBY/OFF	Watts/Amps 27.5V
Comm(RF)	Xpndr-Rcvr P	2/0/0	4.40/0.157	2/0/0	4.40/0.160	2/0/0	4.40/0.157	2/0/0	4.40/0.160
	Xpndr-Xctr P	1/0/1	3.90/0.142	1/0/1	3.90/0.142	1/0/1	3.90/0.139	1/0/1	3.90/0.139
	Driver Amps I	2/0/2	20.63/0.750	2/0/2	20.63/0.750	2/0/2	21.00/0.750	0/0/4	0.00/0.000
	Power Amps I	2/0/2	63.25/2.300	2/0/2	63.25/2.300	2/0/2	64.40/2.300	0/0/4	0.00/0.000
	Switches I	0/1/0	0.20/0.007	0/1/0	0.20/0.007	0/1/0	0.20/0.007	0/1/0	0.20/0.007
	X-Band Xmtr I	1/0/0	15.13/0.550	0/0/1	0.00/0.000	0/0/1	0.00/0.000	0/3/1	0.00/0.000
Subtotal			107.50/3.903		92.21/3.353		93.90/3.353		8.42/0.306
Data	TM Proces. I	1/0/1	7.78/0.283	1/0/1	7.78/0.283	1/0/1	1.92/0.283	1/0/1	5.60/0.200
	PCMEN & Dims I	1/0/1	4.03/0.146	1/0/1	4.03/0.146	1/0/1	4.10/0.146	1/0/1	4.03/0.146
	Data Storage I	1/1/0	3.50/0.127	1/1/0	3.50/0.127	1/1/0	3.56/0.127	1/0/1	1.95/0.071
Subtotal			15.31/0.556		15.31/0.556		15.58/0.556		11.47/0.417
Command	CMD PR & Coms I	2/0/0	9.02/0.328	2/0/0	9.02/0.328	2/0/0	9.18/0.328	2/0/0	9.02/0.328
	CMD Memory I	2/0/0	3.91/0.142	1/0/0	1.95/0.071	1/0/1	1.99/0.071	1/0/1	1.95/0.071
	Pyro Control I	0/1/0	0.06/0.002	0/1/0	0.06/0.002	0/1/0	0.06/0.002	0/1/0	0.06/0.002
Subtotal			12.98/0.472		11.03/0.401		11.23/0.401		11.03/0.401
Controls	AT Data Proc. I	1/0/1	6.78/0.246	1/0/1	6.78/0.246	1/0/1	6.90/0.246	1/0/1	6.78/0.246
	Jet Cont E I	0/0/2	0.00/0.000	0/0/2	0.00/0.000	0/0/2	0.00/0.000	0/0/2	0.00/0.000
	Driver E I	2/0/0	0.03/0.001	2/0/0	0.03/0.001	2/0/0	0.03/0.001	2/0/0	0.03/0.001
	Star Sensor P	1/0/0	1.27/0.046	1/0/0	1.27/0.046	1/1/0	1.29/0.046	1/0/0	0.63/0.023
	Disp. Cont. E I	1/0/1	8.09/0.294	1/0/1	8.09/0.294	1/0/1	8.24/0.294	0/0/2	0.00/0.000
	BAPTA E P	1/0/1	8.09/0.291	1/0/1	8.09/0.291	1/0/1	8.00/0.286	0/0/2	0.00/0.000
	HGA E I	1/0/1	1.41/0.051	0/0/2	0.00/0.000	0/0/2	0.00/0.000	0/0/2	0.00/0.000
	BAPTA Motor R	1/0/0	1.73/0.063	1/0/0	1.73/0.063	1/0/0	1.79/0.064	0/0/1	0.00/0.000
	HGA Position R	1/0/0	12.76/0.464	0/0/1	0.00/0.000	0/0/1	0.00/0.000	0/0/1	0.00/0.000
Subtotal			40.07/1.457		25.89/0.902		26.25/0.937		7.45/0.271
Prop	Jet Valves R	0/7/0	0.00/0.000	0/7/0	0.00/0.000	0/7/0	0.00/0.000	0/7/0	0.00/0.000
	R Jet Htr B14 R	2/0/0	1.43/0.052	2/0/0	1.43/0.052	2/0/0	1.48/0.053	2/0/0	1.43/0.052
	R Jet Htr B23 R	2/0/0	6.71/0.244	2/0/0	6.71/0.244	2/0/0	6.94/0.248	2/0/0	6.71/0.244
	A Jet Htr A5 R	1/0/0	0.71/0.026	1/0/0	0.71/0.026	1/0/0	0.73/0.026	1/0/0	0.71/0.026
	A Jet Htr A6 R	1/0/0	1.65/0.060	1/0/0	1.65/0.060	1/0/0	1.71/0.061	1/0/0	1.65/0.060
	A Jet Htr A7 R	1/0/0	1.37/0.050	1/0/0	1.37/0.050	1/0/0	1.42/0.051	1/0/0	1.37/0.050
	Prop Trk Htrs R	1/0/1	15.87/0.577	1/0/1	15.87/0.577	1/0/1	16.44/0.587	1/0/1	15.87/0.577
	OEM Htrs R	0/0/2	0.00/0.000	0/0/2	0.00/0.000	0/0/2	0.00/0.000	0/0/2	0.00/0.000
Subtotal			7.75/1.009		27.75/1.009		28.73/1.026		27.75/1.009
Power	Batt. Chg. R	0/2/0	0.00/0.000	0/2/0	0.00/0.000	2/0/0	21.02/0.750	0/2/0	0.00/0.000
	Bus Limiter R	0/7/0	0.96/0.034	0/7/0	0.96/0.034	0/7/0	1.00/0.035	0/7/0	0.96/0.034
	Under/Over Sw. R	1/0/0	0.35/0.013	1/0/0	0.35/0.013	1/0/0	0.36/0.013	1/0/0	0.35/0.013
	Current Sens. R	3/0/0	0.27/0.010	3/0/0	0.27/0.010	3/0/0	0.28/0.010	3/0/0	0.27/0.010
	Pwr. Interf. P	1/0/0	0.19/0.007	1/0/0	0.19/0.007	1/0/0	0.20/0.007	1/0/0	0.19/0.007
	Batt. Disch. R	2/3/0	4.95/0.180	2/0/0	2.48/0.090	0/2/0	1.40/0.050	2/0/0	1.65/0.060
Subtotal			6.85/0.239		4.24/0.154		24.24/0.865		3.41/0.124
Distribution Loss			7.09/		6.30/		6.73/		1.89/
S/C Total			217.08/7.936		182.88/6.421		206.39/7.133		71.41/2.528
Science	P	1/0/0	90.00/3.273	1/0/0	90.00/3.273	1/0/0	90.00/3.214	1/0/0	19.37/0.377
Total S/C Loads			307.08/10.909		272.88/9.694		296.39/10.352		91.78/2.905
Solar Panel Power			0.00/00.000		192.50/7.000		337.24/12.044		0.00/0.000
Solar Panel Contingency			/0.000		0.000		1.692		/0.000
Battery Amps Used			/10.909		2.694		0.000		/2.905
Battery Amphrs Required			4.549 Amphrs		0.539 Amphrs		0.000 Amphrs		9.761 Amphrs
Battery Dsch. Depth			30.3%		3.6%		0.0%		65% of nameplate Capacity 72.3% of capacity after L-374.

at the bottom of the page. Battery current, ampere-hours consumed and battery depth of discharge (DOD) are also listed for those mission phases where the battery is active.

Available solar panel power is a function of solar constant, sun angle and radiation loss. The solar panel power listed in the tables assumes that a massive solar flare has occurred. It also includes a 2 degree solar angle uncertainty. If solar flare activity during the mission is not severe, additional solar panel power will be available.

The loads shown in the tables are based on actual measurements but they still have a small amount of conservatism to allow for measurement error, aging effects, etc. Also, the budget allowance for cruise and orbital science is very generous. Actual science loads (except for apoapsis eclipse) are considerably less than the values shown in the tables.

4.3.5.1.1 Prelaunch, Launch and Spin-up. The launch loads shown in the table are powered from the blockhouse until 30 minutes before liftoff, at which time power is transferred to spacecraft batteries.

The Orbiter science bus must be turned on for launch to allow the Infrared Radiometer (OIR) to maintain its sensor temperature at a safe level to withstand launch vibration loads. The total battery DOD expected for pre-launch, launch and initial spin-up is 29.7%. The battery charging circuitry is commanded into the low rate charge state prior to launch so that the batteries will be recharged soon after the sun is acquired.

To prevent trip-off of the switched loads and r.f. transmitter power buses during launch due to vibration induced relay contact chatter, the override or "power system not protected" relay is turned on during pre-launch via PSP1Ø or PSPAØ: "Power System Protection off". This switches in a redundant path for these power buses.

Actuation of the spacecraft separation switches initiates a command sequence stored in the command memories. These automatic commands turn on both Attitude Data Processors (ADP), turn on spin-up jets for approximately 30 seconds, and then turn off one of the redundant ADP units.

4.3.5.1.2 Cruise. During the time period of L +2 hours through L +13 hours, the following events occur:

(a) The OV/OL switch is commanded to "power system protection on".
(b) Both command memories are turned off.
(c) The four magnetometer boom pin pullers are actuated by the battery pyro bus.
(d) The star sensor is turned on.
(e) The Despin Control Electronics and the BAPTA motor are turned on and the high gain antenna is despun with reference to Earth.
(f) Cruise science is turned on (OMAG, OPA, OEPD, OGBD).

Forty watts of solar panel power are allocated to science during cruise for continuous operation and checkout of instruments. The batteries will be in continuous low rate charge (approximately 150 ma per battery).

There will be excess solar panel capacity during the entire cruise period at the normal cruise attitude of the spacecraft. The margin will be at its lowest value at approximately L +48 days when the spacecraft is the farthest from the sun. During the latter part of cruise near Venus, the solar panel margin will exceed 5 amperes. This excess panel capacity will maintain the bus limiters in an active state and maintain the bus voltage at a constant 29.5 volts throughout cruise.

4.3.5.1.3 Thrust Correction Maneuvers. All thrust correction maneuvers (TCM) are planned to be performed with the spin axis perpendicular to the sun line. The only additional major load, beyond the cruise loads, is a jet thruster solenoid. Therefore, the batteries will not be needed

during the TCMs, assuming the planned attitude for these maneuvers. The solar panel will continue to support all loads and maintain the bus voltage at 29.5 volts.

4.3.5.1.4 Orbit Insertion. Prior to orbit insertion, the DCE and the BAPTA motor are turned off and the high gain antenna is allowed to spin up to the rotor rate. A second 10 watt S-band amplifier module is turned on to increase the downlink data rate.

A failure of the UV/OL unit that would result in a trip if the non-essential power buses during OIM insertion would result in loss of critical DSU data and would necessitate up-link and down-lin reacquisition in the blind. Therefore, it would be prudent to command "power subsystem protection off" (override) just prior to reorientation to the insertion attitude. "Power subsystem protection on" should then be commanded on again after orbit insertion.

Two squibs are actuated by the pyro bus to ignite the orbit insertion motor. Approximatly 36 hours after orbit insertion, three pairs of squibs are actuated with one command: the ORAD antenna pin pullers are released, the ONMS cover is ejected, and the OETP radial sensor is deployed.

4.3.5.1.5 Orbital Operations - Illumination Phases. During orbital operations, spacecraft loads increase sharply because two S-band amplifiers are used continuously and the science load increases to approximately 70 watts. This load configuration is described in the "Sunlight" section of Table 4.3.5.1-1. The solar panel still has sufficient margin to force the bus limiters into conduction at this time.

During the periapsis occultation periods, the X-band transmitter and the high gain elevation drive are also turned on. This configuration is described in the "Periapsis Occultation" column. It represents the highest spacecraft load illumination conditions.

During periapsis passes, planetary albedo will increase the illumination intensity and the temperature of the solar panel. The planetary albedo will continue to increase with each orbital pass and reach a peak value during the apoapsis eclipse season. When the planetary albedo is near its maximum value, the temperature of the panel will increase to the point where the spacecraft loads operate well above the knee of the panel voltage/current curve, and the batteries may have to come on line for a few minutes to share the load. This condition is described in Table 4.3.5.1-1 under the "Periapsis Heat" heading.

4.3.5.1.6 Periapsis Eclipse. Periapsis eclipses start approximately 25 days after orbit insertion and continue for approximately 95 days. The eclipses build up to a maximum of 24 minutes. Maximum battery loads occur when the occultation and periapsis seasons overlap. At this time, the X-band transmitter and high gain antenna drive are both on. Battery DOD is expected to reach a maximum value of 30.0 percent during the longest periapsis eclipse.

4.3.5.1.7 Apoapsis Eclipse. During the 6-day apoapsis eclipse season, the eclipse time reaches a maximum value of approximately 3.8 hours. In order to limit battery DOD to a safe level during the long eclipses, it will be necessary to turn off a large number of loads. This includes the S-band amplifiers and exciters which turn off the downlink carrier. It also includes the DCE and BAPTA motor which allow the high gain antenna to spin up to the rotor rate. The science load must also be limited as shown in Table 4.3.5.1.7-1.

The maximum available science current during apoapsis eclipse is a function of the eclipse time and the maximum allowable depth of discharge (DOD). It is desirable to maintain a conservative maximum DOD of 60%. Since the two batteries are rated at 15.0 ampere-hours (AH), this means that 9.0 AH of energy is available. The maximum anticipated eclipse time is 3.31 hours of umbra and 0.49 hours of penumbra. The

spacecraft must be powered down before the penumbra is reached. This will allow the solar panel to support the spacecraft loads during most the penumbra and reduce the effective eclipse time to the umbra time plus 0.05 hour or 3.36 hours.

TABLE 4.3.5.1.7-1

PRIORITIES AMONG SCIENCE INSTRUMENTS FOR STAYING ON DURING THE APOAPSIS ECLIPSE SEASON (Listed in Decreasing Priority)

(1)	OMAG	
(2)	OPA	
(3)	OEFD	(NOTE: OEFD has top priority for staying ON for any time period within one orbital period containing an apoapsis eclipse).
(4)	OGBD	
(5)	OMAD	
(6)	OUVS	
(7)	OMPA	
(8)	OIMS	
(9)	OETP	
(10)	OMMS	
(11)	OIR	
(12)	OCPP	

The maximum available current for spacecraft and science use is:

$$I_T = \frac{\text{ampere-hours available}}{\text{hours of effective eclipse time}} \text{ (amperes)}$$

For the example cited above, the current is 9/3.36 = 2.679 amperes. From Table 4.3.5.1-1, it is determined that the required spacecraft current (excluding science) is 2.528 amperes.

Therefore, the maximum science allocation is 2.679-2.528 = 0.151 amperes.

When the eclipse times are shorter, or if the allowable DOD is greater, the science allocation can be increased accordingly. For instance, when the effective eclipse time remains the same, as in the above example (3.36 hours), and the allowable battery DOD is increased to 65%, the maximum available current for spacecraft and science use increases to 2.905 amperes. This results in a maximum science allocation of 2.905 - 2.528 = 0.377 amperes. This computation shows that a small increase in allowable DOD (or a small decrease in the eclipse time) results in a significant increase in the science allocation. Increasing the maximum allowable DOD from 60 to 65% is well within the capabilities of the batteries.

4.3.5.1.8 Solar Panel Surplus Power. All solar panel contingency values, shown in Table 4.3.5.1-1, specify the surplus current at 28.0 volts. The fact that a surplus exists forces the solar panel voltage upward above 28 volts. If the shelf thermostatic switches are off, the voltage limiters will clamp the power buses at 29.5 volts. If the solar panel is relatively cold and operating over the constant current portion of the solar panel V/I curve, the voltage limiter current at 29.5 volts will be equal to surplus current available at 28 volts. However, if the panel is relatively warm (i.e., near Venus), the operating point will be on the knee of the curve (see Section 4.3.5.3). This means that when the voltage is driven upwards, due to available surplus current, the solar panel current will decrease and the voltage limiter current will be somewhat less than indicated in Tables 4.3.5.1-1. If the surplus at 28 volts is small, there may not be enough current available to drive the panel voltage as high as 29.5 volts. The quiescent stable voltage will then be some value between 28.0 and 29.5 volts and there will be no voltage limiter current.

However, since the solar panel power/current values shown in Table 4.3.5.1-1 are conservative minimum values, it is anticipated that there will be sufficient current available at 29.5 volts to force the voltage limiters into conduction during all illuminated phases, with the possible exception of short time periods during periapsis passes when the solar panel temperature approaches 130°C.

4.3.5.2 Use of Propulsion and Shelf Heaters. Section 3.2 provides a detailed description of the thermal design, including the propulsion heaters.

The propulsion tank heater group is on continuously. At launch the primary group is selected by ground command. Failure of any one of 12 heaters in this group can be determined by temperature sensors and/or special high resolution current sensors (described in Section 3.8.3.8). If a failure occurs, the redundant secondary group can be selected by ground command. Either the primary or redundant tank group heaters are on at all times. They cannot be commanded off. The latch valve (VLVPHS) and fill and drain (VPDVHS) heater current sensors monitor only the primary heaters. If the secondary group of heaters is selected, these two TM signals will read zero.

The radial thrusters and line heaters are on continuously also - they cannot be command off.

The axial thruster and line heaters are commandable, and should be ON at launch.

The three Orbit Insertion Motor (OIM) case, throat, and safe and arm (S&A) heaters are connected in parallel. Heater 1 group is turned on at launch; heater 2 group consists of identical backup heaters.

Operation of the equipment shelf heaters is also described in Sections 3.2.2.1.2 and 3.8.2.3. The Orbiter battery shelves will cool down after launch and the battery shelf heaters will be actuated. Excess solar panel capacity will be

dissipated in the battery shelf heaters during the first few months of cruise. Both battery and RF shelves will cool down during the long apoapsis eclipse periods and will actuate all of the shelf thermostatic switches. Following a long eclipse, the shelf heaters will absorb all excess solar panel capacity for several hours until the shelves reach 55 ±5°F.

The shelf heaters will also come on automatically if the shelf temperatures reach excessively low values due to an undervoltage/overload switch trip.

4.3.5.3 Solar Panel Performance by Mission Phase. The Orbiter solar panel performance is summarized by the voltage/current curves shown in Figure 4.3.5.3-1 and 4.3.5.3-2. Panel sizing is determined by near Venus loads and also by cruise loads when the solar intensity is minimum at L + 48 days. The solar panel has a sufficient number of parallel strings to provide the required current under minimum illumination conditions at L + 48 days and also has a sufficient number of series cells to provide the required voltage under Venus orbit temperatures when operating near the knee of the voltage/current curve.

The curves shown in Figure 4.3.5.3-1 represent minimum predicted output and assume worst case solar flare degradation. The curves shown in Figure 4.3.5.3-2 represent maximum power output with no radiation degradation. The actual flight performance will probably be somewhere between these extremes. With a modest or average amount of solar flare activity, panel performance should be near the maximum curves.

Near Earth, the open circuit voltage of the panel is relatively high and, over the normal operating range of 28.0 to 29.5 volts, the panel is operating under constant current conditions. As the spacecraft moves closer to the sun and the illumination intensity increases, the current at the lower voltages also increases. However, the increasing temperature results in a decrease in the open circuit voltage and the operating range

now slides down to the knee of the curve. In order to guarantee that its operating point does not slide beyond the knee down to the steep portion of the curve, the panel is designed with a sufficient number of series cells to provide adequate output even if the solar panel temperature is 9°F above the predicted value.

During Venus orbital operations, the panel will be subjected to transient thermal conditions at various times. When the spacecraft is distant from the planet, in the spacecraft's elliptical orbit, and is not receiving any significant amount of albedo illumination from Venus, the panel temperature will be near 158°F at 1.93 effective Earth suns. As the spacecraft nears Venus, albedo illumination initially increases the power available. After the effective illumination level becomes greater than 3.0 effective Earth suns, the panel temperature rises to values exceeding 198°F and the open circuit voltage of the panel decreases to the point where the spacecraft loads operate considerably beyond the knee of the panel voltage/current curve. During the apoapsis eclipse season, the illumination intensity is expected to reach 3.4 Earth suns for a few minutes when the spacecraft is at its periapsis point and the panel may reach temperatures as high as 266°F. At this temperature, the panel will no longer be able to support the spacecraft loads by itself and the batteries will come on line for a few minutes to share the load.

The panel will also be subjected to transient thermal conditions during the eclipses. When the solar panel temperature rises to approximately +50°F after an eclipse, its output over the spacecraft operating voltage range reaches a peak of approximately 16-17 amperes. During periapsis eclipses, this poses no problems because the spacecraft loads are approximately 10 amperes and the bus voltage limiters can easily dissipate the excess panel current. However, when the spacecraft passes through this operating region after an apoapsis eclipse, the loads are only 3.5 to 4.0 amperes and the voltage limiters will have

to dissipate approximately 13 amperes. To minimize the burden on the limiters (especially if one of them has failed), it is important that battery charging is commanded to the high charge rate before entering apoapsis eclipse.

4.3.5.4 Battery Management. Battery charging will be commanded to low rate charge state at launch and remain in this configuration approximately 7 months until the periapsis eclipse season. As shown by the curves in Section 3.8.3.4, the trickle charge rate that will exist in the low rate charge state will decrease somewhat as illumination intensity and panel temperature increase. The trickle charge rate will vary from approximately 90 to 200 ma (C/38 to C/85). Although charging efficiency at these low rates is very poor (approximately 50%), this is not important because during the continuous illumination periods, it is only necessary to replace the self discharge stand loss.

In order to stay in energy balance when the periapsis eclipses exceed 10 minutes, it will be necessary to go to the high rate charging state (0.65 amps or C/12) for a few hours every orbit. To prevent overheating, the amount of high rate charge must be time limited. Maximum battery depth of discharge (DOD) during the periapsis eclipse season will be approximately 30% (2.25AH per battery) when the eclipse time reaches a maximum of 24 minutes. Charging efficiency at C/12 is in the range of 80 to 90%. Use Table 4.3.5.4-1 to estimate the approximate amount of high rate energy needed.

During the long apoapsis season maximum battery (DOD) will be approximately 60% when the total eclipse time reaches a maximum of 3.8 hours. At this discharge level it will require approximately 10 hours to recharge at high rate. If the battery temperature does not exceed 85°F under continuous high rate charge during the apoapsis eclipse season, the charge controller can be left in high rate charge during the entire short season.

Table 4.3.5.4-1 summarizes Orbiter mission battery charging management:

TABLE 4.3.5.4-1

ORBITER BATTERY CHARGE MANAGEMENT

MISSION PHASE	RECOM-MENDED CHARGE RATE	RECOMMENDED START TIME FOR THE RECOMMENDED CHARGE RATE	COMMENTS
Cruise	Low	Continuous	Command to low rate during pre-launch.
Pre-orbit Insertion	High	Approximately 1 day before re-orientation for orbit insertion. Charge at high rate until roll-over.	Approximately 2 hours; this maximizes capacity for pyro firings (OIM and science).
Orbit Operation (continuous illumination)	Low	Continuous	
Post Periapsis Eclipse (<10 min eclipse time)	Low	Continuous	Low DOD
Post Periapsis Eclipse (>10 min eclipse time)	High	Post eclipse until roll-over.	Approximate time = 1.2 x AH removed ÷ charge rate.
	Low	Balance of orbit until eclipse.	Transfer to high charge during or right after eclipse.
Pre-Apoapsis Eclipse	High	Approximately 1/2 hour before eclipse.	
Post-Apoapsis Eclipse	High	Post eclipse until roll-over.	Approximate time = 1.2 x AH removed ÷ charge rate.
	Low	Balance of orbit until 1/2 hour before next eclipse	

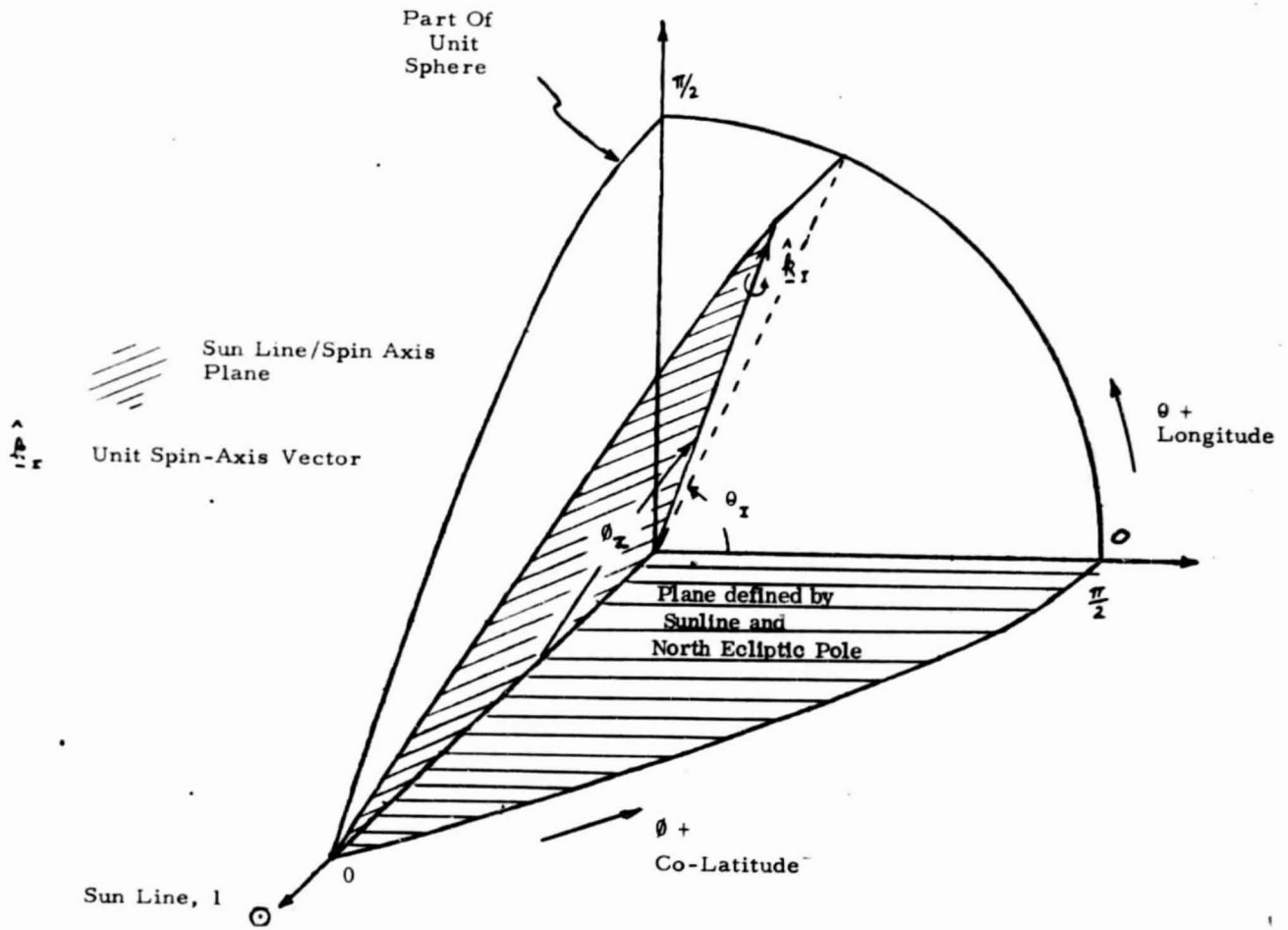


Figure 4.3.1.4.1-1. Spacecraft-Centered Inertial Frame and Definitions of Angles Ø and θ

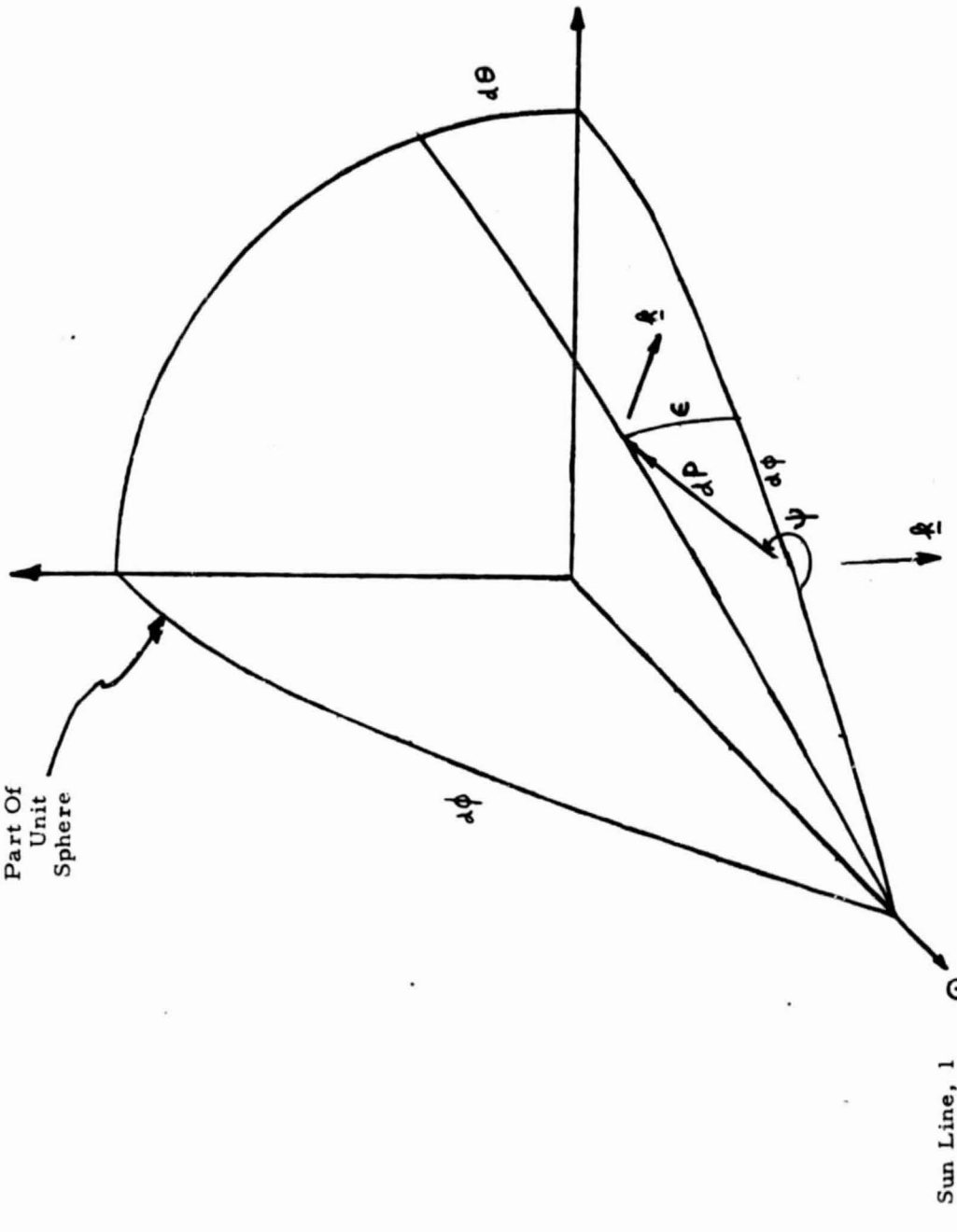


Figure 4.3.1.4.1-2. Schematic of Infinitesimal Precession of Spin Axis

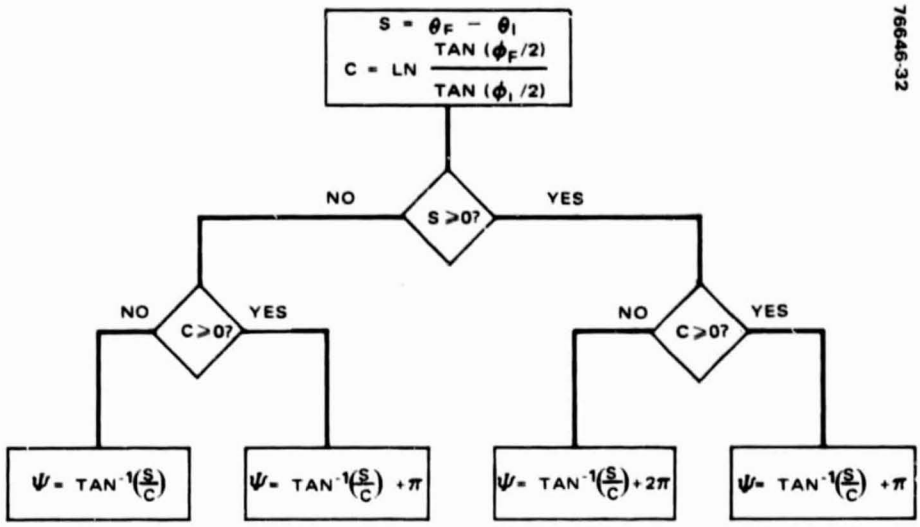
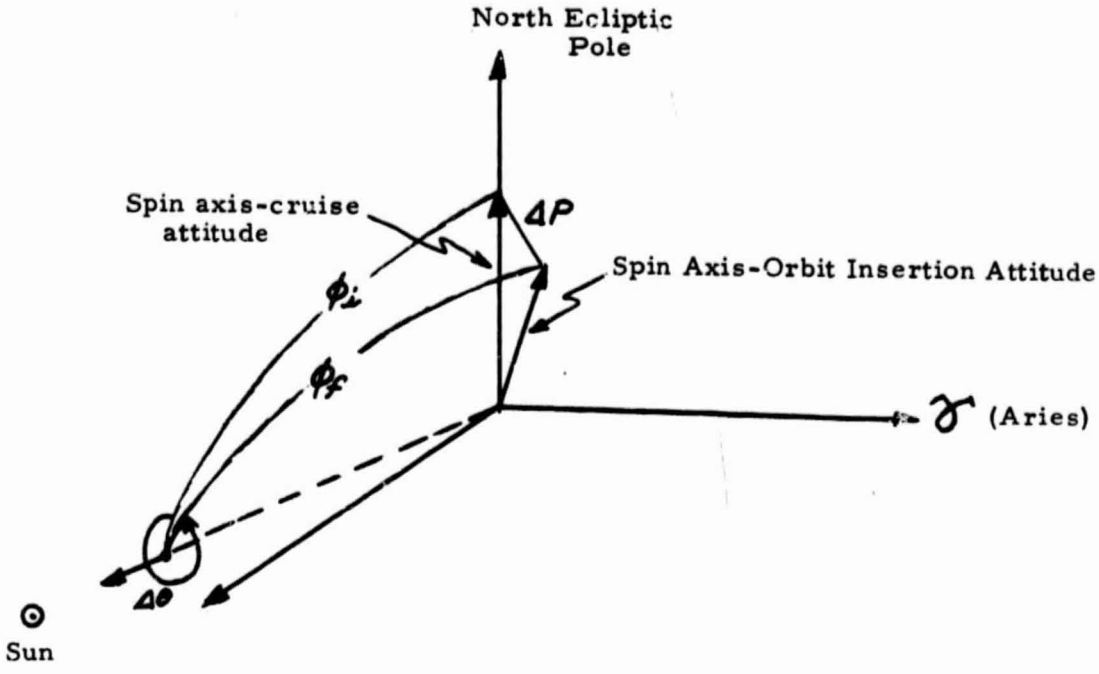
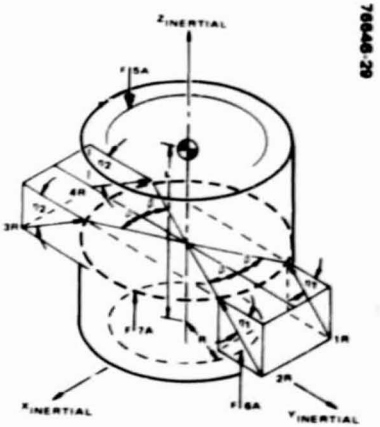


FIGURE 4.3.1.4.2-1. LOGIC FOR DETERMINING PHASE ANGLE, Ψ



Vector	Celestial Latitude (Deg.)	Celestial Longitude (Deg.)
Sunline	-0.6	264
Spin Axis - Initial Cruise Attitude	90	Arbitrary
Spin Axis - Orbit Insertion Attitude	65	348

Figure 4.3.1.4.2-2. Attitude Geometry for Orbit Insertion Reorientation



AXIAL ΔV AND PRECESSION MANEUVERS

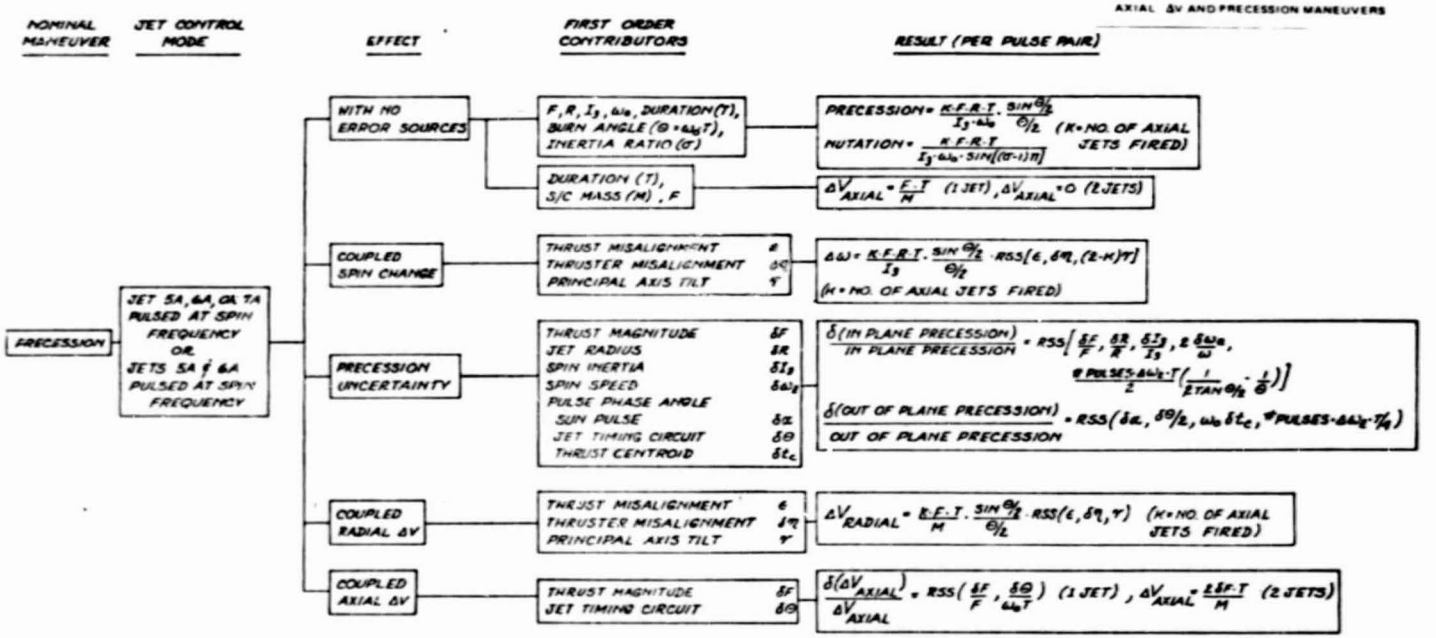


FIGURE 4.3.1.4.2.3-1. PRECESSION MANEUVER ALGORITHMS

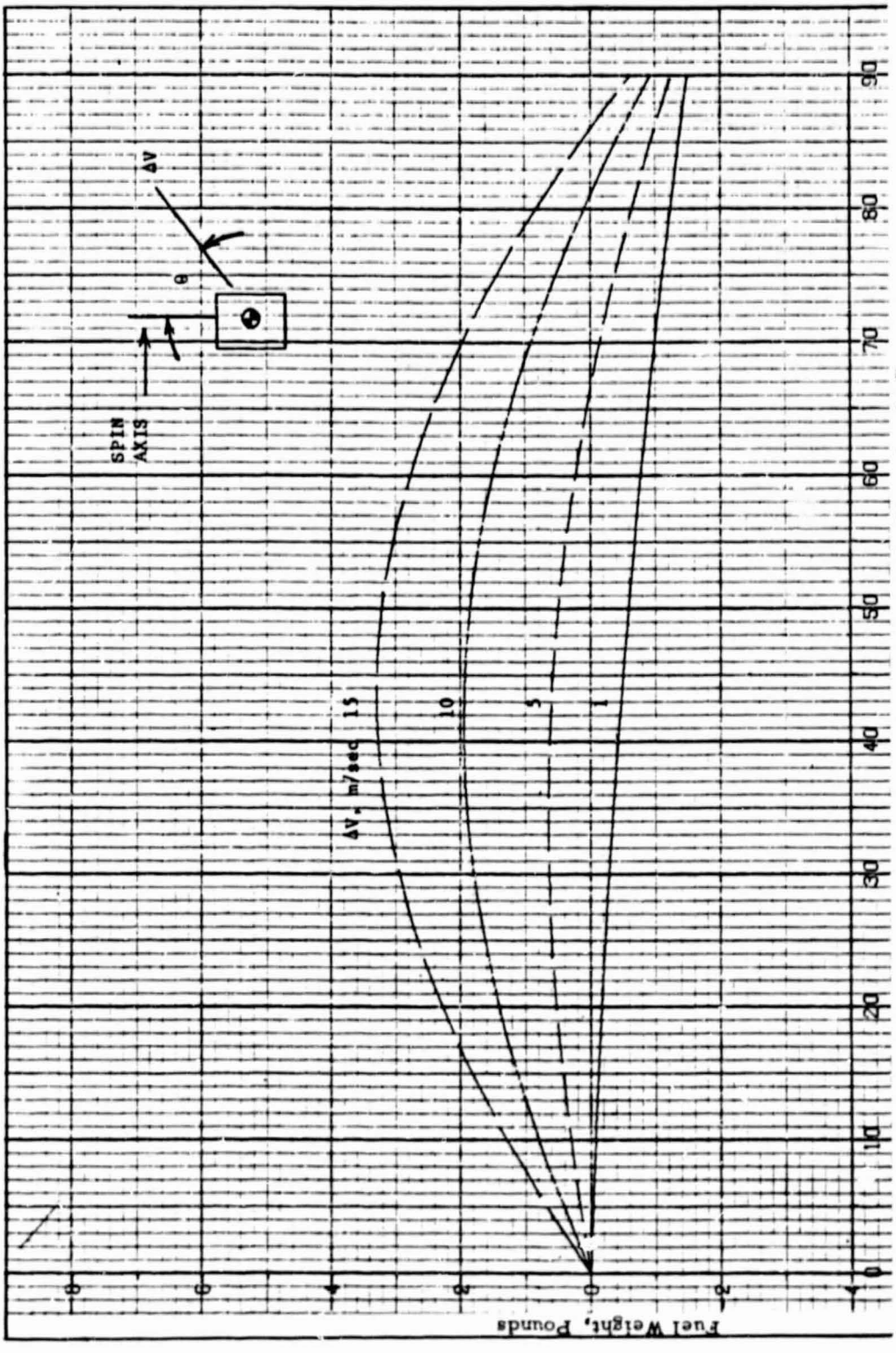


Figure 4.3.1.5.1-1. Fuel Weight Penalty Incurred By Use Of Vector Mode Verses Axial Mode.

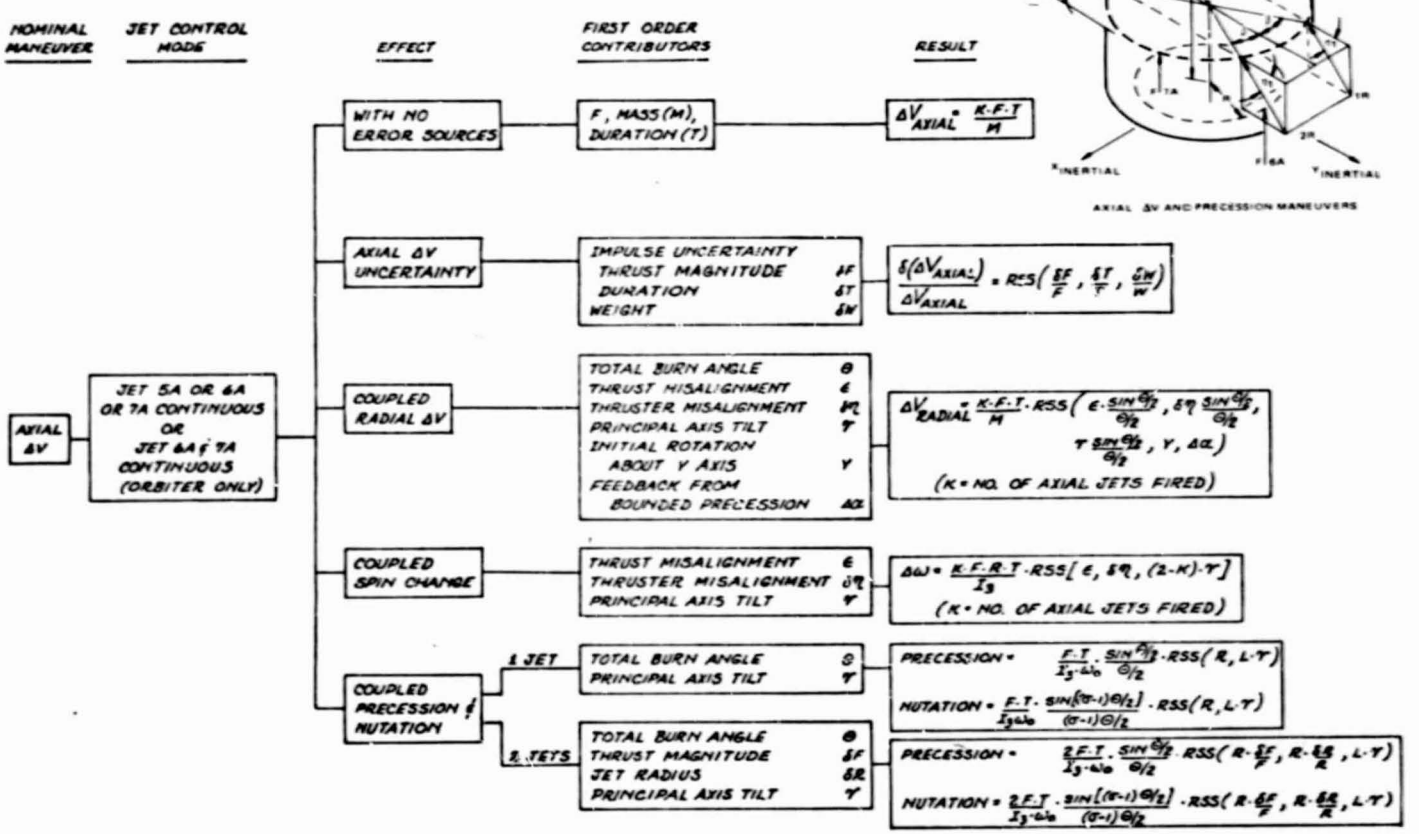
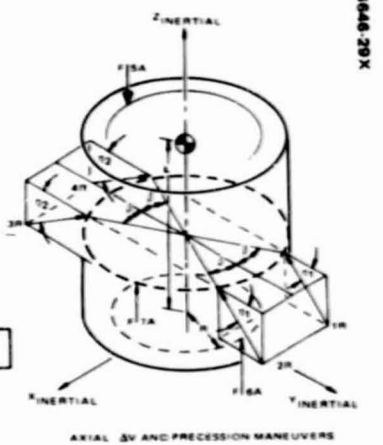
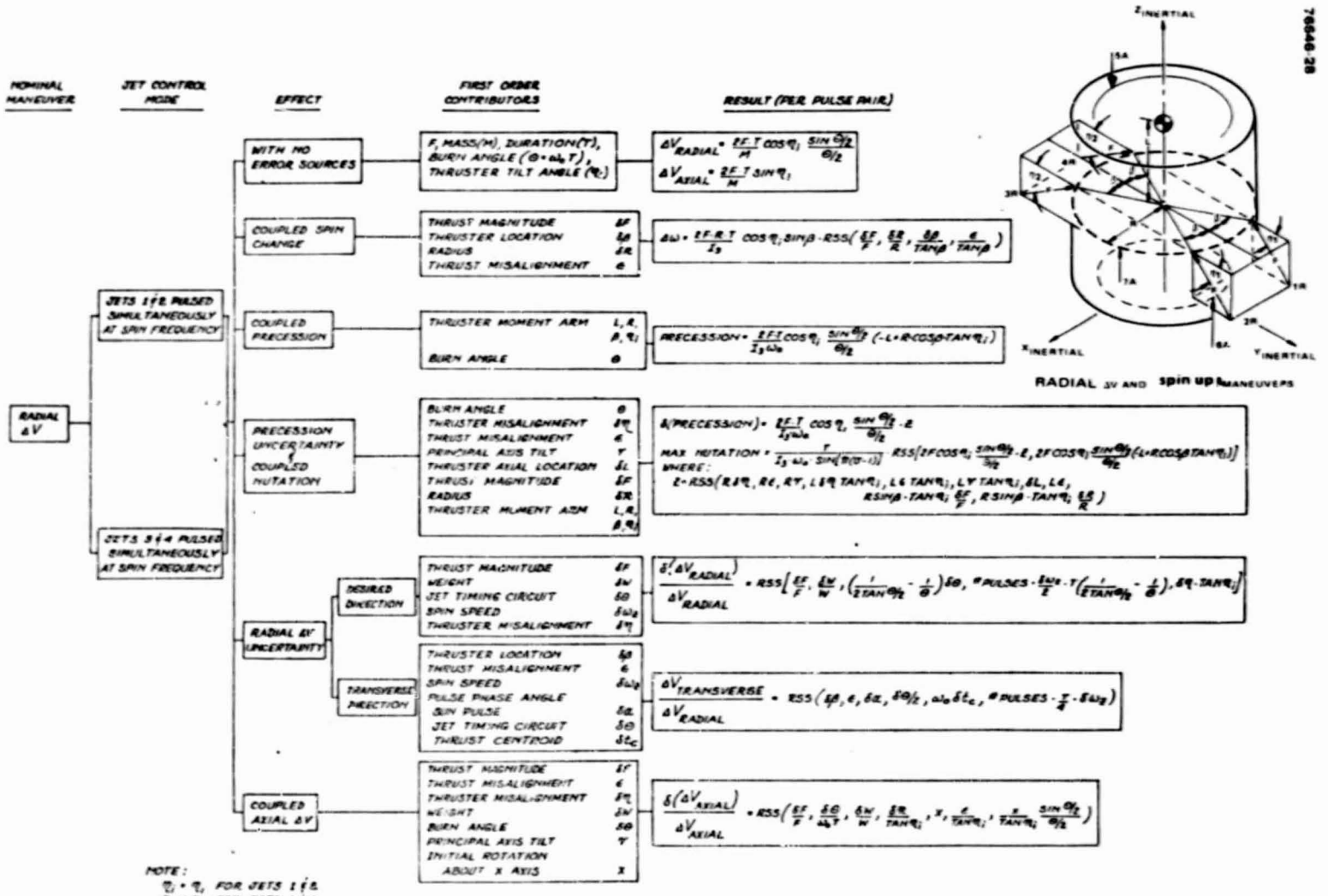


FIGURE 4.3.1.5.2.2-1. AXIAL ΔV MANEUVER ALGORITHMS



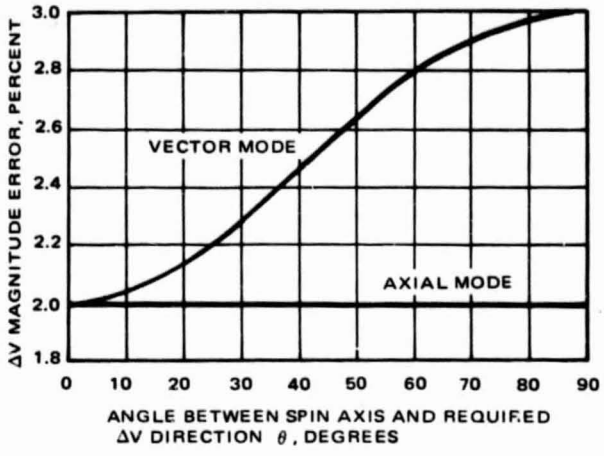
AXIAL ΔV AND PRECESSION MANEUVERS

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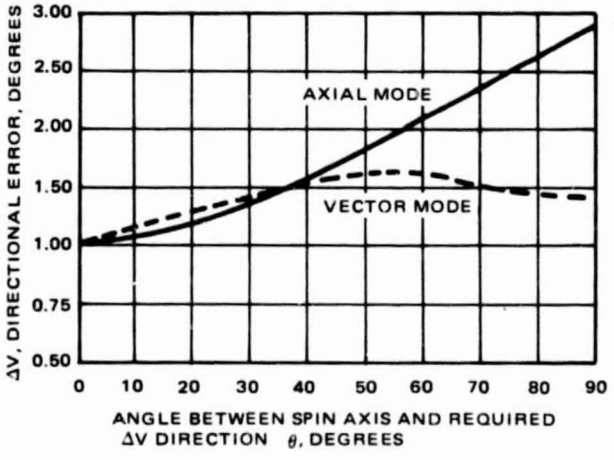


RADIAL ΔV AND spin up MANEUVERS

FIGURE 4.3.1.5.2.2-2. RADIAL ΔV MANEUVER ALGORITHMS

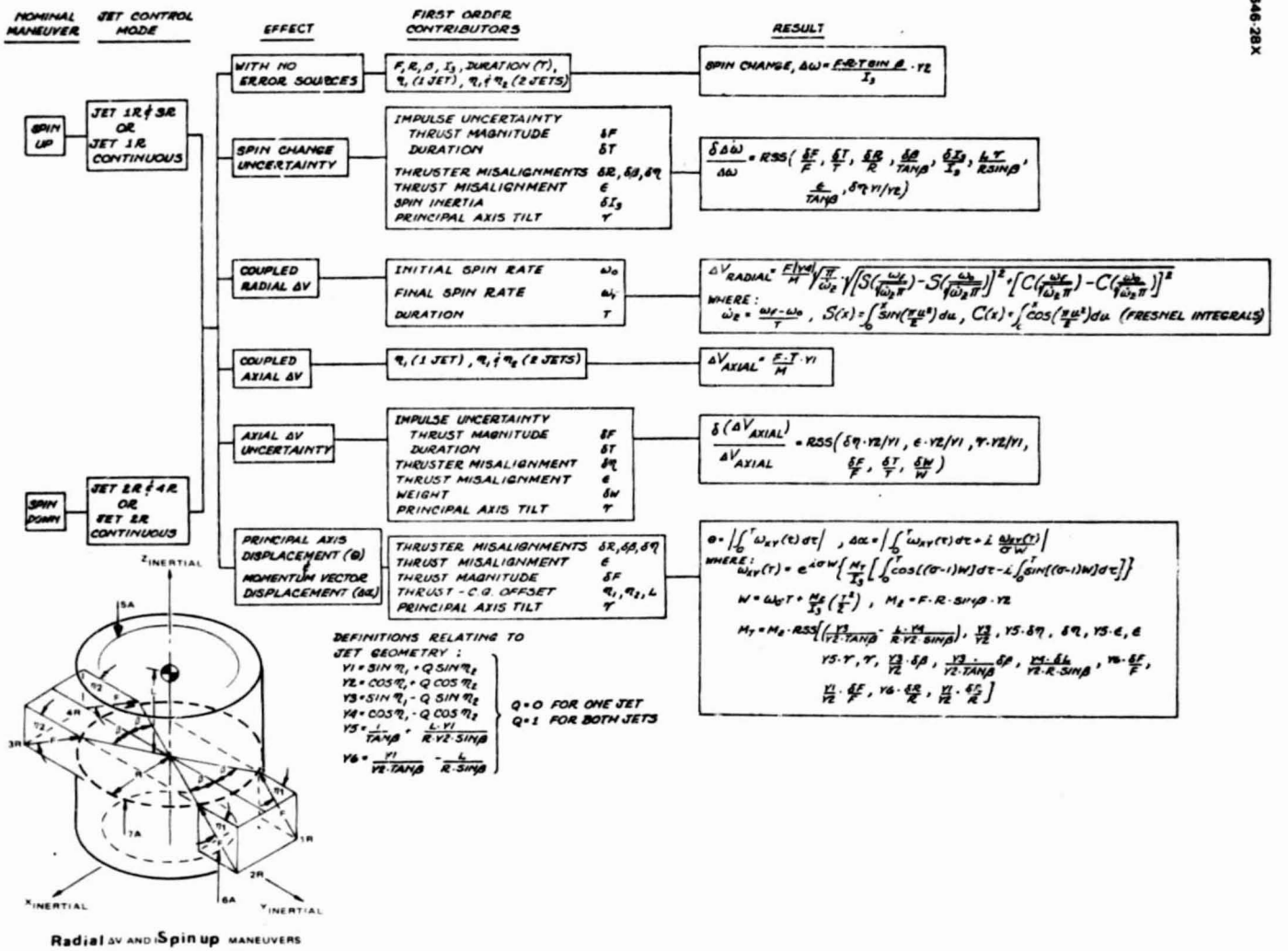


a) MIDCOURSE ΔV MAGNITUDE ERROR



b) MIDCOURSE ΔV DIRECTIONAL ERROR

FIGURE 4.3.1.5.2.2-3. ΔV NOMINAL EXPECTED ERRORS IN MAGNITUDE AND DIRECTION



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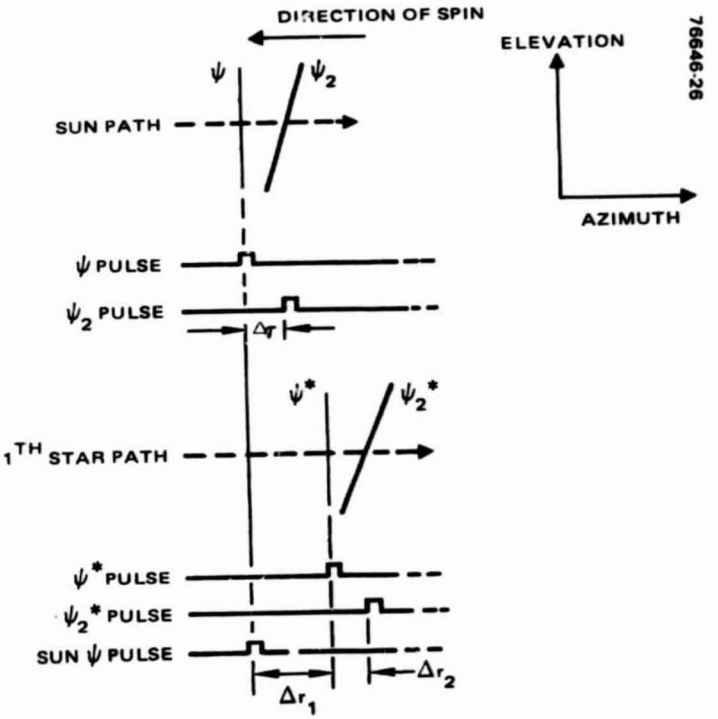
FIGURE 4.3.1.6.2.2-1. SPIN CHANGE MANEUVER ALGORITHMS

ATTITUDE SENSOR CONFIGURATION

- REDUNDANT DUAL SLIT SUN SENSORS
- SINGLE DUAL SLIT STAR SENSOR, INDEPENDENT ELECTRONIC ASSEMBLIES
- TIME INTERVALS BETWEEN SUN AND STAR PULSES DEFINE ROLL RATE, ASPECT ANGLES, AND SPIN ANGLES

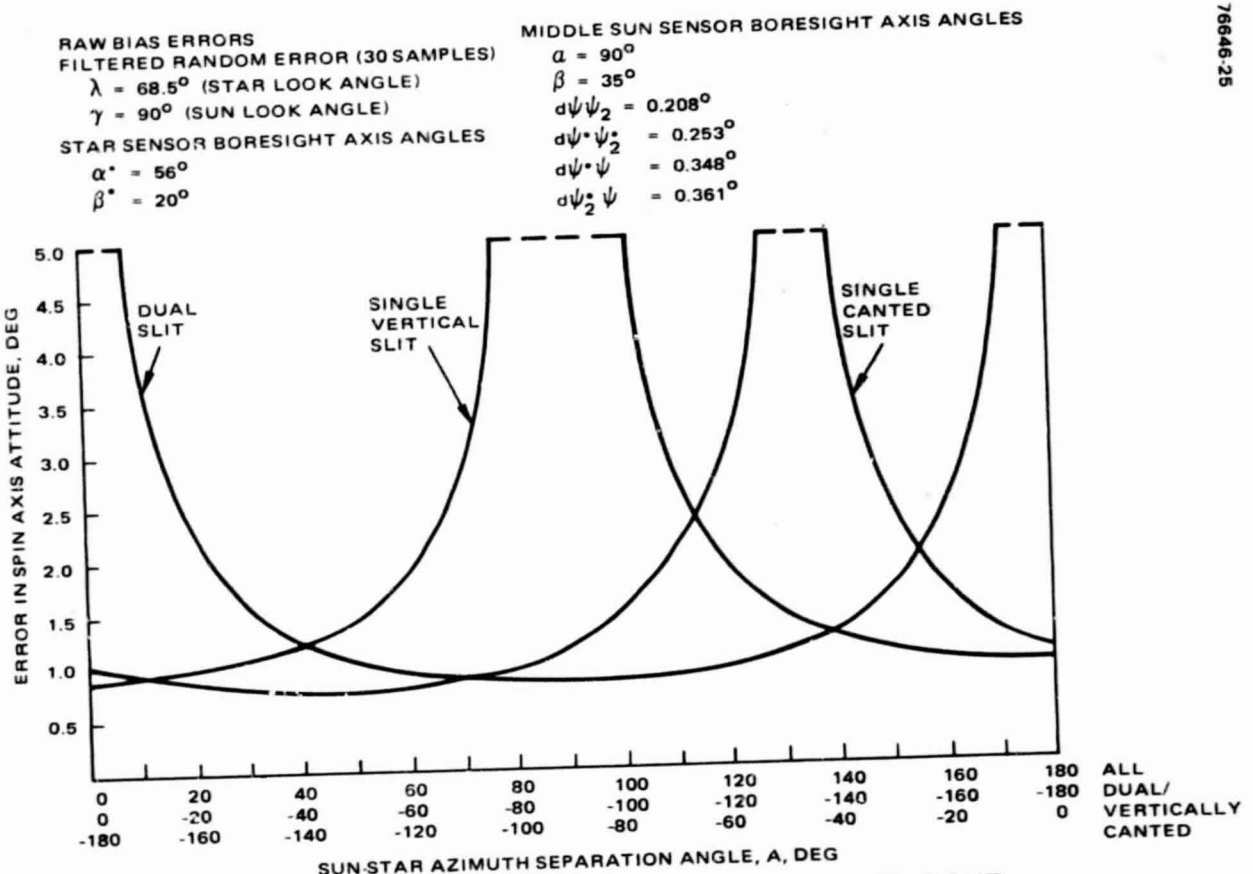
DATA TYPES

- PRIMARY: $\psi\psi_2$, $\psi\psi^*(i)$, $\psi\psi_2^*(i)$
- SECONDARY: $\psi^*(i)\psi_2^*(i)$, $\psi^*(i)\psi^*(j)$, $\psi_2^*(i)\psi_2^*(j)$, $\psi^*(i)\psi_2^*(j)$



NOTE: $\Delta\tau$ MEASURES SUN ASPECT ANGLE γ
$\Delta\tau_1$ MEASURES SUN-STAR ROLL ANGLE A_1
$\Delta\tau_2$ MEASURES STAR ASPECT ANGLE λ_1

FIGURE 4.3.2.3-1. ATTITUDE SENSORS AND DATA TYPES



76646-25

FIGURE 4.3.2.3-2. ATTITUDE DETERMINATION ERROR VS SENSOR SLIT USAGE AND GEOMETRY

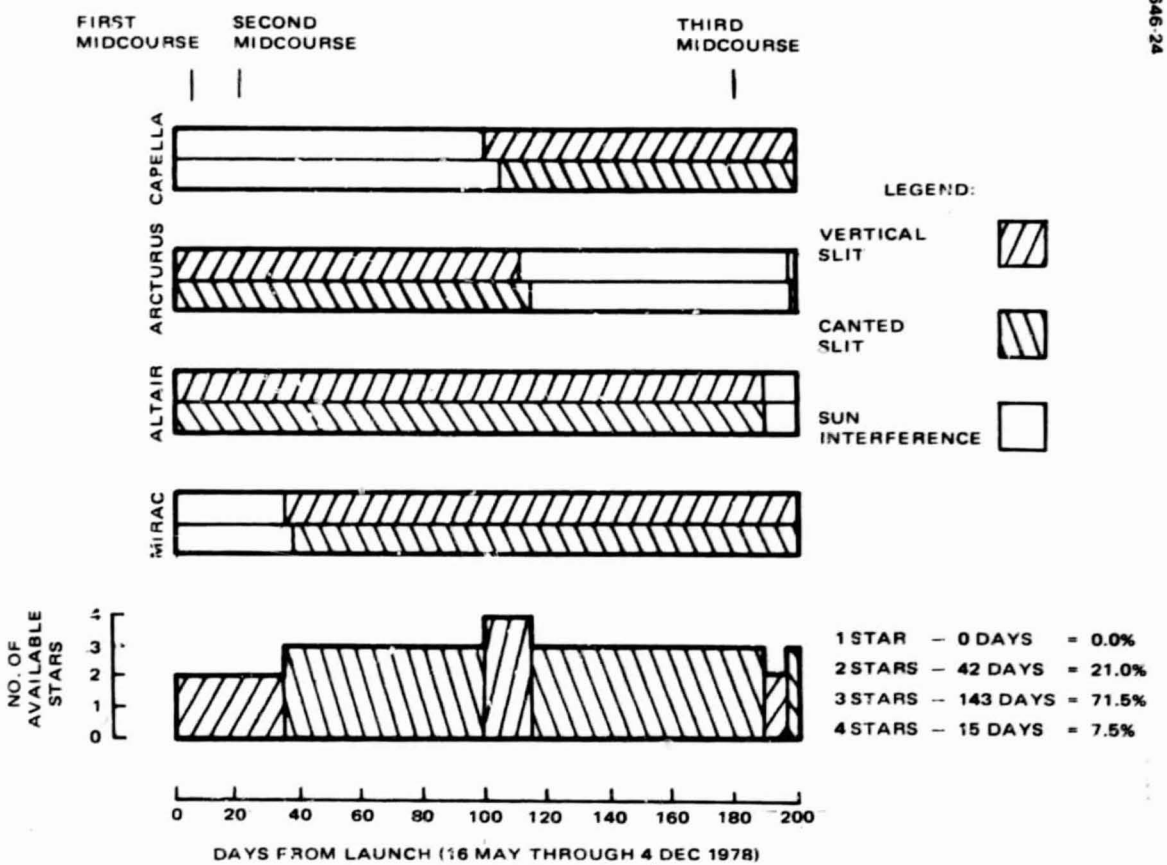


FIGURE 4.3.2.3-3. ORBITER CRUISE STAR AVAILABILITY – NORTHERN HEMISPHERE

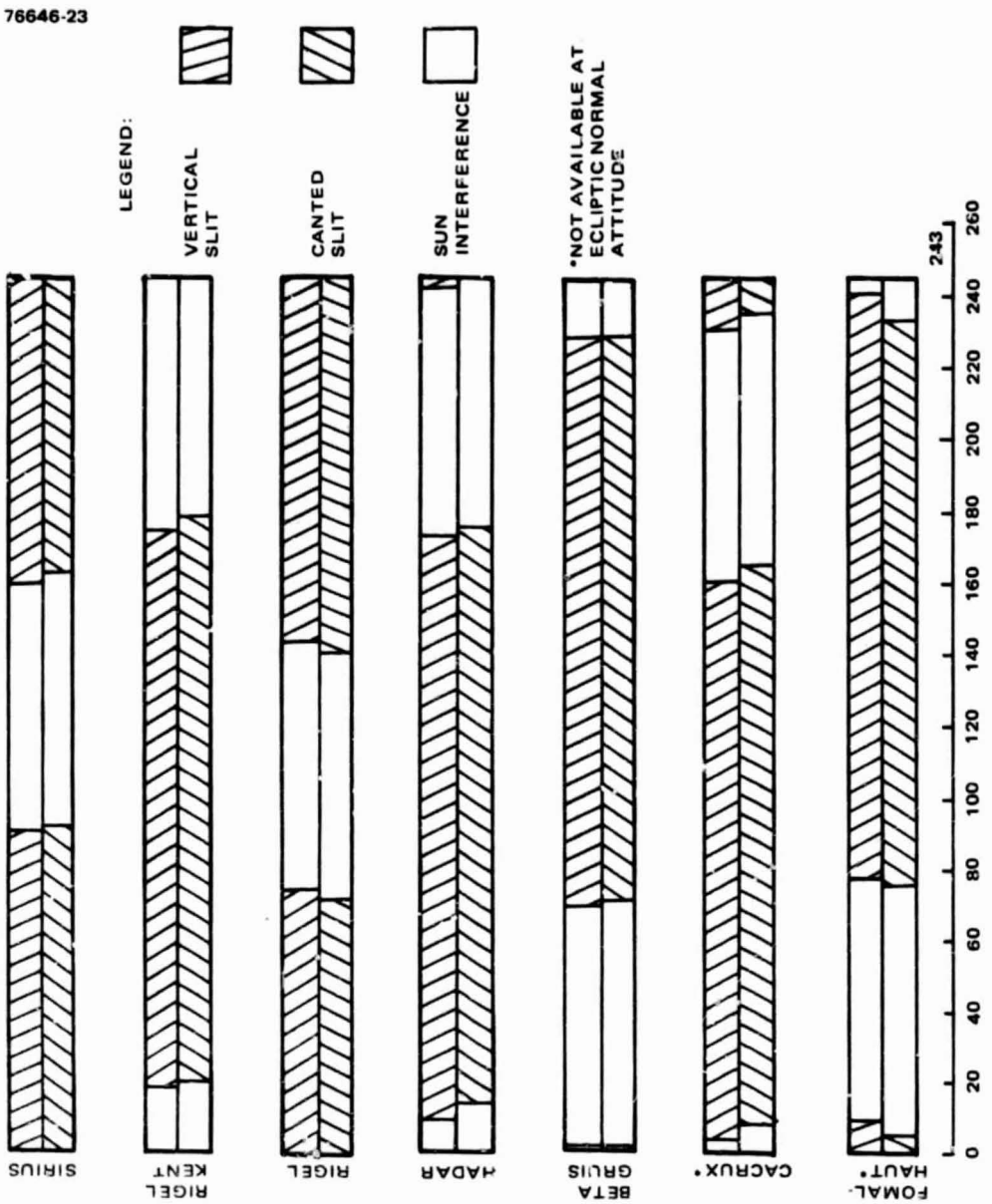


FIGURE 4.3.2.3-4. ORBITER ORBIT STAR AVAILABILITY – SOUTHERN HEMISPHERE (APOAPSIS)

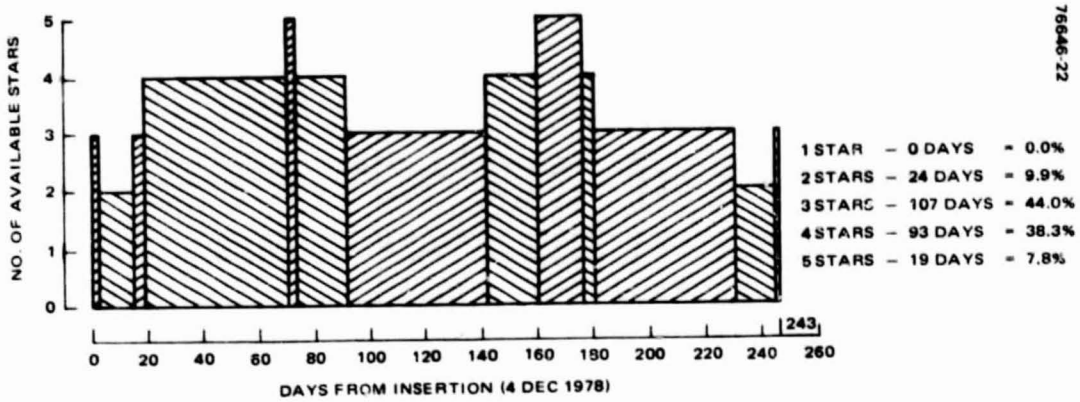


FIGURE 4.3.2.3-5. ORBITER ORBIT – SOUTHERN HEMISPHERE (APOAPSIS)
ECLIPTIC NORMAL ATTITUDE

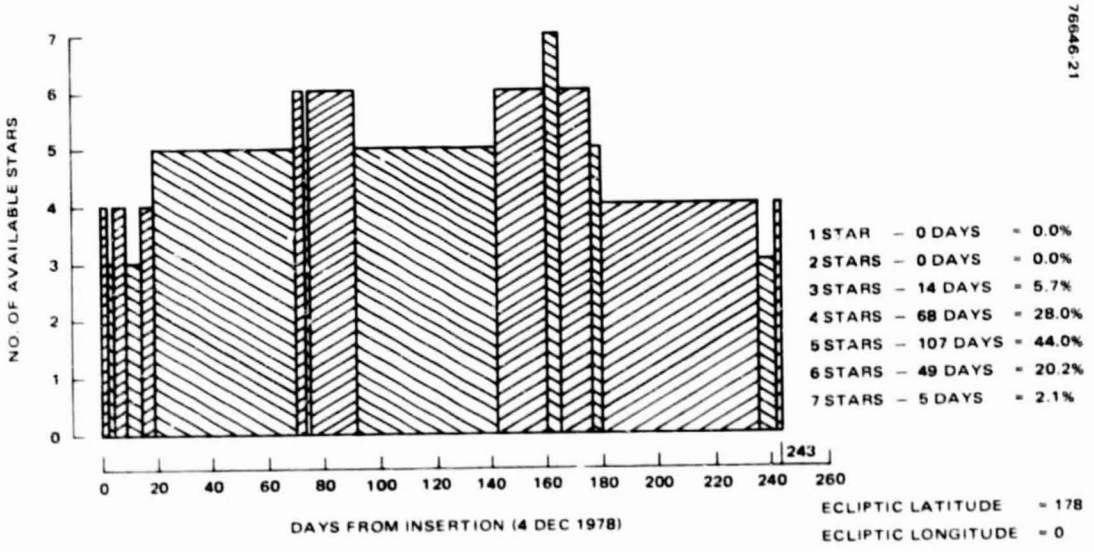


FIGURE 4.3.2.3-6. ORBITER ORBIT – SOUTHERN HEMISPHERE (APOAPSIS)

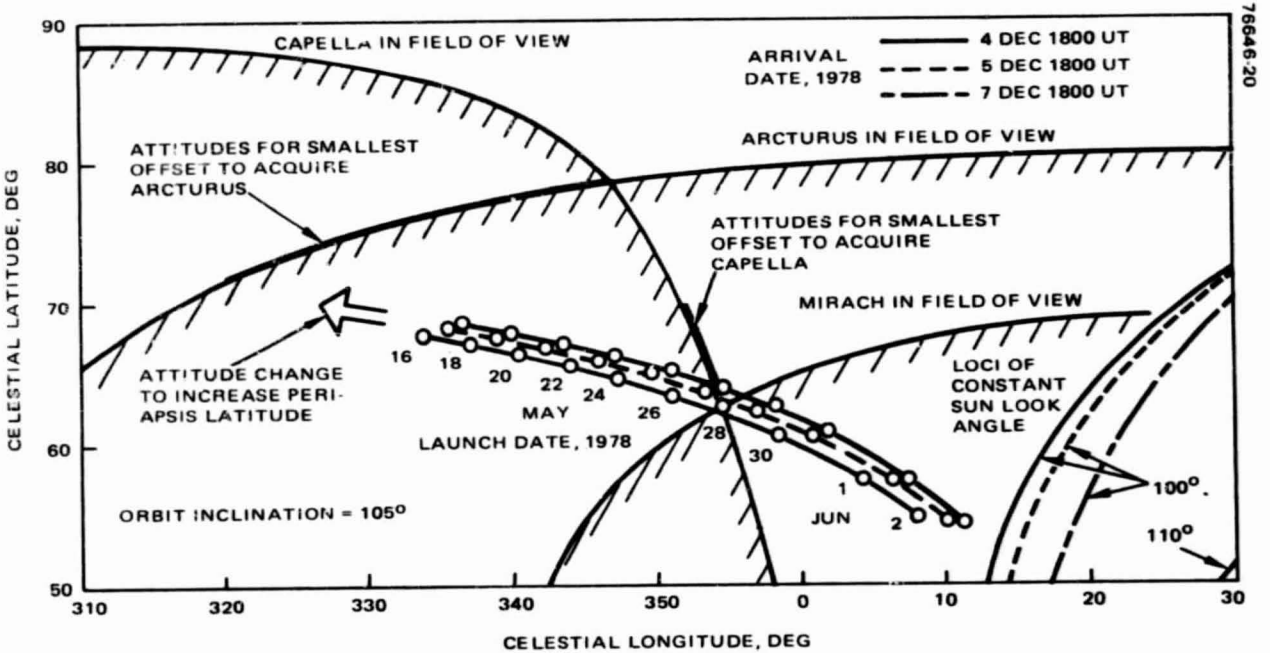


FIGURE 4.3.2.3-7. NOMINAL ORBIT INSERTION ATTITUDES AND STAR AVAILABILITY

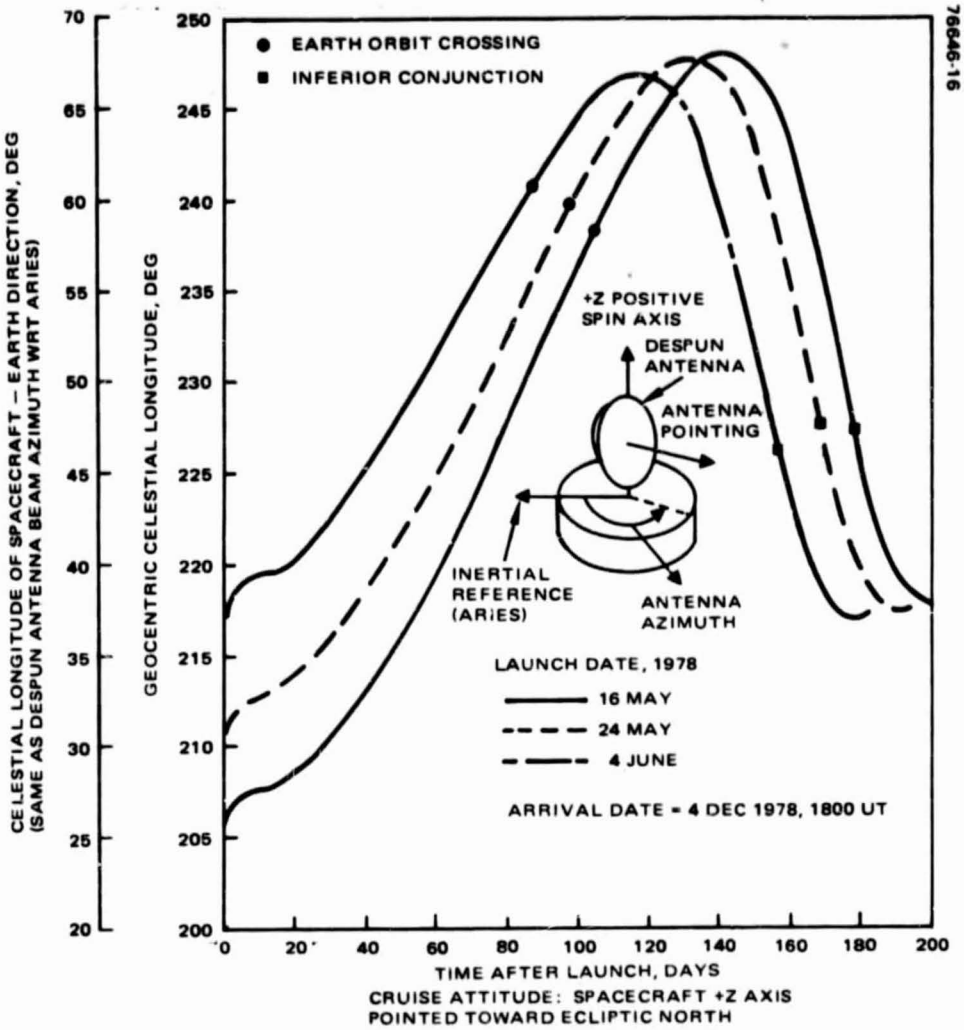


FIGURE 4.3.4.2-1. EARTH AZIMUTH POINTING DIRECTION DURING CRUISE

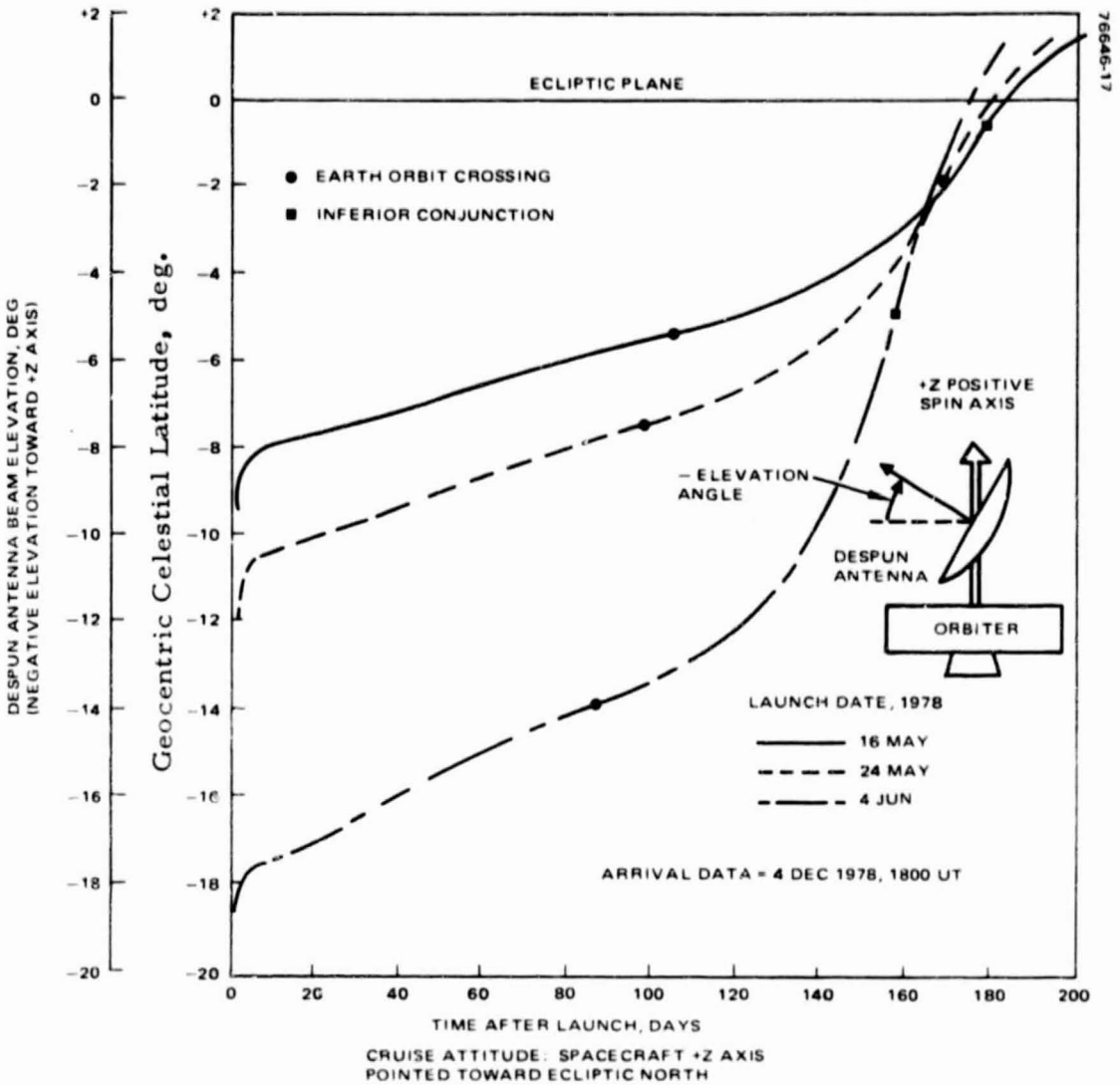


FIGURE 4.3.4.2-2. EARTH ELEVATION POINTING DIRECTION DURING CRUISE

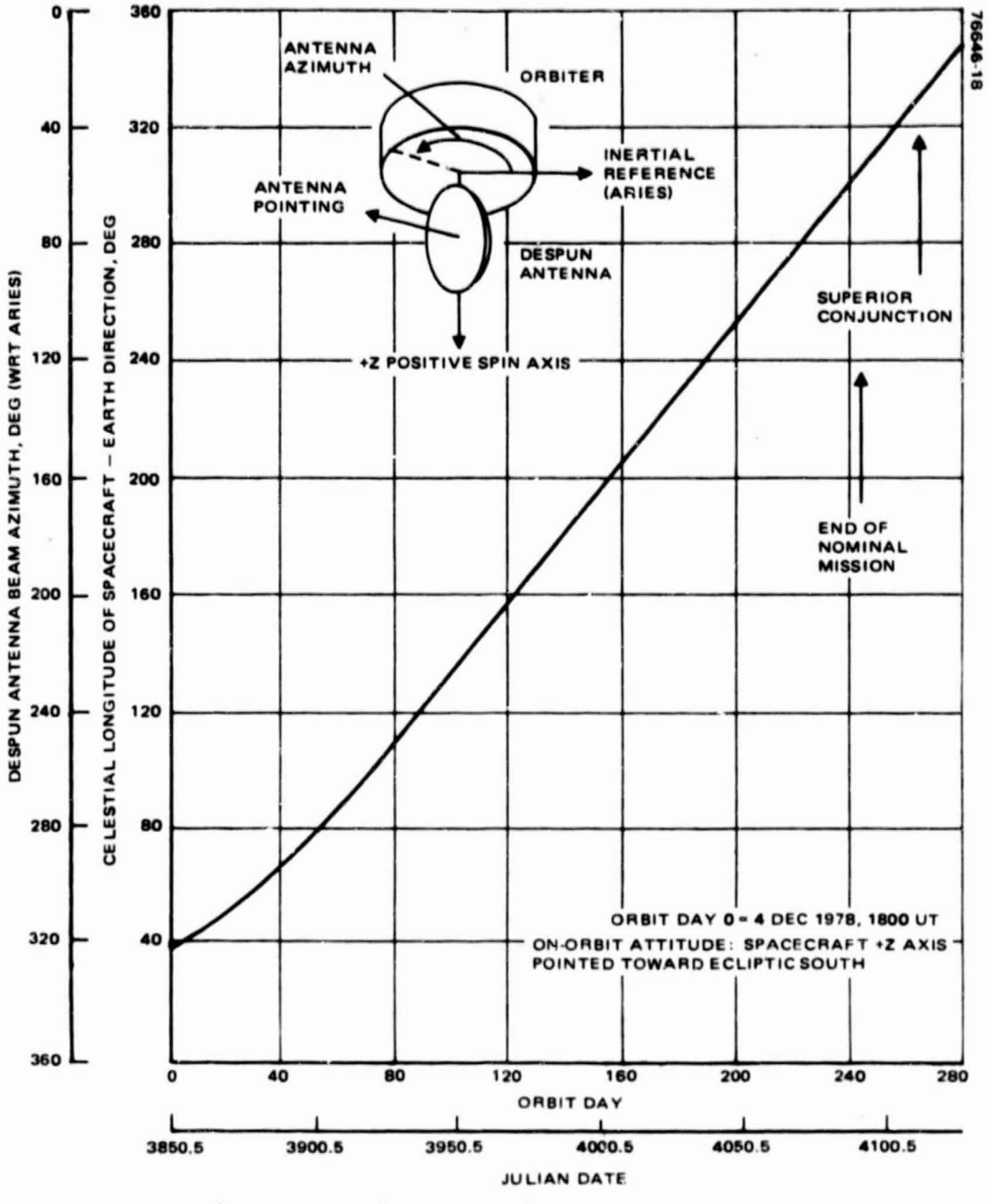
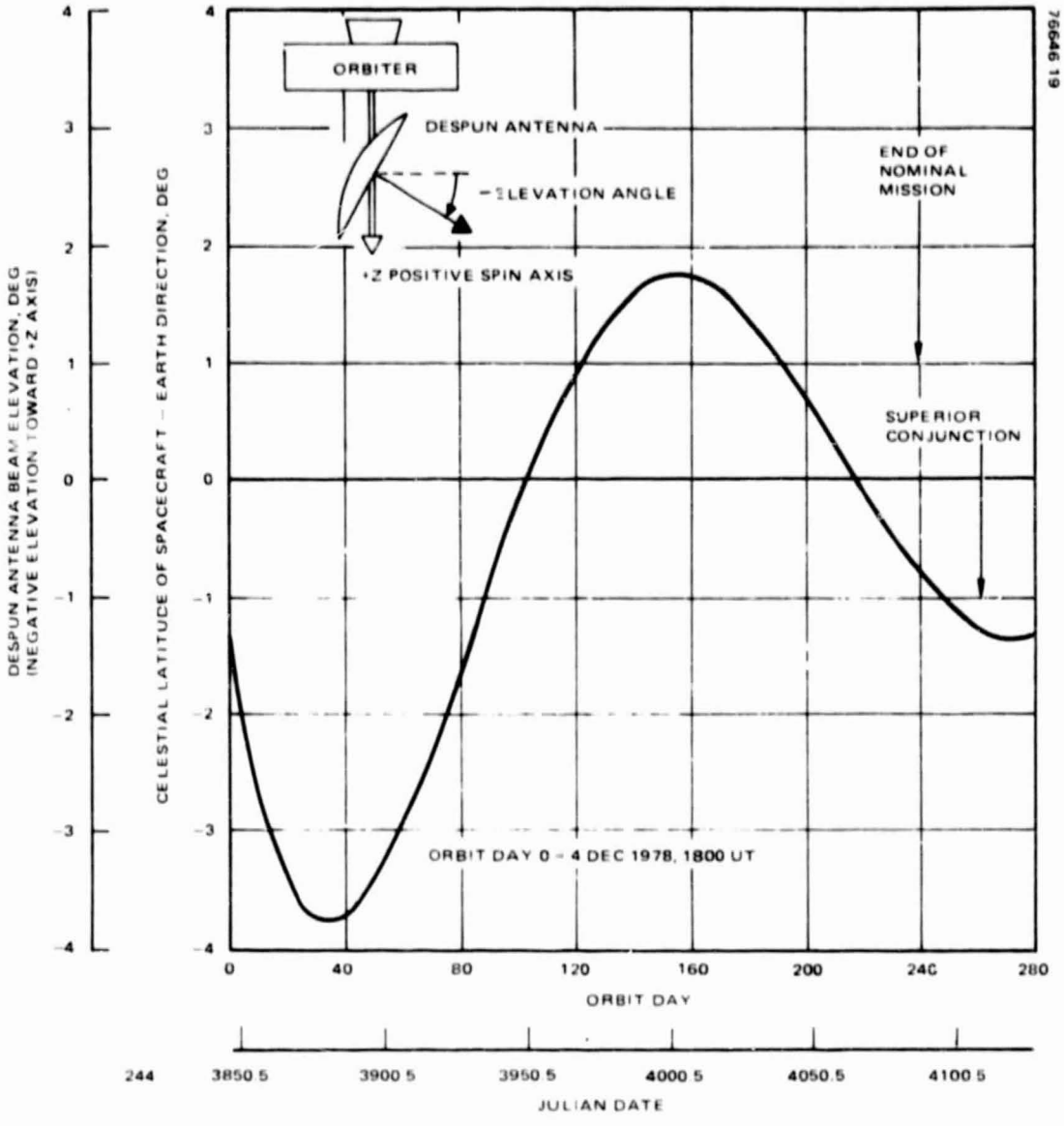


FIGURE 4.3.4.2-3. EARTH AZIMUTH POINTING DIRECTION ON ORBIT



ON ORBIT ATTITUDE SPACECRAFT +Z AXIS POINTED TOWARD ECLIPTIC SOUTH

FIGURE 4.3.4.2-4 EARTH ELEVATION POINTING DIRECTION ON ORBIT

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OF POOR QUALITY

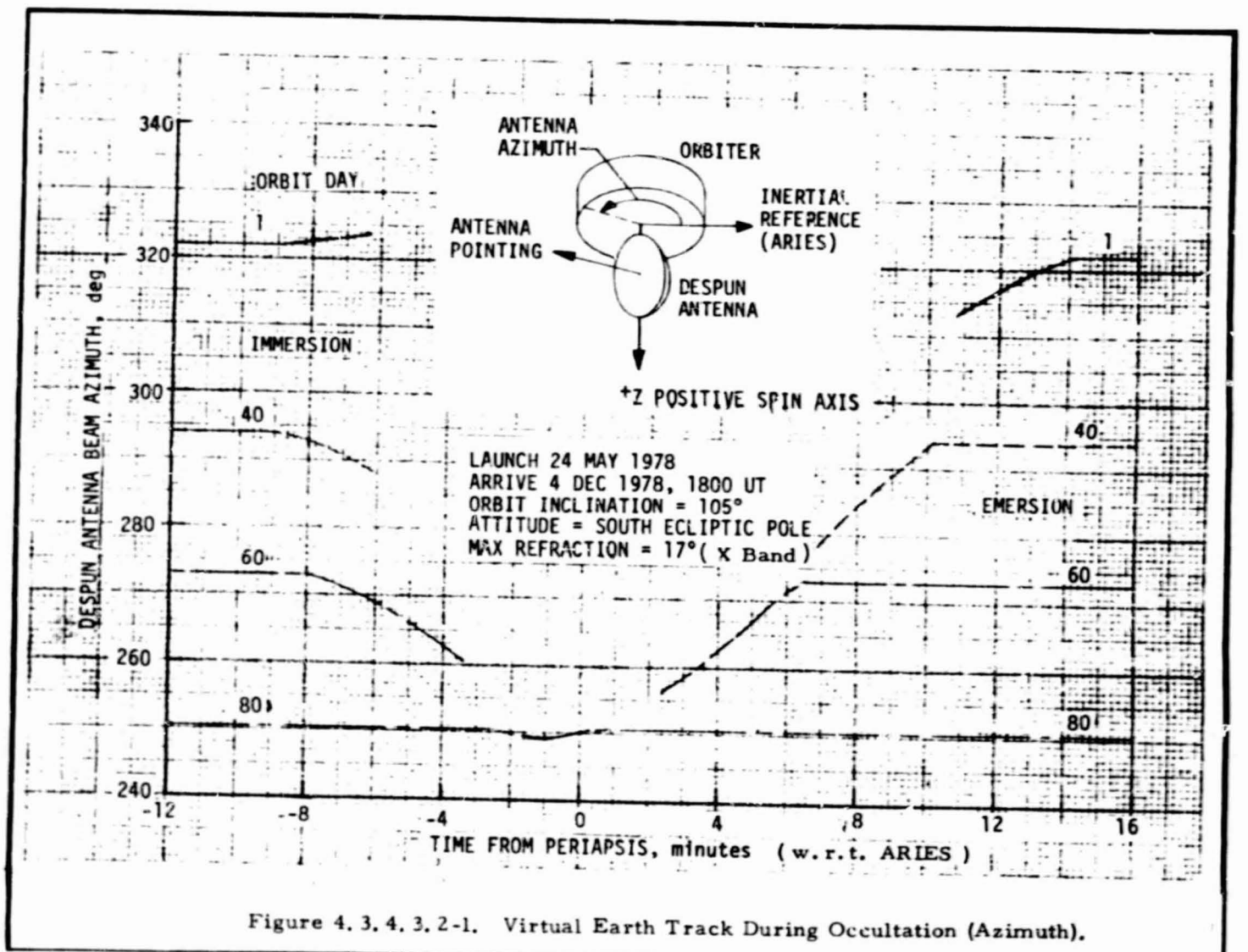


Figure 4.3.4.3.2-1. Virtual Earth Track During Occultation (Azimuth).

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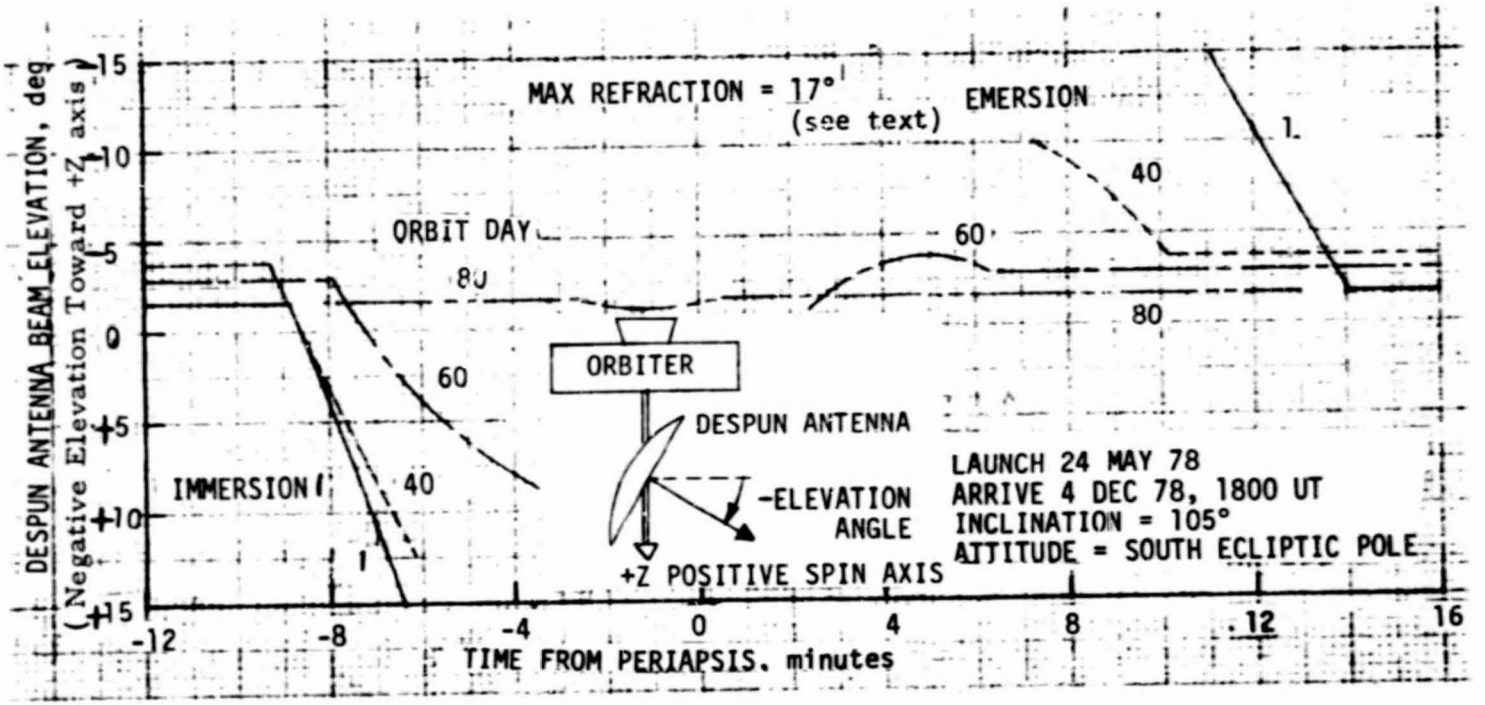


Figure 4.3.4.3.2-2. Virtual Earth Track During Occultation (Elevation)

Figure 4.3.4.3.2.5-1.
DESPUN DISH AZIMUTH CONTROL FOR OCCULTATION POINTING
EXAMPLE USING FIVE LINEAR SEGMENTS ON ORBIT DAY 50

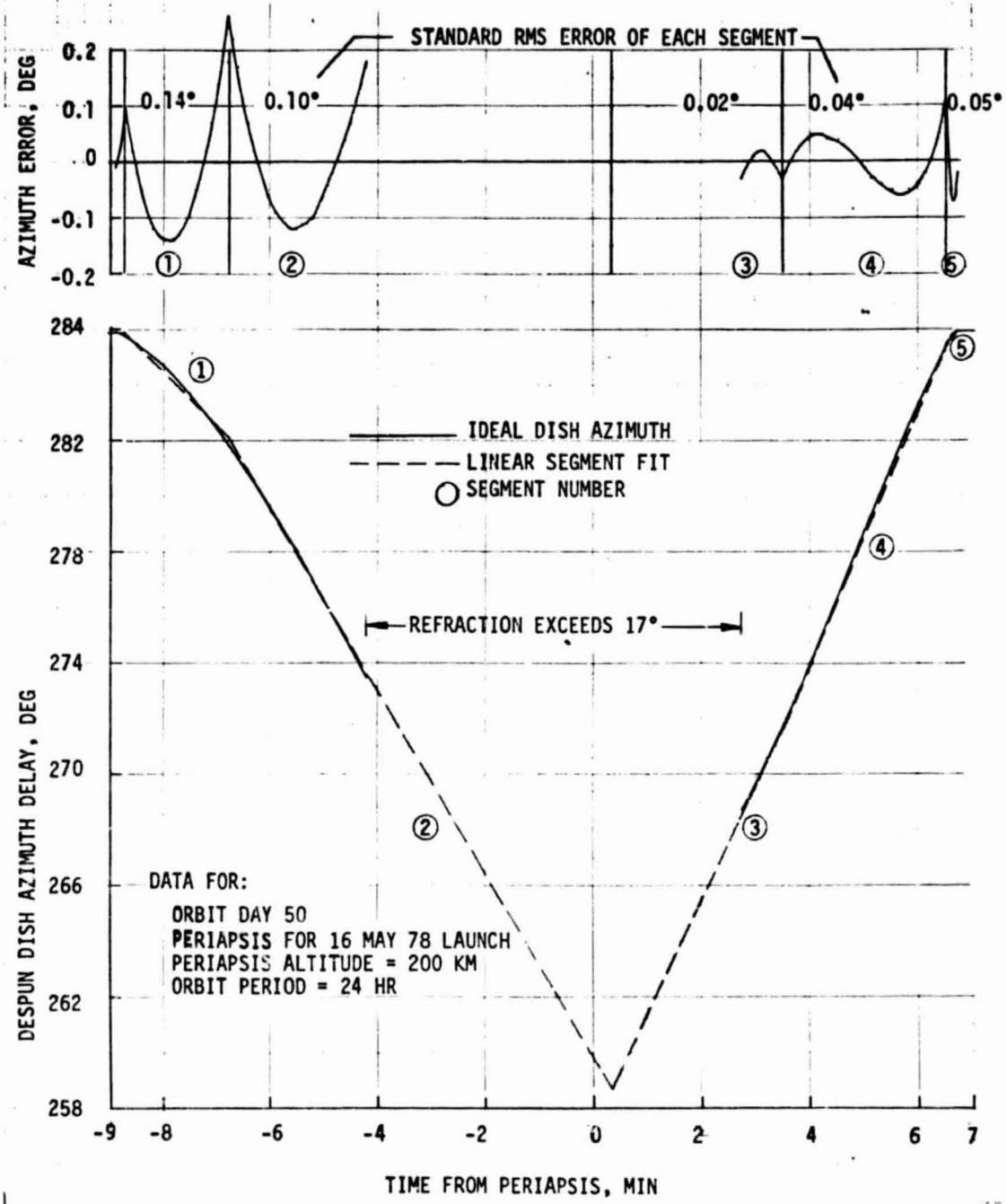
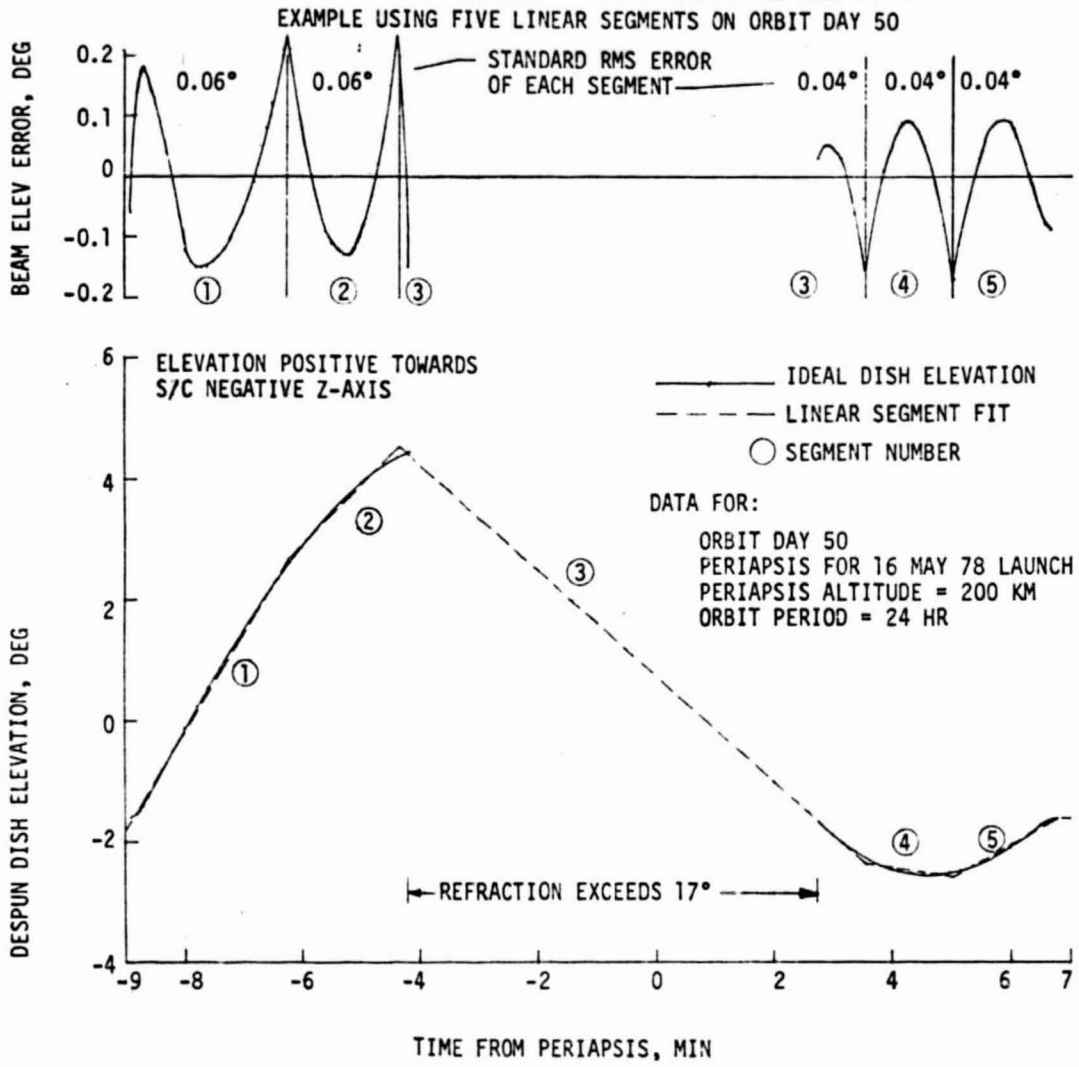


Figure 4.3.4.3.2.5-2.
DESPUN DISH ELEVATION CONTROL FOR OCCULTATION POINTING



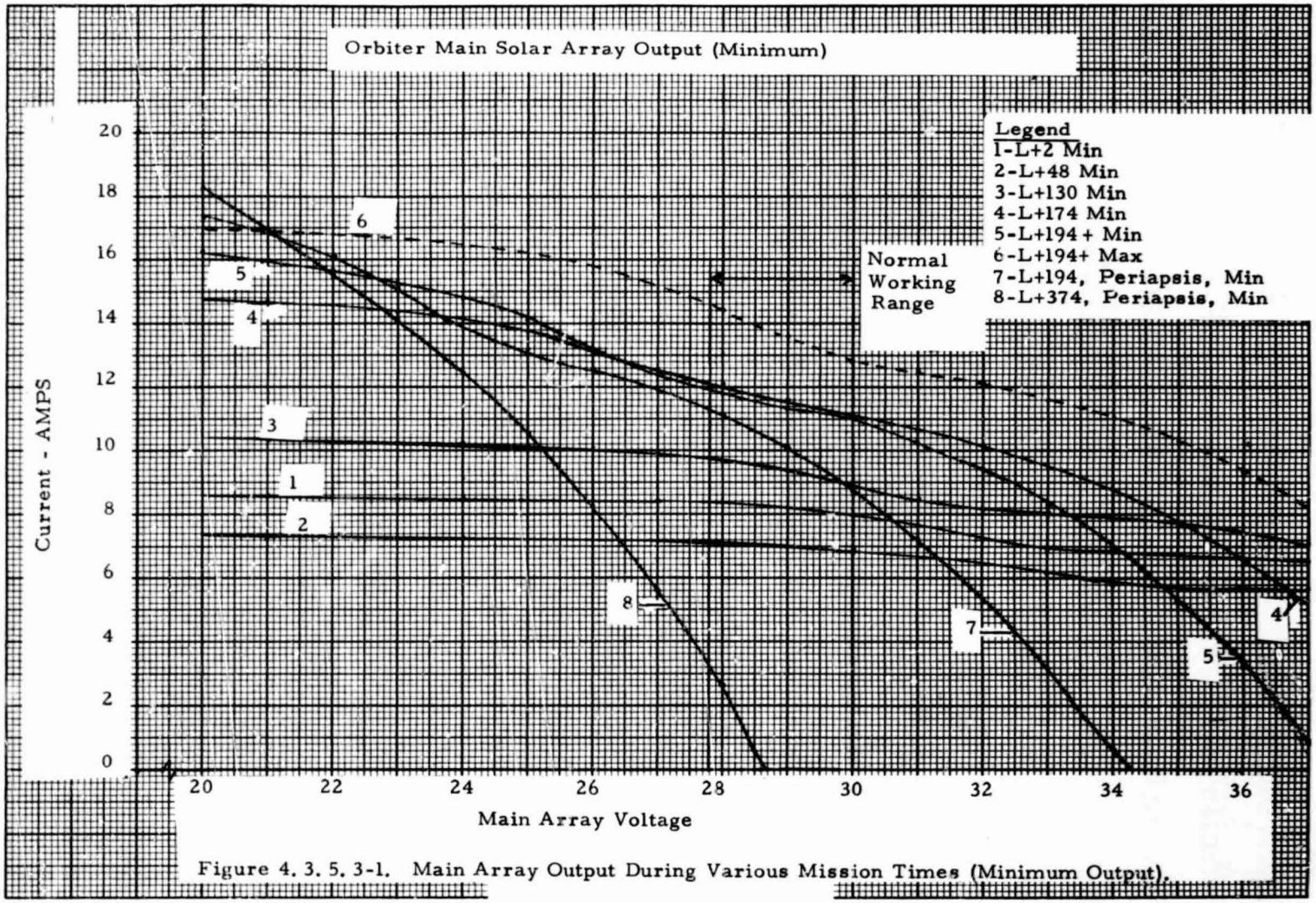


Figure 4.3.5.3-1. Main Array Output During Various Mission Times (Minimum Output).

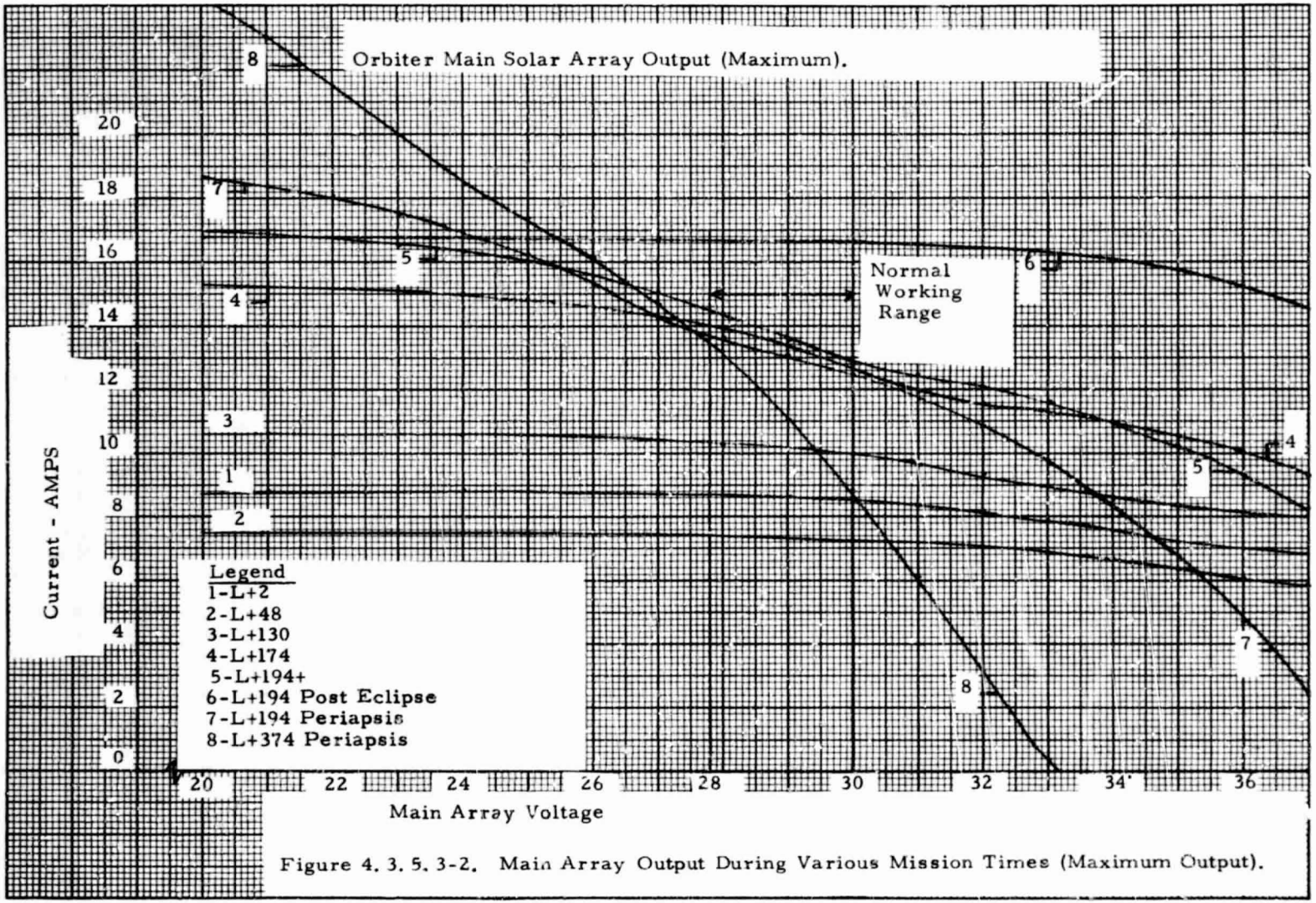


Figure 4.3.5.3-2. Main Array Output During Various Mission Times (Maximum Output).

APPENDIX A

TELEMETRY REFERENCE DATA

(BY SPACECRAFT SUBSYSTEM GROUPING)

APPENDIX A: TELEMETRY REFERENCE DATA (By S/C Subsystem Grouping)

SUBSYSTEM: (A,1) Spacecraft Level

TM MNE-MONIC	TELEMETRY TITLE	TM TYPE & (UNITS)	TELEMETERED RANGE	POWER SOURCE	SOURCE DERIVATION AND REMARKS
	Telemetry Format Selection	Overhead Word #3 - Digital (4 bits) (None)	16 Discrete levels - From 0000 to 1111 - representing any one of 16 selected formats.	Sw. loads bus to ECU in TP that is ON to +5V line.	Location 2 of RAM (scratch pad memory).
	Bit Rate Selection	Overhead Word #3 - Digital (4 bits) (bps)	16 Discrete levels - From 0000 to 1111 - representing any one of 12 selected bit rates. 0000 ≡ 8 bps; 1100 or 1101 or 1111 ≡ 4096 bps.	Sw. loads bus to ECU in TP that is ON to +5V line.	Location 2 of RAM (scratch pad memory).
	Spacecraft ID	Overhead Word #4 - Digital (2 bits) (None)	00 ≡ Bus 10 ≡ Orbiter	Sw. loads bus to ECU in TP that is ON to +5V line.	Location 14 of RAM (scratch pad memory). Pin programming is sampled and stored in RAM.
DSCIDC	Minor Frame Count	Overhead Word #4 - Digital (6 bits) (Decimal)	000000 ≡ Frame #0 111111 ≡ Frame #63	Sw. loads bus to ECU in TP that is ON to +5V line.	Location 14 of RAM (scratch pad memory).
	Spacecraft ID	Overhead Word #8 - Digital (1 bit) (None)	0 ≡ Bus; 1 ≡ Orbiter.	Sw. loads bus to ECU in TP that is ON to +5V line.	Location 0 in RAM (scratch pad memory). Pin programming is sampled and stored in the RAM.
	DSU Rollover Status	Overhead Word #8 - Digital (1 bit) (None)	Always Zero (0).	Sw. loads bus to ECU in either DSU to +5V line.	Hi-to-Lo transition of MSB of address counter in either DSU produces a "1" signal, for a few microseconds, that is sampled & placed in RAM location 0.

SUBSYSTEM: (A.1) Spacecraft Level

TM MNE-MONIC	TELEMETRY TITLE	TM TYPE & (UNITS)	TELEMETERED RANGE	POWER SOURCE	SOURCE DERIVATION AND REMARKS
DDSU2S	DSU 2 Status	Overhead Word #8 - Digital (1 bit) (None)	0 ≡ DSU 2 OFF 1 ≡ DSU 2 ON	Sw. loads bus to ECU in DSU 2 to +15V & +5V lines.	Output of differential amp into which +15V and +5V lines of DSU 2 are fed. This output is sampled and placed in RAM location 0.
DDSU1S	DSU 1 Status	Overhead Word #8 - Digital (1 bit) (None)	0 ≡ DSU 1 OFF 1 ≡ DSU 1 ON	Sw. loads bus to ECU in DSU 1 to +15V & +5V lines.	Output of differential amp into which +15V and +5V lines of DSU 1 are fed. This output is sampled and placed in RAM location 0.
DDSUCS	DSU Control	Overhead Word #8 - Digital (3 bits) (None)	8 Discrete levels - from 000 to 111 - representing any one of four (4) DSU states for each of the two DSUs. 000 or 001 or 111 = STBY = Standby 010 = READ DSU 1 011 = READ DSU 2 100 = STORE DSU 1 101 = STORE DSU 2 110 = STANDBY - RESET	Sw. loads bus to ECU in TP that is ON to +5V line.	Location 0 of RAM (scratch pad memory).

APPENDIX A: TELEMETRY REFERENCE DATA (By S/C Subsystem Grouping)

SUBSYSTEM: (A.2) Structure/Harness

TM MNE-MONIC	TELEMETRY TITLE	TM TYPE & (UNITS)	TELEMETERED RANGE	POWER SOURCE	SOURCE DERIVATION AND REMARKS
SH1Ø1T	Shelf Temperature 1	AC (^{o}F)	64 Counts = -76^{o}F 263 Counts = +266^{o}F	Sw. loads bus to DIM 1 & precision 1.0 milliamp output.	Thermistor, 908631-32.
SH1Ø2T	Shelf Temperature 2	AC (^{o}F)	64 Counts = -67^{o}F 263 Counts = +275^{o}F	Sw. loads bus to DIM 1 & precision 1.0 milliamp output.	Thermistor, 908631-32.
SH1Ø3T	Shelf Temperature 3	AC (^{o}F)	64 Counts = -90^{o}F 263 Counts = +251^{o}F	Sw. loads bus to DIM 1 & precision 1.0 milliamp output.	Thermistor, 908631-32.
SH1Ø4T	Shelf Temperature 4	AC (^{o}F)	64 Counts = -87^{o}F 263 Counts = +256^{o}F	Sw. loads bus to DIM 3 & precision 1.0 milliamp output.	Thermistor, 908631-32.
SH1Ø5T	Shelf Temperature 5	AC (^{o}F)	64 Counts = -76^{o}F 263 Counts = +266^{o}F	Sw. loads bus to DIM 3 & precision 1.0 milliamp output.	Thermistor, 908631-32.
SH1Ø6T	Shelf Temperature 6	AC (^{o}F)	64 Counts = -80^{o}F 263 Counts = +261^{o}F	Sw. loads bus to DIM 3 & precision 1.0 milliamp output.	Thermistor, 908631-32.
SH1Ø7T	Shelf Temperature 7	AC (^{o}F)	64 Counts = -71^{o}F 263 Counts = +271^{o}F	Sw. loads bus to DIM 5 & precision 1.0 milliamp output.	Thermistor, 908631-32.
SH1Ø8T	Shelf Temperature 8	AC (^{o}F)	64 Counts = -75^{o}F 263 Counts = +267^{o}F	Sw. loads bus to DIM 5 & precision 1.0 milliamp output.	Thermistor, 908631-32.
SH1Ø9T	Shelf Temperature 9	AC (^{o}F)	64 Counts = -75^{o}F 263 Counts = +267^{o}F	Sw. loads bus to DIM 5 & precision 1.0 milliamp output.	Thermistor, 908631-32.

SUBSYSTEM: (A.2) Structure/Harness

TM MNE-MONIC	TELEMETRY TITLE	TM TYPE & (UNITS)	TELEMETERED RANGE	POWER SOURCE	SOURCE DERIVATION AND REMARKS
SHL10T	Shelf Temperature 10	AC (^{O}F)	64 Counts = -67^{O}F 263 Counts = +275^{O}F	Sw. loads bus to DIM 7 & precision 1.0 milliamp output.	Thermistor, 908631-32.
SHL11T	Shelf Temperature 11	AC (^{O}F)	64 Counts = -103^{O}F 263 Counts = +238^{O}F	Sw. loads bus to DIM 7 & precision 1.0 milliamp output.	Thermistor, 908631-32.
SHL12T	Shelf Temperature	AC (^{O}F)	64 Counts = -73^{O}F 263 Counts = +269^{O}F	Sw. loads bus to DIM 7 & precision 1.0 milliamp output.	Thermistor, 908631-32.
SORPAT	Retarding Potential Analyzer (ORPA) Temp.	AC (^{O}F)		Sw. loads Bus to DIM 5 & precision 1.0 milliamp output.	Thermistor, 908631-32.
OOIMST	Ion Mass Spectrometer (OIMS) Temperature	AC (^{O}F)		Sw. loads bus to DIM 6 & precision 1.0 milliamp output.	Thermistor, 908631-32.
SOETPT	Electron Temperature Probe (OETP) Temp.	AC (^{O}F)	64 Counts = -78^{O}F 263 Counts = +264^{O}F	Sw. loads bus to DIM 1 & precision 1.0 milliamp output.	Thermistor, 908631-32.
SONMST	Neutral Mass Spectrometer (ONMS) Temp.	AC (^{O}F)	64 Counts = -96^{O}F 263 Counts = +246^{O}F	Sw. loads bus to DIM 1 & precision 1.0 milliamp output.	Thermistor, 908631-32.
SOPAIT	Plasma Analyzer (OPA) Temp.	AC (^{O}F)	64 Counts = -81^{O}F 263 Counts = +261^{O}F	Sw. loads bus to DIM 3 & precision 1.0 milliamp output.	Thermistor, 908631-32.
SORADT	Radar Mapper (ORAD) Temp.	AC (^{O}F)	64 Counts = -78^{O}F 263 Counts = +264^{O}F	Sw. loads bus to DIM 1 & precision 1.0 milliamp output.	Thermistor, 908631-32.

APPENDIX A: TELEMETRY REFERENCE DATA (By S/C Subsystem Grouping)

SUBSYSTEM: (A.3) Control

TM MNE-MONIC	TELEMETRY TITLE	TM TYPE & (UNITS)	TELEMETERED RANGE	POWER SOURCE	SOURCE DERIVATION AND REMARKS
AADP1S	Attitude Data Processor 1 ON/OFF	BL (None)	0 ≡ OFF; 1 ≡ ON.	Sw. loads bus to (ADP 1 power conv. to +5V line).	Output of 5.1 K ohm resistor connected in series to +5V line in ADP 1. (Isolation Resistor)
AADP2S	Attitude Data Processor 2 ON/OFF	BL (None)	0 ≡ OFF; 1 ≡ ON.	Sw. loads bus to (ADP 2 power conv. to +5V line).	Output of 5.1 K ohm resistor connected in series to +5V line in ADP 2.
AJCE1S	Jet Control Electronics 1 Buffer Output Status	BL (None)	0 ≡ Disabled; 1 ≡ Enabled.	Sw. loads bus to (ADP 1 conv. to +15V line)	Resistor divider output from Q2A in microcircuit assembly DP2C in ADP 1.
AJCE2S	Jet Control Electronics 2 Buffer Output Status	BL (None)	0 ≡ Disabled; 1 ≡ Enabled.	Sw. loads bus to (ADP 2 conv. to +15V line).	Resistor divider output from Q2A in microcircuit assembly DP2C in ADP 2.
APOS1S	HGA Positioner Electronics 1 ON/OFF	BL (None)	0 ≡ OFF; 1 ≡ ON.	Sw. loads bus to (Fused 28V bus to HGA 1 power latch to switched bus).	28V Switched bus via buffer (In HGA Positioner Electronics 1).
APOS2S	HGA Positioner Electronics 2 ON/OFF	BL (None)	0 ≡ OFF; 1 ≡ ON.	Sw. loads bus to (Fused 28V bus to HGA 2 power latch to switched bus).	28V Switched bus via buffer (In HGA Positioner Electronics 2).
AMTR1S	Despin Motor Drive 1 ON/OFF	BL (None)	0 ≡ OFF; 1 ≡ ON.	Sw. loads bus to (Fused 28V bus in DCE 1) to Motor Drive 1 Latch).	Motor Drive 1 Latch.
AMTR2S	Despin Motor Drive 2 ON/OFF	BL (None)	0 ≡ OFF; 1 ≡ ON.	Sw. loads bus to (Fused 28V in DCE 2) to Motor Drive 2 Latch.	Motor Drive 2 Latch.

SUBSYSTEM: (A.3) Control

TM MNE-MONIC	TELEMETRY TITLE	TM TYPE & (UNITS)	TELEMETERED RANGE	POWER SOURCE	SOURCE DERIVATION AND REMARKS
ADCE1T	Despin Control Electronics 1 Temperature	AC (^{O}F)	81.0 Counts = +20^{O}F 126.0 Counts = +120^{O}F	Sw. loads bus to DIM 6 & precision 1.0 ma output.	Thermistor, 908631-31.
ADCE2T	Despin Control Electronics 2 Temperature	AC (^{O}F)	81.5 Counts = +17^{O}F 145.5 Counts = +150^{O}F	Sw. loads bus to DIM 7 & precision 1.0 ma output.	Thermistor, 908631-31.
ADCE1V	Despin Control Electronics 1 Power Supply Voltage	AN (Volts)	0 Counts = 0 VDC 255 Counts = 5.1 VDC	Sw. loads bus to (Fuses, filter and DCE 1 power conv. to +5V line).	+5V line in DCE 1 to Buffer.
ADCE2V	Despin Control Electronics 2 Power Supply Voltage	AN (Volts)	0 Counts = 0 VDC 255 Counts = 5.1 VDC	Sw. loads bus to (Fuses, filter and DCE 2 power conv. to +5V line).	+5V line in DCE 2 to Buffer.
AMOTRI	Despin Motor Drive Current	AN (Amps)	0 Counts = 0 Amps 145 Counts = 1.09 Amps	Sw. loads bus to DCE that is ON to its convert. to +12V and -12V lines.	Sine Motor Drive and Cos Motor Drive to Diff. Amp. (In DCE that is ON). (Buffer DCSE).
ARATEQ	Rate Torque	AN (Ft-Lbs)	90 Counts = 7.0 rpm 210 Counts = 3.0 rpm } For S/C rate of 4.88 rpm. 77.5 Counts = 25 rpm 257.5 Counts = 7.0 rpm } For S/C spin rate of 15 rpm.	Sw. loads bus to DCE that is ON to +15V and -15V lines.	Relative rate error register to D/A converter to Buffer (In DCE that is ON). (Microcircuit Assembly DC1K).
APOSTQ	Position Torque	AN (RPM)	100 Counts = 0.36 ft-lb 145 Counts = 0.0 ft-lb } For S/C spin rate of 4.88 rpm 100 Counts = 0.32 ft-lb 140 Counts = 0.0 ft-lb } For S/C spin rate of 15 rpm 100 Counts = 0.22 ft-lb 127.5 Counts = 0.0 ft-lb } For S/C spin rate of 30 rpm.	Sw. loads bus to DCE that is ON to its converte. to +15V and -15V lines.	Position Register to position D/A converter to limiter to Buffer. (In DCE that is ON). (Microcircuit Assembly DC1J).

APPENDIX A: TELEMETRY REFERENCE DATA (By S/C Subsystem Grouping)

SUBSYSTEM: (A.3) Control

TM MNE-MONIC	TELEMETRY TITLE	TM TYPE & (UNITS)	TELEMETERED RANGE	POWER SOURCE	SOURCE DERIVATION AND REMARKS
AELEVD	HGA Elevation Position	AN (Angular Degrees)	196.5 Counts = -15.91° { X-Band Beam 28.5 Counts = +15.33° { Position. 196.5 Counts = -15.77° { S-Band Beam 28.5 Counts = +15.09° { Position.	Sw. loads bus to DCE that is ON, to it's converter to +15V, -15V lines and to position pot. Ref (= +4.8V Reg.).	Wiper of 28K ohm pot. that is driven mechanically via screw actuator. (In HGA Positioner mechanism).
APOSMT	HGA Position Motor Temp.	AC (°F)	38 Counts = -298°F 215 Counts = +212°F	Sw. loads bus to DIM 5 and precision 1.0 ma output.	Thermistor 908683-5.
A*1BRM	PSI* Brightness	AN (Volts)	0 Counts = 0 Volts { Linear 255 Counts = +5.10 Volts { The equivalent silicon magnitude is related non-linearly to TM volts, via parameters of sensor temperature, bandpass state, S/C spin rate, and star elevation boresight axis. Exact relationship is described in Ref: Para. 1.5.21.	Sw. loads bus to PSI* power converter to +15V, -15V and +9V lines.	PSI* Detector to peak detector.
A*1THS	PSI* Threshold Setting	AN (Star Magnitude)	0 Counts = 0 Volts { Linear 255 Counts = 5.1 Volts { (Ref: 1.5.21) Threshold Setting: TM Millivolts: 8 258 7 330 6 403 5 495 4 813 3 1312 2 2101 1 3334		PSI* Bandpass and Threshold Register.
A*2BRM	PSI2* Brightness	AN (Volts)	Same as for A*1BRM.	Sw. loads bus to PSI2* Power conv. to +15V, -15V and +9V lines.	PSI2* Detector to variable Threshhold Detector.

SUBSYSTEM: (A.3) Control

TM MNE-MONIC	TELEMETRY TITLE	TM TYPE & (UNITS)	TELEMETERED RANGE	POWER SOURCE	SOURCE DERIVATION AND REMARKS
A*2THS	PSI2* Threshold Setting	AN (Star Magnitude)	0 Counts = 0 Volts 255 Counts = 5.1 Volts (Linear) (Ref: 1.5.21) Threshold Setting: TM Millivolts: 8 296 7 379 6 463 5 567 4 930 3 1500 2 2397 1 3807	Sw. loads bus to PSI2* power conv. to +15V, -15V and +9V lines.	PSI2* Bandpass and Threshold Register.
A*1ONS	PSI* ON/OFF	BL (None)	0 ≡ OFF; 1 ≡ ON.	Sw. loads Bus to PSI* Power Conv.	
A*2ONS	PSI2* ON/OFF	BL (None)	0 ≡ OFF; 1 ≡ ON.	Sw. loads bus to PSI2* Power Conv.	
ASTART	Star Sensor Temperature	AC (^{O}F)		Sw. loads bus to DIM 7 and precision 1.0 milliamp output.	Thermistor, 908631-32.
ABAPIT	BAPTA Inner Race Bearing Temperature	AC (^{O}F)	42 Counts ≡ +120^{O}F 234 Counts ≡ +39.2^{O}F	Sw. loads bus to DIM 5 and precision 1.0 milliamp output.	Thermistor, 908632-2.
ABAPOT	BAPTA Outer Race Bearing Temperature	AC (^{O}F)	254 Counts ≡ 0^{O}F 20 Counts ≡ +160^{O}F	Sw. loads bus to DIM 3 and precision 1.0 milliamp output.	Thermistor, 908632-2.
A*1BPS	PSI* Bandpass State Status	BL (None)	0 ≡ HI (19 Hz); 1 ≡ LO (2 Hz).	Sw. loads bus to PSI* Power conv. to +15V and -15V lines.	PSI* Bandpass and Threshold Register.

APPENDIX A: TELEMETRY REFERENCE DATA (By S/C Subsystem Grouping)

SUBSYSTEM: (A.3) Control

TM MNE-MONIC	TELEMETRY TITLE	TM TYPE & (UNITS)	TELEMETERED RANGE	POWER SOURCE	SOURCE DERIVATION AND REMARKS
A*2BPS	PSI2* Bandpass State Status	BL (None)	0 ≡ HI (19 Hz); 1 ≡ LO (2 Hz).	Sw. loads bus to PSI2* power conv. to +15V and -15V lines.	PSI2* Bandpass and Threshold Register.
SBOOMS	Magnetometer Boom Deployed Status	BL (None)	0 ≡ Deployed; 1 ≡ Stowed.	Sw. loads bus to DIM 2 and precision 1.0 milliamp output.	Microswitch, located at hinge support of mag. boom, and actuated by a locking detent pin when boom assembly rotates to fully deployed position. Switch is shorted to signal ground for "deployed" state; open circuit for "stowed" state.
	ATTITUDE MEASUREMENT	SD (8 LSBs of 16 bit words). (Seconds)	0 TM Counts ≡ 0 Seconds; 65535 TM Counts ≡ 16.38 Seconds. (Resolution ≡ 0.25 msec).	Sw. loads bus to ADP that is ON to its converter and to +5V line.	Output of 16 bit counter in microcircuit assembly DP1M in ADP that is ON. Count start and stop controlled in turn by selection of azimuth time measurements (measurements A and B).
ATTM1Z	Attitude Measurement				
ATTM2Z	Attitude Measurement	SD (8 MSBs of 16 bit words). (Seconds)			
AM1ADS	Measurement A Address	SD (3 bits) (None)	7 Discrete levels - from 000 to 101 - representing any one of 7 azimuthal time measurements from SRR.	Sw. loads bus to ADP that is ON to its converter and to +5V line.	Storage registers U15 and U16 (3 bits total) in microcircuit assembly DP1D in ADP that is ON.
AM2ADS	Measurement B	SD	7 Discrete levels - from 000 to 101 - representing any one of 7 azimuthal time measurements from SRR.	Sw. loads bus to ADP that is ON to its converter and to +5V line.	Storage register U16 (3 bits total) in microcircuit assembly DP1D in ADP that is ON.

SUBSYSTEM: (A.3) Control

TM MNE-MONIC	TELEMETRY TITLE	TM TYPE & (UNITS)	TELEMETERED RANGE	POWER SOURCE	SOURCE DERIVATION AND REMARKS
ATTMSS	Data Measure-ment	SD (1 Bit) (None)	0 ≡ Attitude Measurement applies for Measurement B; 1 ≡ Attitude Measurement applies for Measurement A.	Sw. loads bus to ADP that is ON to its converter and to +5V line.	Storage register U14 in micro-circuit assembly DP1H in ADP that is ON.
	ADP STATUS				
AJMAGC	JCE Countdown	SD (12 Bits) (Decimal pulse counts, if AMAGCS = 0;	0 to 4095 Pulse Counts.	Sw. loads bus to ADP that is ON to its converter and to +5V line.	Jet sequence converter in micro-circuit assembly DP2B in ADP that is ON.
		Seconds if AMAGCS = 0.)	0 Counts ≡ 0 Seconds; 4095 Counts ≡ 2096.64 seconds.		
APULLS	Pulse Width Select.	SD (1 Bit) (Milli-seconds of Pulse Width).	0 ≡ 128 milliseconds pulse width selected; 1 ≡ 512 milliseconds pulse width selected.	Sw. loads bus to ADP that is ON to its converter and to +5V line.	Storage register U23 in micro-circuit assembly DP1E in ADP that is ON.
AMAGCS	Pulse/Time Count Select	SD (1 Bit) (None)	0 ≡ Time Count selected for AJMAGC; 1 ≡ Pulse Count selected for AJMAGC.	Sw. loads bus to ADP that is ON to its converter and to +5V line.	Storage register U23 in micro-circuit assembly DP1E in ADP that is ON.
AJETMS* (Word 2, Bit 6)	Continuous/ Pulse Fire Select	SD (1 Bit) (None)	0 ≡ Pulse Fire selected; 1 ≡ Continuous Fires selected.	Sw. loads bus to ADP that is ON to its converter and to +5V line.	Storage register U23 in micro-circuit assembly DP1E in ADP that is ON.

APPENDIX A: TELEMETRY REFERENCE DATA (By S/C Subsystem Grouping)

SUBSYSTEM: (A.3) Control

TM MNE-MONIC	TELEMETRY TITLE	TM TYPE & (UNITS)	TELEMETERED RANGE	POWER SOURCE	SOURCE DERIVATION AND REMARKS
	ADP STATUS (Continued)				
ASDETS	Spin Rate Detector	SD (1 Bit) (None)	0 ≡ Inhibited; 1 ≡ Enabled.	Sw. loads bus to ADP that is ON to its converter and to +5V line.	Storage register U23 in microcircuit assembly DP1E in ADP that is ON.
AJETMS (Word 3, Bit 0)	Normal/Alternate Fire Mode	SD (1 Bit) (None)	0 ≡ Alternate fire selected; 1 ≡ Normal fire selected.	Sw. loads bus to ADP that is ON to its converter and to +5V line.	Storage register U22 in microcircuit assembly DP1E in ADP that is ON.
AJET1S	Jet Select Bits	SD (7 Bits) (None)	Bit 1: 0 ≡ Radial Jet 1 Disabled; 1 ≡ Radial Jet 1 Enabled.	Sw. loads bus to ADP that is ON to its converter and to +5V line.	Storage register U21 in microcircuit assembly DP1E in ADP that is ON.
AJET2S			Bit 2: 0 ≡ Radial Jet 2 Disabled; 1 ≡ Radial Jet 2 Enabled.	Sw. loads bus to ADP that is ON to its converter and to +5V line.	Storage register U21 in microcircuit assembly DP1E in ADP that is ON.
AJET3S			Bit 3: 0 ≡ Radial Jet 3 Disabled; 1 ≡ Radial Jet 3 Enabled.	Sw. loads bus to ADP that is ON to its converter and to +5V line.	Storage register U21 in microcircuit assembly DP1E in ADP that is ON.
AJET4S			Bit 4: 0 ≡ Radial Jet 4 Disabled; 1 ≡ Radial Jet 4 Enabled.	Sw. loads bus to ADP that is ON to its converter and to +5V line.	Storage register U21 in microcircuit assembly DP1E in ADP that is ON.
AJET5S			Bit 5: 0 ≡ Axial Jet 5 Disabled; 1 ≡ Axial Jet 5 Enabled.	Sw. loads bus to ADP that is ON to its converter and to +5V line.	Storage register U22 in microcircuit assembly DP1E in ADP that is ON.

SUBSYSTEM: (A.3) Control

TM MNE-MONIC	TELEMETRY TITLE	TM TYPE & (UNITS)	TELEMETERED RANGE	POWER SOURCE	SOURCE DERIVATION AND REMARKS
	ADP STATUS (Continued)				
AJET6S	Jet Select Bits (Continued)	SD (7 Bits) (None)	Bit 6: 0 ≡ Axial Jet 6 Disabled; 1 ≡ Axial Jet 6 Enabled.	Sw. loads bus to ADP that is ON to its converter and to +5V line.	Storage register U22 in microcircuit assembly DP1E in ADP that is ON.
AJET7S			Bit 7: 0 ≡ Axial Jet 7 Disabled; 1 ≡ Axial Jet 7 Enabled.	Sw. loads bus to ADP that is ON to its converter and to +5V line.	Storage register U22 in microcircuit assembly DP1E in ADP that is ON.
AJCEFS	JCE Fire Enable Status	SD (1 Bit) (None)	0 ≡ Disabled; 1 ≡ Enabled.	Sw. loads bus to ADP that is ON to its converter & to +5V line.	Main gate U13D in microcircuit assembly D2PB in ADP that is ON.
A*GASS	Star Gate A; Channel 1/2 Select	SD (1 Bit) (None)	0 ≡ Channel 2 (ψ*2) Selected; 1 ≡ Channel 1 (ψ *) Selected.	Sw. loads bus to ADP that is ON to its converter & to +5V line.	Storage register U21 in microcircuit assembly DP1F in ADP that is ON.
A*GBSS	Star Gate B; Channel 1/2 Select	SD (1 Bit) (None)	0 ≡ Channel 2 (ψ*2) Selected. 1 ≡ Channel 1 (ψ*) Selected.	Sw. loads bus to ADP that is ON to its converter & to +5V line.	Storage register U22 in microcircuit assembly DP1F in ADP that is ON.
ASUNSS	Sun Sensor Select	SD (2 Bits) (None)	3 Discrete levels - from 00 to 10 - representing any one of three sun sensor ranges selected.	Sw. loads bus to ADP that is ON to its converter & to +5V line.	Storage register U22 (2 bits total) in microcircuit assembly DP1F in ADP that is ON.
ASRRMS	Sun/Star Select; Simulated SRR Select	SD (2 Bits) (None)	Bit 5: 0 ≡ Star Selected; 1 ≡ Sun Selected. Bit 6: 0 ≡ Simulated SRR Selected; 1 ≡ SRR Selected.	Sw. loads bus to ADP that is ON to its converter & to +5V line.	Storage register U22 & U23 (2 bits total) in microcircuit assembly DP1F in ADP that is ON.

APPENDIX A: TELEMETRY REFERENCE DATA (By S/C Subsystem Grouping)

SUBSYSTEM: (A.3) Control

TM MNE-MONIC	TELEMETRY TITLE	TM TYPE & (UNITS)	TELEMETERED RANGE	POWER SOURCE	SOURCE DERIVATION AND REMARKS
	ADP STATUS (Continued)				
A*ACQS	Star Acquisition/ Normal	SD (1 Bit) (None)	0 ≡ Normal Selected; 1 ≡ Star Acquisition Selected.	Sw. loads bus to ADP that is ON to its converter & to +5V line.	Storage register U23 in microcircuit assembly DP1F in ADP that is ON.
ASPINS	PLL Spin Range Select	SD (2 Bits) (RPM)	4 Discrete levels - from 00 to 11 - representing any one of four spin ranges selected.	Sw. loads bus to ADP that is ON to its converter & to +5V line.	Storage register U21 (2 bits total) in microcircuit assembly DP1F in ADP that is ON.
ALOLES	PLL Loss of Lock Enable/ Inhibit	SD (1 Bit) (None)	0 ≡ Disabled; 1 ≡ Enabled.	Sw. loads bus to ADP that is ON to its converter & to +5V line.	Storage register U21 in microcircuit assembly DP1F in ADP that is ON.
ASUNGS	Enable/Disable Sun Gate	SD (1 Bit) (None)	0 ≡ Disabled; 1 ≡ Enabled.	Sw. loads bus to ADP that is ON to its converter & to +5V line.	Storage register U23 in microcircuit assembly DP1F in ADP that is ON.
ADVANS	SRR Advance Status	SD (1 Bit) (None)	0 ≡ Not Advanced; 1 ≡ Advanced.	Sw. loads bus to ADP that is ON to its converter to +5V line.	Storage register U23 in microcircuit assembly DP1F in ADP that is ON.
ARIPAD	Roll Index Delay Magnitude	SD (10 Bits) (Angular degrees in Azimuth)	0 TM Counts ≡ 0 Degrees; 1024 TM Counts ≡ 359.65 Degrees. Resolution ≡ 0.351 Degrees.	Sw. loads bus to ADP that is ON to its converter to +5V line.	Storage registers U24, U25 & U26 (10 bits total) in microcircuit assembly DP1F in ADP that is ON.
ALOCKS	PLL Loss of Lock	SD (1 Bit) (None)	0 ≡ Loss of Lock; 1 ≡ No Loss of Lock.	Sw. loads bus to ADP that is ON to its converter & to +5V line.	Storage register U21 in microcircuit assembly DP1F in ADP that is ON.
AMISSC	Missed SRR Count (0-8) Counts	SD (4 Bits) (Decimal Counts)	9 Discrete levels from 0000 to 1000 - representing number of successive out-of-PLL-Lock SRR pulses.	Sw. loads bus to ADP that is ON to its converter to +5V line.	Storage register U21 in microcircuit assembly DP4K in ADP that is ON. Counter is a 4-bit counter; 3 bits are used for the count & the 4th bit is an overflow bit.

SUBSYSTEM: (A.3) Control

TM MNE-MONIC	TELEMETRY TITLE	TM TYPE & (UNITS)	TELEMETERED RANGE	POWER SOURCE	SOURCE DERIVATION AND REMARKS
	ADP STATUS (Continued)				
AACSAD	ACS Angle Delay Magnitude	SD (10 Bits (Angular Degrees in Azimuth)	0 TM Counts ≡ 0.352 Degrees; 1024 TM Counts ≡ 360.00 Degrees. Resolution ≡ 0.352 Degrees.	Sw. loads bus to ADP that is ON to its converter to +5V line.	Storage registers U24, U25 & U26 (10 bits total) in microcircuit assembly DP1E in ADP that is ON.
ASIMSZ	PLL Spin Period Magnitude	SD (16 Bits) (Seconds)	0 TM Counts ≡ 0 Seconds; 65535 TM Counts ≡ 16.384 Seconds. Resolution ≡ 0.25 milliseconds.	Sw. loads bus to ADP that is ON to its converter to +5V line.	Storage registers U17, U18, U19 & U20 (16 bits total) in microcircuit assembly DP1D in ADP that is ON.
	DCE STATUS				
AVPD1D	RAM Variable Phase Delay	SD (12 Bits) (Seconds)	0 TM Counts ≡ 0 Seconds; 4095 TM Counts ≡	Sw. loads bus to DCE that is ON to its converter to +5V line.	Storage registers U4, U5, & U6 (12 bits total) in microcircuit assembly DC2D in DCE that is ON.
AVPD2D	NADIR Variable Phase Delay	SD (12 Bits) (Seconds)	0 TM Counts ≡ 0 Seconds 4095 TM Counts ≡	Sw. loads but to DCE that is ON to its converter to +5V line.	Storage registers U4, U5, & U6 (12 bits total) in microcircuit assembly DC2E in DCE that is ON.
ADSPNS	Despin Phase Reset Enable/ Disable	SD (1 Bit) (None)	0 ≡ Disabled; 1 ≡ Enabled.	Sw. loads but to DCE that is ON to its converter to +5V line.	JK Flip-flop U3B in microcircuit assembly DC3D in DCE that is ON.

APPENDIX A: TELEMETRY REFERENCE DATA (By S/C Subsystem Grouping)

SUBSYSTEM: (A.4) Propulsion

TM MNE-MONIC	TELEMETRY TITLE	TM TYPE & (UNITS)	TELEMETERED RANGE	POWER SOURCE	SOURCE DERIVATION AND REMARKS
VALV1S	Latch Valve 1 OPEN/CLOSED	BL (None)	0 ≡ Closed; 1 ≡ Open.	Sw. loads bus to DIM 2 & precision 1.0 milliamp output.	State of SPDT switch operated by Latch Valve 1 mechanism. Switch is grounded for "valve closed" state; open for "valve open" state.
VALV2S	Latch Valve 2 OPEN/CLOSED	BL (None)	0 ≡ Closed; 1 ≡ Open.	Sw. loads bus to DIM 2 & precision 1.0 milliamp output.	State of SPDT switch operated by Latch Valve 2 mechanism. Switch is grounded for "valve closed" state; open for "valve open" state.
VCASEI	OIM Case Temperature	AC (^{o}F)		Switch loads bus to DIM 5 & precision 1.0 milliamp output.	Thermistor, 908631-32.
VJET1T	Radial Jet 1 Temperature	AC (^{o}F)	38 Counts = -298^{o}F 216 Counts = +212^{o}F	Sw. loads bus to DIM 1 & precision 1.0 milliamp output.	Thermistor, 908683-4.
VJET2T	Radial Jet 2 Temperature	AC (^{o}F)	38 Counts = -298^{o}F 215 Counts = +212^{o}F	Sw. loads bus to DIM 1 & precision 1.0 milliamp output.	Thermistor, 908683-4.
VJET3T	Radial Jet 3 Temperature	AC (^{o}F)	38 Counts = -298^{o}F 216 Counts = +212^{o}F	Sw. loads bus to DIM 6 & precision 1.0 milliamp output.	Thermistor, 908683-4.
VJET4T	Radial Jet 4 Temperature	AC (^{o}F)	38 Counts = -298^{o}F 215 Counts = +212^{o}F	Sw. loads bus to DIM 6 & precision 1.0 milliamp output.	Thermistor, 908683-4.
VJET5T	Forward Axial Jet 5 Temperature	AC (^{o}F)	38 Counts = -298^{o}F 216 Counts = +212^{o}F	Sw. loads bus to DIM 6 & precision 1.0 milliamp output.	Thermistor, 908683-4.

SUBSYSTEM: (A.4) Propulsion

TM MNE-MONIC	TELEMETRY TITLE	TM TYPE & (UNITS)	TELEMETERED RANGE	POWER SOURCE	SOURCE DERIVATION AND REMARKS
VJET6T	Aft Axial Jet 6 Temperature	AC (^{O}F)	38 Counts = -298^{O}F 216 Counts = +212^{O}F	Sw. loads bus to DIM 1 & precision 1.0 milliamp output.	Thermistor, 908683-4.
VJET7T	Aft Axial Jet 7 Temperature	AC (^{O}F)	38 Counts = -298^{O}F 216 Counts = +212^{O}F	Sw. loads bus to DIM 7 & precision 1.0 milliamp output.	Thermistor, 908683-4.
VLIN1T	Propellant Line Temperature 1	AC (^{O}F)	103 Counts = +18^{O}F 230 Counts = +168^{O}F	Sw. loads bus to DIM 1 & precision 1.0 milliamp output.	Thermistor, 908631-32.
VLIN2T	Propellant Line Temperature 2	AC (^{O}F)	92 Counts = +18^{O}F 206 Counts = +168^{O}F	Sw. loads bus to DIM 1 & precision 1.0 milliamp output.	Thermistor, 908631-32.
VLIN3T	Propellant Line Temperature 3	AC (^{O}F)	92 Counts = +18^{O}F 205 Counts = +168^{O}F	Sw. loads bus to DIM 6 & precision 1.0 milliamp output.	Thermistor, 908631-32.
VLIN4T	Propellant Line Temperature 4	AC (^{O}F)	88 Counts = +18^{O}F 197 Counts = +168^{O}F	Sw. loads bus to DIM 6 & precision 1.0 milliamp output.	Thermistor, 908631-32.
VTANKP	Tank Pressure	AC (PSIA)	0 Counts = 0 PSIA 229.6 Counts = 363.5 PSIA	Sw. loads bus to DIM 6 & precision 1.0 milliamp output.	Potentiometer type transducer.
VTNK1T	Tank 1 Temperature	AC (^{O}F)	87 Counts = +16^{O}F 203 Counts = +171^{O}F	Sw. loads bus to DIM 6 & precision 1.0 milliamp output.	Thermistor, 908631-32.
VTNK2T	Tank 2 Temperature	AC (^{O}F)	87 Counts = +16^{O}F 200 Counts = +171^{O}F	Sw. loads bus to DIM 1 & precision 1.0 milliamp output.	Thermistor, 908631-32.

APPENDIX A: TELEMETRY REFERENCE DATA (By S/C Subsystem Grouping)

SUBSYSTEM: (A.4) Propulsion

TM MNE-MONIC	TELEMETRY TITLE	TM TYPE & (UNITS)	TELEMETERED RANGE	POWER SOURCE	SOURCE DERIVATION AND REMARKS
VTHROT	OIM Throat Temperature	AC (^{O}F)		Sw. loads bus to DIM 3 & precision 1.0 milliamp output.	Thermistor, 908631-32.
VSAFET	OIM Safe and Arm Temperature	AC (^{O}F)		Sw. loads bus to DIM 3 & precision 1.0 milliamp output.	Thermistor, 908631-32.
VJ12HS	R1 and R2 Line Heater Status	AN (Amps)	0 Counts = 0 Amps 50 Counts = 0.294 Amps	Essential bus downstream of fuses.	Resistive divider, between negative terminal of R1 and R2 line heaters, and power ground.
VJ34HS	R3 and R4 Line Heater Status	AN (Amps)	0 Counts = 0 Amps 50 Counts = 0.220 Amps	Essential bus, downstream of fuses.	Resistive divider, between negative terminals of R3 and R4 line heaters and power ground.
VLVPHS	Latch Valve Primary Heater Status	AN (Amps)	0 Counts = 0 Amps 50 Counts = 0.061 Amps	Essential bus, downstream of Primary/ Secondary Propulsion Tank heater select relay terminal (primary line) and power ground.	Resistive divider, between negative terminals of Latch Valve 1 heater, Latch Valve 2 heater, F&D #1 line heater, F&D #2 line heater - all tied together - and power ground.
VFDVHS	Fill and Drain Valve Primary Heater Status	AN (Amps)	0 Counts = 0 Amps 50 Counts = 0.125 Amps	Essential bus, downstream of Primary/ Secondary Propulsion Tank heater select relay terminal (primary line) and power ground.	Resistive divider, between negative terminals of F&D Valve #1 heater, F&D Valve #2 heater, F&D Valve #3 heater, F&D #3 Gas line - all tied together - and power ground.

SUBSYSTEM: (A.5) Data Handling

TM MNE-MONIC	TELEMETRY TITLE	TM TYPE & (UNITS)	TELEMETERED RANGE	POWER SOURCE	SOURCE DERIVATION AND REMARKS
DCLOK1	Spacecraft Clock (8 MSB)	SD (24 Bits) (Seconds)	0 TM Counts ≡ 0 Seconds. (2^{24} - 1) TM Counts ≡ 2,097,351 Seconds (= 24.27 Days). Resolution ≡ 0.125 Seconds.	Sw. loads bus to ECU in TP that is ON to +5V line.	Location 10, 11, & 12 of RAM (scratch pad memory). Origin of RAM update is minor frame rate signal counted down from 1.048 MHz clock in TP that is ON.
DCLOK2	Spacecraft Clock (8 Middle Bits)				
DCLOK3	Spacecraft Clock (8 LSB)				
DPCM1S	PCM Encoder 1 ON/OFF	BL (None)	0 ≡ OFF 1 ≡ ON	Sw. loads bus to Power Supply in PCME1 to +5V line.	Output of 1.0 Kohm resistor connected in series to +5V line in PCME1.
DPCM2S	PCM Encoder 2 ON/OFF	BL (None)	0 ≡ OFF 1 ≡ ON	Sw. loads bus to Power Supply in PCME2 to +5V line.	Output of 1.0 Kohm resistor connected in series to +5V line in PCME2.
DPCM1V	PCM Encoder 1 Calibrate	AC (Volts)	0 Counts ≡ 0.0 Volts 255 Counts ≡ +5.100 Volts	Sw. loads bus to Power Supply in PCME1 to -15V line.	Precision 1 milliampere current source into 4.87 Kohm ±0.2% resistor.
DPCM2V	PCM Encoder 2 Calibrate	AC (Volts)	0 Counts ≡ 0.0 Volts 255 Counts ≡ +5.100 Volts	Sw. loads bus to Power Supply in PCME2 to -15V line.	Precision 1 milliampere current source into 4.87 Kohm ±0.2% resistor.
DTLM1S	Telemetry Processor 1 ON/OFF	BL (None)	0 ≡ OFF 1 ≡ ON	Sw. loads bus to ECU to +5V line in TP 1.	Output of 8.2 Kohm resistor connected in series to +5V line in TP 1.
DTLM2S	Telemetry Processor 2 ON/OFF	BL (None)	0 ≡ OFF 1 ≡ ON	Sw. loads bus to ECU to +5V line in TP 2.	Output of 8.2 Kohn resistor connected in series to +5V line in TP 2.

APPENDIX A: TELEMETRY REFERENCE DATA (By S/C Subsystem Grouping)

SUBSYSTEM: (A.5) Data Handling

TM MNE-MONIC	TELEMETRY TITLE	TM TYPE & (UNITS)	TELEMETERED RANGE	POWER SOURCE	SOURCE DERIVATION AND REMARKS
MRO	Data Handling Memory Read-out	SD (65,536 x 8 Bits maximum for the selected DSU).		Sw. loads bus to Power Supply of commanded DSU to +15V, -15V, +5V lines (Power Supply is identical to that in TP).	Memory of DSU that was commanded to be read out.

SUBSYSTEM: (A. 6) Command

TM MNE-MONIC	TELEMETRY TITLE	TM TYPE & (UNITS)	TELEMETERED RANGE	POWER SOURCE	SOURCE DERIVATION AND REMARKS
CREJ1S	CP1 Command Reject	SD (1 Bit) (None)	0 ≡ Real time command is accepted; 1 ≡ Real time command is rejected.	Essential bus to CP1 ECU to +15V & +5V lines.	Parallel in-serial out 8 bit shift register (U6) in Telemetry Logic Module of CP1.
CCMD1C	Command Processor 1 Real Time Command Status	SD (7 Bits) (Decimal Count)	0 TM Counts ≡ 0 Commands 127 TM Counts ≡ 127 Commands. (Number of real-time commands verified & executed since last reset of command counter).		
CLOG1S	CP1 Stored Command Logic State	SD (4 Bits) (None)	16 Discrete levels - from 0000 to 1111 - representing any one of 16 operating states.	Essential bus to CP1 ECU to +15V & +5V lines.	Parallel in-serial out 8 bit shift register (U4) in Telemetry Logic Module of CP1.
CSCL1S	CP1 Stored Command Logic ON/OFF State	SD (1 Bit) (None)	0 ≡ OFF; 1 ≡ ON.		
CDMD1S	CP1 Demod. Squelch Status	SD (1 Bit) (None)	0 ≡ Squelched; 1 ≡ Not Squelched. (i.e., signal is present).		
RREV1C	CP1 MSB Bits of Receiver Reverse Timer	SD (2 Bits) (Hours)	01 = 9.1 Hours 10 = 18.2 Hours 11 = 27.3 Hours } Since Last Automatic Reversal.		
SSEP1S	CP1 Separation (Switch) Status	SD (1 Bit) (None)	0 ≡ Not Separated; 1 ≡ Separated.	Essential bus to 20 V zener diode in CP1.	Wiper arm of one pole of DPDT switch (spacecraft/attach fitting separation switch #1).
CMEM1C	Command Memory 1 Address Pointer	SD (7 Bits) (Decimal)	0 TM Counts ≡ First Command Memory Cell Location; 127 TM Counts ≡ 128th Command Memory Cell Location.	Essential bus to CP1 ECU to +5V & -12V lines.	Parallel in-serial out 8 bit shift register in SCL 1 (includes SSEP1S).

SUBSYSTEM: (A.6) Command

TM MNE-MONIC	TELEMETRY TITLE	TM TYPE & (UNITS)	TELEMETERED RANGE	POWER SOURCE	SOURCE DERIVATION AND REMARKS
CMEM1	Command Memory 1 Readout	SD (128 x 24 Bits Maximum in 8-bit Segments)	Any Commands and Time Delays.	Essential bus to CP1 ECU to +5V and -12V lines.	Command Memory in CP1.
CREV1S	Command Processor 1 Receiver Reverse State	BL (None)	0 ≡ Reverse Commanded; 1 ≡ Normal Commanded.	Essential bus to CP1 ECU to +5V line.	Output of 1 Kohm resistor connected to gate in series that is pulled up by 7.5 Kohm resistor connected to +5V.
CPWR1S	Command Processor 1 Power Status	BL (None)	0 ≡ OFF; 1 ≡ ON.	Essential bus to CP1 ECU to +5V line.	Output of 10 Kohm resistor connected in series to +5V line in CP1.
CREJ2S	CP2 Command Reject	SD (1 Bit) (None)	0 ≡ Real Time Command is Accepted; 1 ≡ Real Time Command is Rejected.	Essential bus to CP2 ECU to +15V & +5V lines.	Parallel in-serial out 8 bit shift register (U6) in telemetry logic module of CP2.
CCMD2C	Command Processor 2 Real Time Command Status	SD (7 Bits) (Decimal Count)	0 TM Counts ≡ 0 Commands; 127 TM Counts ≡ 127 Commands. (Number of real-time commands verified & executed since last reset of command counter).		
CLOG2S	CP2 Stored Command Logic State	SD (4 Bits) (None)	16 Discrete levels - from 0000 to 1111 - representing any one of 16 operating states.	Essential bus to CP2 ECU to +15V and +5V lines.	Parallel in-serial out 8 bit shift register (U4) in telemetry logic module of CP2.
CSCL2S	CP2 Stored Command Logic ON/OFF Status	SD (1 Bit) (None)	0 ≡ OFF; 1 ≡ ON.		
CDMD2S	CP2 Demod. Squelch Status	SD (1 Bit) (None)	0 ≡ Squelched; 1 ≡ Not Squelched. (i.e., signal is present).		

SUBSYSTEM: (A.6) Command

TM MNE-MONIC	TELEMETRY TITLE	TM TYPE & (UNITS)	TELEMETERED RANGE	POWER SOURCE	SOURCE DERIVATION AND REMARKS
RREV2C	CP2 MSB of Receiver Reverse Timer	SD (2 Bits) (Hours)	01 = 9.1 Hours 10 = 18.2 Hours 11 = 27.3 Hours } Since Last Automatic Reversal	Essential bus to CP2 ECU to +15V & +5V lines.	Parallel in-serial out 8 bit shift register (U4) in telemetry logic module of CP2.
SSEP2S	CP2 Separation Status	SD (1 Bit) (None)	0 ≡ Not Separated; 1 ≡ Separated.	Essential bus to 20V Zener diode in CP2.	Wiper arm of one pole of DPDT switch (spacecraft/attach fitting separation switch #2).
CMEM2C	Command Memory 2 Address Pointer	SD (7 Bits) (Decimal)	0 TM Counts ≡ First Command Memory Cell Location; 127 TM Counts ≡ 128th Command Memory Cell Location.	Essential bus to CP2 ECU to +5V & -12V lines.	Parallel in-serial out 8 bit shift register in SCL2 (includes SSEP2S).
CMEM2	Command Memory 2 Readout	SD (128 x 24 Bits Maximum in 8-bit Segments)	Any commands and time delays.	Essential bus to CP2 ECU to +5V & -12V lines.	Command Memory in CP 2.
CREV2S	Command Processor 2 Receiver Reverse State	BL (None)	0 ≡ Reverse Commanded; 1 ≡ Normal Commanded.	Essential bus to CP2 ECU to +5V line.	Output of 1 Kohm resistor connected to gate in series that is pulled up by 7.5 Kohm resistor that is connected to +5V.
CPWR2S	Command Processor 2 Power Status	BL (None)	0 ≡ OFF; 1 ≡ ON.	Essential bus to CP2 ECU to +5V line.	Output of 10 Kohm resistor connected in series to +5V line in CP2.

SUBSYSTEM: (A. 6) Command

TM MNE-MONIC	TELEMETRY TITLE	TM TYPE & (UNITS)	TELEMETERED RANGE	POWER SOURCE	SOURCE DERIVATION AND REMARKS
RREV2C	CP2 MSB of Receiver Reverse Timer	SD (2 Bits) (Hours)	01 = 9.1 Hours 10 = 18.2 Hours 11 = 27.3 Hours } Since Last Automatic Reversal	Essential bus to CP2 ECU to +15V & +5V lines.	Parallel in-serial out 8 bit shift register (U4) in telemetry logic module of CP2.
SSEP2S	CP2 Separation Status	SD (1 Bit) (None)	0 ≡ Not Separated; 1 ≡ Separated.	Essential bus to 20V Zener diode in CP2.	Wiper arm of one pole of DPDT switch (spacecraft/attach fitting separation switch #2).
CMEM2C	Command Memory 2 Address Pointer	SD (7 Bits) (Decimal)	0 TM Counts ≡ First Command Memory Cell Location; 127 TM Counts ≡ 128th Command Memory Cell Location.	Essential bus to CP2 ECU to +5V & -12V lines.	Parallel in-serial out 8 bit shift register in SCL2 (includes SSEP2S).
CMEM2	Command Memory 2 Readout	SD (128 x 24 Bits Maximum in 8-bit Segments)	Any commands and time delays.	Essential bus to CP2 ECU to +5V & -12V lines.	Command Memory in CP 2.
CREV2S	Command Processor 2 Receiver Reverse State	BL (None)	0 ≡ Reverse Commanded; 1 ≡ Normal Commanded.	Essential bus to CP2 ECU to +5V line.	Output of 1 Kohm resistor connected to gate in series that is pulled up by 7.5 Kohm resistor that is connected to +5V.
CPWR2S	Command Processor 2 Power Status	BL (None)	0 ≡ OFF; 1 ≡ ON.	Essential bus to CP2 ECU to +5V line.	Outpit of 10 Kohm resistor connected in series to +5V line in CP2.

APPENDIX A: TELEMETRY REFERENCE DATA (By S/C Subsystem Grouping)

SUBSYSTEM: (A.7) Communications

TM MNE-MONIC	TELEMETRY TITLE	TM TYPE & (UNITS)	TELEMETERED RANGE	POWER SOURCE	SOURCE DERIVATION AND REMARKS
RAGC1V	Receiver 1 Automatic Gain Control	AN (dBm)	95 Counts = -152.0 dBm; 198.8 Counts = -70.0 dBm.	Essential bus to (fuses to power converter to +9V & -6V lines in RCVR 1).	Resistive voltage divider at output of active AGC loop filter in RCVR 1.
RAGC2V	Receiver 2 Automatic Gain Control	AN (dBm)	65 Counts = -152.0 dBm; 192.45 Counts = -70.0 dBm.	Essential bus to (fuses to power converter to +9V & -6V lines in RCVR 2).	Resistive voltage divider at output of active AGC loop filter in RCVR 2.
RAMP1T	Power Amplifier 1 Temperature	AC (^{o}F)	39 Counts = -298^{o}F 216 Counts = +212^{o}F	Sw. loads bus to DIM3 & precision 1.0 milliamp output.	Thermistor, P/N 908683-4.
RAMP2T	Power Amplifier 2 Temperature	AC (^{o}F)	38 Counts = -298^{o}F 215 Counts = +212^{o}F	Sw. loads bus to DIM3 & precision 1.0 milliamp output.	Thermistor, P/N 908683-4.
RAMP3T	Power Amplifier 3 Temperature	AC (^{o}F)	38 Counts = -298^{o}F 215 Counts = +212^{o}F	Sw. loads bus to DIM5 & precision 1.0 milliamp output.	Thermistor, P/N 908683-4.
RAMP4T	Power Amplifier 4 Temperature	AC (^{o}F)	38 Counts = -298^{o}F 216 Counts = +212^{o}F	Sw. loads bus to DIM5 & precision 1.0 milliamp output.	Thermistor, P/N 908683-4.
RAMP1W	RF Power Output 1	AN (Watts)	0 Counts = 0.0 Watts; 187.5 Counts = 11.54 Watts.	Transmitter RF bus to (fuses to series reg. in power amp 1 unit).	RF pickup probe, rectified by a diode and amplified in Power Amp 1 Unit.
RAMP2W	RF Power Output 2	AN (Watts)	0 Counts = 0.0 Watts; 148.5 Counts = 11.7 Watts.	Transmitter RF bus to (fuses to series reg. in power amp 2 unit).	RF pickup probe, rectified by a diode and amplified in Power Amp 2 Unit.

APPENDIX A: TELEMETRY REFERENCE DATA (By S/C Subsystem Grouping)

SUBSYSTEM: (A.7) Communications

TM MNE-MONIC	TELEMETRY TITLE	TM TYPE & (UNITS)	TELEMETERED RANGE	POWER SOURCE	SOURCE DERIVATION AND REMARKS
RAMP3W	RF Power Output 3	AN (Watts)	0 Counts = 0.0 Watts; 154 Counts = 10.5 Watts.	Transmitter RF bus to (fuses to series reg. in Power Amp 3 Unit).	RF pickup probe, rectified by a diode and amplified in Power Amp 3 Unit.
RAMP4W	RF Power Output 4	AN (Watts)	0 Counts = 0.0 Watts; 200 Counts = 12.1 Watts.	Transmitter RF bus to (fuses to series reg. in power amp 4 Unit).	RF pickup probe, rectified by a diode and amplified in Power Amp 4 Unit.
RSPE1V	Receiver 1 Static Phase Error	AN (kHz)	34.85 Counts = +200.0 kHz; 209.6 Counts = -200.0 kHz.	Essential bus to (fuses to power converter to +9V & -6V lines in RCVR 1).	Resistive voltage divider at output of active loop filter in RCVR 1.
RSPE2V	Receiver 2 Static Phase Error	AN (kHz)	36.8 Counts = +200.0 kHz; 208.75 Counts = -200 kHz.	Essential bus to (fuses to power converter to +9V & -6V lines in RCVR 2).	Resistive voltage divider at output of active loop filter in RCVR 2.
RVCO1T	Receiver 1 VCO Temperature	AC (^{o}F)	93 Counts = -22^{o}F 191 Counts = +158^{o}F	Sw. loads bus to DIM 3 & precision 1.0 milliamp output.	Resistor, P/N TI RTH - 42.
RVCO2T	Receiver 2 VCO Temperature	AC (^{o}F)	78 Counts = -22^{o}F 182 Counts = +176^{o}F	Sw. loads bus to DIM 5 & precision 1.0 milliamp output.	Resistor, P/N TI RTH - 42.
ROSC1T	Transmitter Auxiliary Oscillator 1 Temperature	AC (^{o}F)	84 Counts = -22^{o}F 176 Counts = +176^{o}F	Sw. loads bus to DIM 3 & precision 1.0 milliamp output.	Resistor, P/N TI RTH - 42.
ROSC2T	Transmitter Auxiliary Oscillator 2 Temperature	AC (^{o}F)	90 Counts = -22^{o}F 210 Counts = +176^{o}F	Sw. loads bus to DIM 5 & precision 1.0 milliamp output.	Resistor, P/N TI RTH - 42.

APPENDIX A: TELEMETRY REFERENCE DATA (By S/C Subsystem Grouping)

SUBSYSTEM: (A.7) Communications

TM MNE-MONIC	TELEMETRY TITLE	TM TYPE & (UNITS)	TELEMETERED RANGE	POWER SOURCE	SOURCE DERIVATION AND REMARKS
RXMTRT	X-Band Transmitter Temperature	AC (°F)	114 Counts = +39°F 162 Counts = +120°F	Sw. loads bus to DIM 5 & precision 1.0 milliamp output.	Thermistor, P/N 908631-32.
RXMTRW	X-Band Transmitter RF Output	AN (Watts)	0 Counts = 0 Watts 143.5 Counts = 1.24 Watts (@210.045 MHz & +86°F). 144.35 Counts = 1.28 Watts (@ 211.035 MHz)	Transmitter bus to (fuse to reg. in X-Band Transmitter).	RF pickup probe, rectified by a diode and amplified in X-band Transmitter Unit.
RLOK1S	Receiver 1 Phase Lock Status	BL (None)	0 ≡ Not Locked; 1 ≡ Locked.	Essential bus to (fuse to power converter to +9V & -6V in RCVR 1).	Resistive voltage divider driven by LM 108A trigger circuit in RCVR 1.
RLOK2S	Receiver 2 Phase Lock Status	BL (None)	0 ≡ Not Locked; 1 ≡ Locked.	Essential bus to (fuse to power converter to +9V & -6V RCVR 2).	Resistive voltage divider driven by LM 108A trigger circuit in RCVR 2.
RCOH1S	Exciter 1 Inhibit/Restore Coherent Mode Status	BL (None)	0 ≡ Restore; 1 ≡ Inhibit.	Transmitter bus to (fuse to power converter to +9V & -6V in Exciter 1).	State of S/R Flip-flop in Exciter control module in Transponder 1.
RCOH2S	Exciter 2 Inhibit/Restore Coherent Mode Status	BL (None)	0 ≡ Restore; 1 ≡ Inhibit.	Transmitter bus to (fuse to power converter to +9V & -6V in Exciter 2).	State of S/R Flip-flop in Exciter control module in Transponder 2.
RXCT1S	Exciter ON/OFF	BL (None)	0 ≡ OFF; 1 ≡ ON.	Transmitter bus to (fuse to power converter to +9V in Exciter 1).	Resistive voltage divider in CMD interface module in Transponder 1.

SUBSYSTEM: (A.7) Communications

TM MNE-MONIC	TELEMETRY TITLE	TM TYPE & (UNITS)	TELEMETERED RANGE	POWER SOURCE	SOURCE DERIVATION AND REMARKS
RXCT2S	Exciter 2 ON/OFF	BL (None)	0 ≡ OFF; 1 ≡ ON.	Transmitter bus to (fuse to power converter to +9V in Exciter 2).	Resistive voltage divider in CMD interface module in Transponder 2.
RRCVRS	Receiver Normal/ Reverse Switch Position	BL (None)	0 ≡ Reverse (aft omni to RCVR 1; a despun antenna to RCVR 2); 1 ≡ Normal (aft omni to RCVR 2); a despun antenna to RCVR 1).	Essential bus to (fuse in switch driver).	State of S/R flip-flop in switch driver, intended to indicate state of DPDT RF transfer switch between receivers and antennas. (See Note 3
RXCTRS	Exciter 1/2 Switch Position	BL (None)	0 ≡ Exciter 2 Selected; 1 ≡ Exciter 1 Selected.	Essential bus, to (fuse in switch driver)	State of S/R flip-flop in switch driver intended to indicate state of SPDT RF Transfer switch between Exciters & Power amps. (See Note 3).
RAMP1S	Amp 1/2 to Hi/ Low Power Status	BL (None)	0 ≡ Amp 2 to Hi power output line (3 dB Hybrid); Amp 1 to Low Power output line). 1 ≡ Amp 1 to Hi power output line; Amp 2 to Lo power output line.	Essential bus, to (Fuse in switch driver).	State of S/R flip-flop in switch driver intended to indicate state of DPDT RF Transfer switch between Amps and output lines to antennas (See Note 3).
RAMP3S	Amplifier 3/4 Switch Position	BL (None)	0 ≡ Amp 4 Selected; 1 ≡ Amp 3 Selected.	Essential bus, to (Fuse in switch driver).	State of S/R flip-flop in switch driver intended to indicate state of SPDT RF Transfer switch between Amps & Hi Power (3 dB Hybrid) output line (See Note 3).
RANT3S	High Gain Backup/Forward Omni Switch Position	BL (None)	0 ≡ Omni Selected; 1 ≡ High Gain Backup Selected.	Essential bus, to (Fuse in switch driver).	State of S/R flip-flop in switch driver intended to indicate state of SPDT RF Transfer Switch between input/output RF line on the despun platform and the two "low gain" antennas. (See Note 3).

APPENDIX A: TELEMETRY REFERENCE DATA (By S/C Subsystem Grouping)

SUBSYSTEM: (A.7) Communications

TM MNE-MONIC	TELEMETRY TITLE	TM TYPE & (UNITS)	TELEMETERED RANGE	POWER SOURCE	SOURCE DERIVATION AND REMARKS
RANT1S	Aft Omni/ Despin Antenna to Hi/Low Power Status	BL (None)	0 = Lo Power output line to Aft Omni; Hi Power output line to a Despun Antenna. 1 = Hi Power output line to Aft Omni; Lo Power output line to a despun Antenna.	Essential bus, to (Fuse in switch driver).	State of S/R flip-flop in switch driver intended to indicate state of DPDT RF Transfer Switch between output lines and antennas (See Note 3)
RANT2S	High Gain Antenna/Low Gain Antenna Switch Position	BL (None)	0 = "Low Gain" antenna (Hi Gain Backup or Forward Omni) selected; 1 = HGA Selected.	Essential bus, to (Fuse in switch driver).	State of S/R flip-flop in switch driver intended to indicate state of SPDT RF Transfer Switch between input/output RF line on despun platform and the two categories of antennas (See Note 3).

SUBSYSTEM: (A.8) Power

TM MNE-MONIC	TELEMETRY TITLE	TM TYPE & (UNITS)	TELEMETERED RANGE	POWER SOURCE	SOURCE DERIVATION AND REMARKS
PB1P1T	Battery 1 Pack 1 Temperature	AC (^{o}F)	41 Counts = +86^{o}F 142 Counts = +32^{o}F	Sw. loads bus to DIM 1 & precision 1.0 milliamp output.	Thermistor, 908635-4.
PB1P2T	Battery 1 Pack 2 Temperature	AC (^{o}F)	41 Counts = +86^{o}F 142 Counts = +32^{o}F	Sw. loads bus to DIM 6 & precision 1.0 milliamp output.	Thermistor, 908635-4.
PB2P1T	Battery 2 Pack 1 Temperature	AC (^{o}F)	41 Counts = +86^{o}F 142 Counts = +32^{o}F	Sw. loads bus to DIM 1 & precision 1.0 milliamp output.	Thermistor, 908635-4.
PB2P2T	Battery 2 Pack 2 Temperature	AC (^{o}F)	41 Counts = +86^{o}F 142 Counts = +32^{o}F	Sw. loads bus to DIM 6 & precision 1.0 milliamp output.	Thermistor, 908635-4.
PB1R1S	Battery 1, Relay 1 Status	BL (None)	0 ≡ Lo Charge Selected; 1 ≡ Hi Charge Selected.	Battery 1 or boost charge array 1 downstream of K1 & K2 wiper arms in chg/dischg controller 1.	Resistive divider, between Hi terminal of one pole of DPDT latching relay (K2) in charge/discharge controller 1 and signal ground.
PB2R1S	Battery 2, Relay 1 Status	BL (None)	0 ≡ Lo Charge Selected; 1 ≡ Hi Charge Selected.	Battery 2 or boost charge array 2 downstream of K1 & K2 wiper arms in chg/dischg controller 2.	Resistive divider, between Hi terminal of one pole of DPDT latching relay (K2) in charge/discharge controller 2, and signal ground.
PBAT1V	Battery 1 Terminal Voltage	AN (Volts)	0 Counts = 0 VDC 256 Counts = 40.0 VDC	Battery 1 or boost charge Array 1 downstream of switching relays in charge/discharge controller 1.	Resistive divider, between positive terminal of Battery 1 & signal ground.

APPENDIX A: TELEMETRY REFERENCE DATA (By S/C Subsystem Grouping)

SUBSYSTEM: (A.8) Power

TM MNE-MONIC	TELEMETRY TITLE	TM TYPE & (UNITS)	TELEMETERED RANGE	POWER SOURCE	SOURCE DERIVATION AND REMARKS
PBAT2V	Battery 2 Terminal Voltage	AN (Volts)	0 Counts = 0 VDC 256 Counts = 40.0 VDC	Battery 2 or boost charge array 2 downstream of switching relays in Chg/disch. Controller 2.	Resistive divider, between positive terminal of Battery 2 & signal ground.
PB1R2S	Battery 1, Relay 2 Status	BL (None)	0 ≡ Lo Charge Selected; 1 ≡ Hi Charge Selected.	Battery 1 or boost charge array 1, downstream of K1 & K2 wiper arms in chg/dischg. controller 1.	Resistive divider, between Hi terminal of one pole of DPDT latching relay (K1) in charge/discharge controller 1, and signal ground.
PB2R2S	Battery 2, Relay 2 Status	BL (None)	0 ≡ Lo Charge Selected; 1 ≡ Hi Charge Selected.	Battery 2 or boost charge array 2, downstream of K1 & K2 wiper arms in charge/discharge controller 2.	Resistive divider, between Hi terminal of one pole of DPDT latching relay (K1) in charge/discharge controller 2, and signal ground.
PHTTKS	Primary/ Secondary Propulsion Tank Heaters Status	BL (None)	0 ≡ Secondary heaters selected; 1 ≡ Primary heaters selected.	Essential bus, downstream of heaters select relay.	Resistive divider between power line to primary heaters (& upstream of fuses) and signal ground (in PIU).
PLDBUS	Switched Loads Bus Reset Relay Status	BL (None)	0 ≡ Switched loads UV/OL Relay has tripped Bus OFF. 1 ≡ Switched loads UV/OL Relay has not tripped (Bus ON).	Switched loads bus downstream of UV/ OL switch (K2).	Resistive divider between terminal of one pole of DPST (Normally closed) latching relay (K2) controlled by UV/OL trip circuit, and signal ground.
PLIM1S	Bus Limiter 1 Enable/Disable	BL (None)	0 ≡ Disabled; 1 ≡ Enabled.	Essential bus (Main array output).	Resistive divider, between terminal of SPST latching relay and signal ground, in Bus Limiter 1.

SUBSYSTEM: (A.8) Power

TM MNE-MONIC	TELEMETRY TITLE	TM TYPE & (UNITS)	TELEMETERED RANGE	POWER SOURCE	SOURCE DERIVATION AND REMARKS
PLIM2S	Bus Limiter 2 Enable/Disable	BL (None)	0 ≡ Disabled; 1 ≡ Enabled.	Essential Bus (Main array output)	Resistive divider, between terminal of SPST latching relay and signal ground, in Bus Limiter 2.
PLIM3S	Bus Limiter 3 Enable/Disable	BL (None)	0 ≡ Disabled; 1 ≡ Enabled.	Essential Bus (Main array output)	Resistive divider, between terminal of SPST latching relay and signal ground, in Bus Limiter 3.
PLIM4S	Bus Limiter 4 Enable/Disable	BL (None)	0 ≡ Disabled; 1 ≡ Enabled.	Essential Bus (Main array output)	Resistive divider, between terminal of SPST latching relay and signal ground, in Bus Limiter 4.
PLIM5S	Bus Limiter 5 Enable/Disable	BL (None)	0 ≡ Disabled; 1 ≡ Enabled.	Essential Bus (Main array output)	Resistive divider, between terminal of SPST latching relay and signal ground, in Bus Limiter 5.
PLIM6S	Bus Limiter 6 Enable/Disable	BL (None)	0 ≡ Disabled; 1 ≡ Enabled.	Essential Bus (Main array output)	Resistive divider, between terminal of SPST latching relay and signal ground, in Bus Limiter 6.
PLIM7S	Bus Limiter 7 Enable/Disable	BL (None)	0 ≡ Disabled; 1 ≡ Enabled.	Essential Bus (Main array output)	Resistive divider, between terminal of SPST latching relay and signal ground, in Bus Limiter 7.
PLIMTI	Bus Voltage Limiter Current	AN (Amps)	0 Counts = 0.0 Amps; 256 Counts = +12.0 Amps.	Essential Bus (Main array output to all Bus Limiters).	Dual 50 mV current shunt and a hybrid differential amplifier.
PPAN1T	Solar Panel Temperature 1	AC (^{o}F)	38.165 Counts = -298^{o}F 215.475 Counts = +212^{o}F	Sw. loads bus to DIM 3 & precision 1.0 milliamp output.	Platinum resistor, 908683-4.
PPAN2T	Solar Panel Temperature 2	AC (^{o}F)	38.165 Counts = -298^{o}F 215.475 Counts = +212^{o}F	Sw. loads bus to DIM 7 & precision 1.0 milliamp output.	Platinum resistor, 908683-4.

APPENDIX A: TELEMETRY REFERENCE DATA (By S/C Subsystem Grouping)

SUBSYSTEM: (A.8) Power

TM MNE-MONIC	TELEMETRY TITLE	TM TYPE & (UNITS)	TELEMETERED RANGE	POWER SOURCE	SOURCE DERIVATION AND REMARKS
PPANLI	Solar Panel Current	AN (Amps)	0 Counts = 0.0 Amps; 256 Counts = +18.0 Amps.	Essential bus at output of main array.	Dual 50 mV current shunt and a hybrid differential amplifier.
PREG1S	Primary/ Redundant Discharge Regulator 1 Status	BL (None)	0 ≡ Redundant selected in Controller #1. 1 ≡ Primary selected in Controller #1.	Essential bus downstream of primary discharge regulator in Charge/Discharge Controller #1.	Resistive divider, between output of Primary Discharge Regulator and Signal Ground (in Charge/Discharge Controller #1).
PREG2S	Primary/ Redundant Discharge Regulator 2 Status	BL (None)	0 ≡ Redundant selected in Controller #2. 1 ≡ Primary selected in Controller #2.	Essential bus downstream of primary discharge regulator in Charge/Discharge Controller #2.	Resistive divider, between output of primary discharge regulator & signal ground (in charge/discharge Controller #2).
PRFBUS	RF XMTR Bus Reset Relay Status	BL (None)	0 ≡ XMTR bus UV/OL Relay has tripped Bus OFF; 1 ≡ XMTR bus UV/OL Relay has not tripped (Bus ON).	XMTR Bus, downstream of UV/OL switch (K3).	Resistive divider between terminal of one pole of DPST (normally closed) latching relay (K3) controlled by UV/OL trip circuit, and signal ground.
PSCBUS	Science Bus Reset Relay Status	BL (None)	0 ≡ Science Bus UV/OL has tripped Bus OFF 1 ≡ Science Bus UV/OL Relay has not tripped (Bus ON).	Science Bus, downstream of UV/OL switch (K1).	Resistive divider between terminal of one pole of DPST (normally closed) latching relay (K1) controlled by UV/OL trip circuit, and signal ground.
PSCICI	Science Current	AN (Amps)	0 Counts = 0 Amps 256 Counts = 4.5 Amps	Science Bus, downstream of UV/OL switch.	Dual 50 mV current shunt and a hybrid differential amplifier.
PSPROS	Power System Protection ON/ OFF	BL (None)	0 ≡ OFF (UV/OL Logic bypassed; all buses ON). 1 ≡ ON (UV/OL Logic controls ON/OFF state for each of three buses).	Essential Bus, downstream of Relay K5 (in UV/OL switch unit).	For DIM2: Resistive divider between ON terminal of SPST latching Relay (K5) & signal ground (in UV/OL sw. Unit). For DIM4: Separate resistive divider from above, tied to same points as for divider above.

SUBSYSTEM: (A.8) Power

TM MNE-MONIC	TELEMETRY TITLE	TM TYPE & (UNITS)	TELEMETERED RANGE	POWER SOURCE	SOURCE DERIVATION AND REMARKS
PRCHGS	Precharge ON/ OFF Status	BL (None)	0 = OFF; 1 = ON.	Essential bus downstream of 1 amp current limited switch (in UV/OL Switch Unit).	Resistive divider between output of 1 amp current limited switch and signal ground (in UV/OL Switch Unit).
PBUSLI	Spacecraft Loads Current	AN (Amps)	0 Counts = 0.0 Amps; 256 Counts = +18.0 Amps.	Essential bus, just upstream from fan-out to four buses.	Dual 50 mV current shunt and a hybrid differential amplifier.
PCHG1I	Battery 1 Charge Current	AN (Amps)	0 Counts = 0 Amps; 256 Counts = 1.0 Amps.	At output of boost charge array 1 & downstream of chg. switching relays in chg/disch controller #1.	Dual 50 mV current shunt and a hybrid differential amplifier (in Charge/Discharge Controller 1).
PCHG2I	Battery 2 Charge Current	AN (Amps)	0 Counts = 0 Amps; 256 Counts = 1.0 Amps.	At output of boost charge array 2, & downstream of chg. switching relays in chg/disch.controller #2.	Dual 50 mV current shunt and a hybrid differential amplifier (in charge/discharge controller #2).
PDIS1I	Battery 1 Discharge Current	AN (Amps)	0 Counts = 0 Amps; 256 Counts = 12.0 Amps.	Battery 1 between Battery 1 & disch. Regulators in chg/ disch controller #1.	Dual 50 mV current shunt and a hybrid differential amplifier (in charge/discharge controller #1).
PDIS2I	Battery 2 Discharge Current	AN (Amps)	0 Counts = 0 Amps 256 Counts = 12.0 amps.	Battery 2 between battery 2 & disch. Regs. in chg/disch. controller #2.	Dual 50 mV current shunt and a hybrid differential amplifier (in charge/discharge controller #2).

APPENDIX A: TELEMETRY REFERENCE DATA (By S/C Subsystem Grouping)

SUBSYSTEM: (A.5) Power

TM MNE-MONIC	TELEMETRY TITLE	TM TYPE & (UNITS)	TELEMETERED RANGE	POWER SOURCE	SOURCE DERIVATION AND REMARKS
PEBUSV	Essential Bus Voltage	AN (Volts)	0 Counts = 0.0 Volts; 255.5 Counts = +30.8 Volts.	Essential bus (In UV/OL Switch Unit)	For DIM3: Resistive divider between essential bus & signal ground (In UV/OL Switch Unit). For DIM5: Separate resistive divider from above, tied to same points as for divider above.
PHT1MS	OIM Heater 1 ON/OFF	BL (None)	0 = OFF; 1 = ON.	Essential bus, downstream of two DPST relays in series (heater select relays).	Resistive divider between terminal of one pole of DPST latching relay and signal ground (In PIU).
PHT2MS	OIM Heater 2 ON/OFF	BL (None)	0 = OFF; 1 = ON.	Essential bus, downstream of two DPST relays in series (heater select relays).	Resistive divider between terminal of one pole of DPST latching relay and signal ground (In PIU).
PHTJ1S	Radial Jets 1 & 2 ON/OFF	BL (None)	0 = OFF; 1 = ON.	Essential bus, downstream of two DPST relays in series (heater select relays).	Resistive divider between terminal of one pole of DPST latching relay and signal ground (In PIU).
PHTJ3S	Radial Jets 3 & 4 Heaters ON/OFF	BL (None)	0 = OFF; 1 = ON.	Essential bus, downstream of single DPST relay (heater select relay).	Resistive divider between terminal of one pole of DPST latching relay and signal ground (In PIU).

NOTES

NOTE 1: AN = Analog = 8 Bits
BL = Bilevel = 1 Bit
SD = Serial Digital (Various numbers of Bits)

NOTE 2: All analog telemetered ranges are 0 counts. The data shown are the maximum and minimum points of actual calibration data used in the Orbiter TM Conversion Handbook, Reference: Paragraph 1.5.16.

NOTE 3: Flip-flop resets for power source OFF condition, whereas RF Xfer switch remains latched to last commanded state.

APPENDIX B

TELEMETRY REFERENCE DATA

(MNEMONICS IN ALPHABETICAL ORDER)

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APPENDIX B: TELEMETRY REFERENCE DATA
(MNEMONICS IN ALPHABETICAL ORDER)

MNEMONIC	DESCRIPTION	APPL S/C	FORMATS	WORD (BIT)	DATA TYPE	STATUS 1BIT -	STATUS 0 BIT -	STATUS MULT	E.V. CONVERSION UNIT	E.V. CONVERSION QUANTIZ	RANGE	SAMPLE TIME	REMARKS/REF
AACSAD	ACS Angle Delay Magnitude	B,O	BACS, OACS	18(6-7),19	SD	-	-	-	DEG	.352	.352-360 DEG		18(6)=LSB
AADP2S	Alt. Data Processor 1 Power Status	B,O	BSUBA, OSUBA	3(0),3(0)	BL	ON	OFF	-	-	-	-	-	
AADP2S	" " " 2 " "	B,O	, OSUBA	3(1),3(1)	BL	ON	OFF	-	-	-	-	-	
ABAP1T	BAPTA Inner Race Bearing Temp	O	SUBB	19	AC	-	-	-	°C	-	+49.0 to +6.0	-	
ABAP0T	" Outer " " "	O	SUBB	20	AC	-	-	-	°C	-	+70.0 to -20.0	-	
ADCE1T	Despin Control Electronics 1 Temp	O	SUBB	40	AC	-	-	-	°C	-	-10.0 to + 50.0	-	
ADCE2T	" " " " 2 "	O	SUBB	48	AC	-	-	-	°C	-	-10.0 to + 70.0	-	
ADCE1V	" " " " 1 Pwr. Supply Volt.	6	SUBB	41	AH	-	-	-	Volts	-	0.0 to 5.1	-	
ADCE2V	" " " " 2 " " "	O	SUBB	49	AH	-	-	-	Volts	-	0.0 to 5.1	-	
ADSPRS	" Phase Reset Status	O	SUBB	40(0)	SD	ENAB	DSAB	-	-	-	-	-	
ADVANS	SRB Advance Status	B,O	BACS, OACS	16(4)	SD	ADV	REST	-	-	-	-	-	
AELEVD	HGA Elevation Position (Dish)	O	LACR, ERGA,	23,55	AH	-	-	-	DEG	-	±74.0°	-	
		O	SUBB	13,45	AH	-	-	-	DEG	-	±16.0°	-	
AJCE1S	Jet Control Electronics 1 Buffer Output Status	B,O	BSUBA, OSUBA	3(2)	BL	ON	OFF	-	-	-	-	-	
AJCE2S	" " " " 2 " " "	B,O	BSUBA, OSUBA	3(3)	BL	ON	OFF	-	-	-	-	-	
AJCEFS	" " " Fire Status	B,O	BACS, OACS	15(0)	BL	ENAB	DSAB	-	-	-	-	-	
AJETMS	Jet Fire Mode Status	B,O	BACS, OACS	13(6)+14(0)	SD	00=APUL	01=HPUL	11=CUBT	-	-	-	-	
AJET1S	Radial Jet 1 Status	B,O	BACS,OACS	14(1)	SD	ENAB	DSAB	-	-	-	-	-	
AJET2S	" " 2 "	B,O	BACS, OACS	14(2)	SD	ENAB	DSAB	-	-	-	-	-	
AJET3S	" " 3 "	B,O	BACS, OACS	14(3)	SD	ENAB	DSAB	-	-	-	-	-	
AJET4S	" " 4 "	B,O	BACS, OACS	14(4)	SD	ENAB	DSAB	-	-	-	-	-	
AJET5S	Axial Jet 5 Status	B,O	BACS, OACS	14(5)	SD	ENAB	DSAB	-	-	-	-	-	
AJET6S	" " 6 "	B,O	BACS, OACS	14(6)	SD	ENAB	DSAB	-	-	-	-	-	
AJET7S	" " 7 "	O	ACS	14(7)	SD	ENAB	DSAB	-	-	-	-	-	
AJMAGC	JCE Countdown (Magnitude)	B,O	BACS, OACS	12+13(0-3)	SD	-	-	-	Counts Sec		0-900C 0-2096.645	-	
ALOLES	PLL Loss-Of-Lock Status	B,O	BACS, OACS	18(0)	SD	LOCK	LOSS	-	-	-	-	-	
ALOLES	PLL Loss-Of-Lock Enable Status	B,O	BACS, OACS	16(2)	SD	ENAB	INHB	-	-	-	-	-	
AJETCD	Time From SRB to Jet Firing	B,O	N/A	N/A	CALC	-	-	-	DEG		0-360	-	08/3.4.10.1
AJETCR	Thruster Firings	B,O	N/A	N/A	CALC	-	-	-	Counts	-	-	-	08/3.4.4.10.2

APPENDIX B: TELEMETRY REFERENCE DATA
(MNEMONICS IN ALPHABETICAL ORDER)

MNEMONIC	DESCRIPTION	APPL S/C	FORMATS	WORD (BIT)	DATA TYPE	STATUS			E.V. CONVERSION		RANGE	SAMPLE TIME	REMARKS/REF
						1BIT =	"0" BIT =	MULT.	UNIT	QUANTIZ.			
AA1ADS	Attitude Measurement 1 Address	B,0	BETY,	18(0-2)	SD	000=0=SAR(Sel. Roll Ref)			-	-	-	-	Bit 0 = LSB
			BACS, OACS	10(0-2)		100-1=M	S(Minor Fr.Sync		-	-	-	-	
						010-2=R	P(Roll ndx. Pul		-	-	-	-	
						110-3=M	P(Mast. ndx Pul)		-	-	-	-	
						001=4=GYB	(Gated Star B)		-	-	-	-	
						101=5=	S25 (Ps 2)		-	-	-	-	
						011=6=MAS	(Maj. Fr. Sync		-	-	-	-	
AA2ADS	Attitude Measurement 2 Address	B,0	BETY,	18(3-5)	SD	000=0=SAR	(Sel. Roll Ref)		-	-	-	-	Bit 3 = LSB
			BACS,OACS	10(3-5)		100=1=M	S(Minor Fr.Sync		-	-	-	-	
						010=2=R	P(Roll ndx Pul		-	-	-	-	
						110=3=M	P(Mast. ndx Pul		-	-	-	-	
						001=4=GYB	(Gated Star B)		-	-	-	-	
						101=5=	S25 (Ps 2)		-	-	-	-	
						011=6=MAS	(Maj. Fr. Sync)		-	-	-	-	
AAAGCS	Jet Pulse/Time Count Select Status	B,0	BACS,OACS	13(5)	SD	PULS	Time	-	-	-	-	-	
AAISSC	Missed SAR Count	B,0	BACS, OACS	18(1-4)	SD	-	-	-	Counts	1	0-8 Counts	-	Bit 1 = LSB
AAMTR1	Gospin Motor Drive Current	0	ENG,SUBB,	59,42	AN	-	-	-	Amps	..	0.0 to 1.2	-	
AAMTR1S	" " " 1 Status	0	SUBA,	3(6)	BL	ON	OFF	-	-	-	-	-	
AAMTR2S	" " " 2 Status	0	SUBA,	3(7)	BL	ON	OFF	-	-	-	-	-	
APRELS	Small Probe Release Pulse Status	B,0	BACS,OACS	16(5)	BL	GEN	NGEN	-	-	-	-	-	
APOSMT	Hi Gain Antenna Position Motor Temp	0	SUBB	44	AC	-	-	-	°C	-	-180.0 to +100.0	-	
APOSTQ	Position Torque	0	SUBB,ENG	50,61	AN	-	-	-	FLBS	-	0.35 to 0.0		
APOS1S	Hi Gain Dish Positioner Electronics 1 Pwr. Status	0	SUBA	3(4)	BL	ON	OFF	-	-	-	-	-	
APOS2S	" " " " " 2 " "	0	SUBA	3(5)	BL	ON	OFF	-	-	-	-	-	
APULLS	Jet Pulse Width Select Status	B,0	BACS,OACS	13(4)	SD	512ms	128ms	-	-	-	-	-	
ARATEQ	Rate Torque	0	ENG,SUBB	60,43	AN	-	-	-	RPM	-	25.0 to 3.0	-	2 Ranges 7-25
ARIPAD	Roll Index Delay Magnitude	B,0	BACS,OACS	16(6)-17(7)	SD	-	-	-	DEG	.351	0-359.65		Bit 16(6) = LSB
ASBETS	Spin Rate Detector Status	B,0	BACS,OACS	13(7)	BL	ENAB	INHB	-	-	-	-	-	
ASG*BD	Time From SAR to Gated Star B (Angle)	B,0	N/A	N/A	CALC	-	-	-	DEG			-	
ASG*BZ	Time From SAR to Gated St B (Magnitude)	B,0	N/A	N/A	CALC	-	-	-	SEC			-	

APPENDIX B: TELEMETRY REFERENCE DATA
(MNEMONICS IN ALPHABETICAL ORDER)

MNEMONIC	DESCRIPTION	APPL S/C	FORMATS	WORD (BIT)	DATA TYPE	STATUS			E.V. CONVERSION		RANGE	SAMPLE TIME	REMARKS/REF
						1BIT =	"0" BIT =	MULT	UNIT	QUANTIZ			
ASIMSZ	Simulated Spin Period Magnitude	B,O	BACS,OACS	20(8)-21(7)	SD	-	-	-	Counts SEC	.25ms	0-65535 0-16 Bits		20(0) = LSB
ASMASZ	Time From SRB To MAS (Major Frame Pulse)	B,O	N/A	N/A	CALC	-	-	-	SEC			-	
ASRIPO	Angle From SRB To MIP (Master Index Pulse	B,O	N/A	N/A	CALC	-	-	-	DEG			-	.04/3.6.4.7.2
ASMISZ	Time From SRB to MIS (Minor Frame Rate)	B,O	N/A	N/A	CALC	-	-	-	SEC			-	
ASPAVZ	Averaged Spin Period	B,O	N/A	N/A	CALC	-	-	-	SEC		# 0-8.85 RPM		
ASPIRS	PLL Spin Range Select	B,O	BACS,OACS	BACS,OACS	SD	-	-	00	RPM	-	0.0-17.0 RPM		
						-	-	10	RPM	-	16.0-35.4 RPM		
						-	-	01	RPM	-	32-70.8 RPM		
						-	-	11	RPM	-			.04/3.6.4.5.2
ASPINZ	Non-Averaged Spin Period (SRB to SRB)	B,O	N/A	N/A	CALC	-	-	-	SEC			-	
ASRIPO	Angle from SRB to RIP (Roll Index Pulse)	B,O	N/A	N/A	CALC	-	-	-	DEG			-	.04/3.6.4.6.2
ASRRRS	Selected Roll Reference Select Status	B,O	BACS,OACS	15(5-6)	SD	SIM(Simulator)=		10	-	-	-	-	
						*** (Star) =		01	-	-	-	-	
						SUN (Sun) =		11	-	-	-	-	
						SIM (Simulator)=		00	-	-	-	-	
ASS2SD	Time From SRB To SUN2 Pulse (Angle)	B,O	N/A	N/A	CALC	-	-	-	DEG				
ASS2SZ	Time From SRB To SUN2 Pulse (Magnitude)	B,O	N/A	N/A	CALC	-	-	-	SEC				
ASTART	Star Sensor Temp	B,O	BETY	19	AC	-	-	-	DEG.C			-	
			BEING,OENG	62	AC	-	-	-	DEG.C			-	
			BSUBB,OSUBB	18	AC	-	-	-	DEG.C			-	
ASUNGS	Sun Gate Status	B,O	BACS,OACS	16(3)	SD	ENAB	DSAB	-	-	-	-	-	
ASUNSS	Sun Sensor Select Status	B,O	BACS,OACS	15(3-4)	SD	MIDR(Midrange)=		00	-	-	-	-	
					SD	EXRL(Lower)=		01	-	-	-	-	Extended Range-
					SD	EXRU(Upper)=		10	-	-	-	-	Extended Range
					SD	INV(Test)=		11	-	-	-	-	
ATTMIZ	Attitude Measurement Magnitude	B,O	BETY	16/17	SD	-	-	-	Counts SEC	0.25 msec	0-65535 0-16 Bits		16(0) = LSB
ATTM2Z			BENG,OLACB,OENG	10/11,42/43		-	-	-	"	"	"		
			BACS,OACS	8/9,24/25		-	-	-	"	"	"		
				40/41		-	-	-	"	"	"		
				56/57		-	-	-	"	"	"		
			BSUBB,OSUBB	0/1,32/33		-	-	-	"	"	"		

APPENDIX B: TELEMETRY REFERENCE DATA
(MNEMONICS IN ALPHABETICAL ORDER)

MNEMONIC	DESCRIPTION	APPL S/C	FORMATS	WORD (BIT)	DATA TYPE	STATUS			E.V. CONVERSION		RANGE	SAMPLE TIME	REMARKS/REF
						1BIT =	'0' BIT =	MULT	UNIT	QUANTIZ			
ATTDSS	Attitude Measurement Selected	B,O	BETY,	18(6)	SD	1	2	-	-	-	-		
			BENG,GLACR,DENG	12(6),	SD	1	2	-	-	-	-		
				44(6)	SD	1	2	-	-	-	-		
			BACS,OACS	10(6)	SD	1	2	-	-	-	-		
				26(6),	SD	1	2	-	-	-	-		
				42(6),	SD	1	2	-	-	-	-		
				58(6)	SD	1	2	-	-	-	-		
			BSUBB,OSUBB	2(6),34(6)	SD	1	2	-	-	-	-		
AVPDRD	DCE Variable Phase Delay 1 Magnitude (RAM)	O	OSUBB	27-28(0-3)	SD	-	-	-					27(0) = MSB
AVPDRE	DCE Variable Phase Delay 2 Magnitude (BASIR)	O	OSUBB	28(4)-29(7)	SD	-	-	-					28(4) = MSB
A*ACQS	Star Acquisition Status	B,O	BACS,OACS	15(7)	BL	STAR	WORM	-	-	-	-	-	
A*GASS	Star Gate A Channel Select Status	B,O	BACS,OACS	15(1)	BL	ONE	TWO	-	-	-	-	-	
A*GBSS	Star Gate B Channel Select Status			15(2)	BL	ONE	TWO	-	-	-	-	-	
A*1BPS	Star 1 Bandpass Status	B,O	BSUBA,OSUBA	10(0)	BL	TWO	19	-	-	-	-	-	
A*2BPS	Star 2 Bandpass Status	B,O	BSUBA,OSUBA	10(1)	BL	TWO	19	-	-	-	-	-	
A*1BRT	Star 1 Brightness	B,O	BENG	23,	AM	-	-	-	VOLTS	-	0.0 to 5.1		
			BACS,OACS	27,	AM	-	-	-	VOLTS	-	"		
			BSUBB,OSUBB	14	AM	-	-	-	VOLTS	-	"		
A*2BRT	Star 2 Brightness	B,O	BENG	55,	AM	-	-	-	VOLTS	-	"		
			BACS,OACS	59,	AM	-	-	-	VOLTS	-	"		
			BSUBB,OSUBB	15,	AM	-	-	-	VOLTS	-	"		
A*1THS	Star 1 Threshold Setting	B,O	BENG	29,	AM	-	-	-	VOLTS	-	0.0 to 5.1		
			BACS,OACS	28,	AM	-	-	-	VOLTS	-	"		
			BSUBB,OSUBB	16	AM	-	-	-	VOLTS	-	"		
A*2THS	Star 2 Threshold Setting	B,O	BENG	36,	AM	-	-	-	VOLTS	-	"		
			BACS,OACS	60,	AM	-	-	-	VOLTS	-	"		
			BSUBB,OSUBB	17	AM	-	-	-	VOLTS	-	"		
A*1PWS	Star 1 Power Status	B,O	BSUBA,OSUBA	3(4),7(6)	BL	ON	OFF	-	-	-	-	-	
A*2PWS	Star 2 Power Status	B,O	BSUBA,OSUBA	3(5),7(1)	BL	ON	OFF	-	-	-	-	-	

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APPENDIX B: TELEMETRY REFERENCE DATA
(MNEMONICS IN ALPHABETICAL ORDER)

MNEMONIC	DESCRIPTION	APPL S/C	FORMATS	WORD (BIT)	DATA TYPE	STATUS 1BIT	STATUS "0" BIT	STATUS MULT	E.V CONVERSION UNIT	E.V CONVERSION QUANTIZ	RANGE	SAMPLE TIME	REMARKS/REF
C1TIMC	Small Probe 1 Coast Timer Verification	B	BSUBC	21-22	SD	-	-	-					21(0) = LSB
C2TIMC	" " 2 " " "	B		37-38	SD	-	-	-					
C3TIMC	" " 3 " " "	B		53-54	SD	-	-	-					13(1) = MSB
CCMD1C	Command Counter 1	B,D	BENG,DENG,OSUBA,	13(1-6)	SD	-	-	-	CMNDS	7 Bit Bin	0-127	-	
			BSUBA		SD	-	-	-	CMNDS	7 Bit Bin	0-127	-	45(1) = MSB
CCMD2C	Command Counter 2	B,D	BENG,DENG	45(1-6)	SD	-	-	-	CMNDS	7 Bit Bin	0-127	-	
			BSUBA,OSUBA		SD	-	-	-	CMNDS	7 Bit Bin	0-127	-	
CDMD1S	Demodulator 1 Squelch Status	B	BENG,	14(5)	BL	UNSQ	SQLD	-	-	-			
CDMD2S	Demodulator 2 Squelch Status	B	BENG	46(5)	BL	UNSQ	SQLD	-	-	-			
CLOG1S	Stored Command Logic 1 State	B,D	BENG,OLACB,DENG	14(0-3)	SD	-	-	0000 = SEP0 (SEPARATION STATE 0)				-	
			BACS,OACS	30(0-3)	SD	-	-	0001 = TIM0 (TIMED START 0)				-	
			BSUBA,OSUBA	14(0-3)	SD	-	-	0010 = STP1 (STOP 1)				-	
						-	-	0011 = STP2 (STOP 2)				-	
						-	-	0100 = SEP1 (SEPARATION STATE 1)				-	
						-	-	0101 = TIM1 (TIMED START 1)				-	
						-	-	0110 = ADVC (ADVANCE COMPLETE)				-	
						-	-	0111 = INVD (INVALID)				-	
						-	-	1000 = PROC (PROCESS COMMAND)				-	
						-	-	1001 = SUB1 (SUBTRACT 1)				-	
						-	-	1010 = READ (READ)				-	
						-	-	1011 = ADV (ADVANCE)				-	
						-	-	1100 = RUN (RUN				-	
						-	-	1101 = LTIM (LOAD TIMER)				-	
						-	-	1110 = LOAD (LOADING)				-	
						-	-	1111 = STBY (STANDBY)				-	
CLOG2S	Stored Command Logic 2 State	B,D	BENG,OLACB,	46(0-3)	SD	-	-					-	
			DENG,			-	-					-	
			BACS,OACS	62(0-3)	SD	-	-	(Same as Above)				-	
			BSUBA,OSUBA	17(0-3)	SD	-	-					-	
CLSWC	Large Probe Acceleration Switch Status	LP	LDESC,LBLKD	3(1)	BL	ACTV	NACT	-	-	-	-	-	
CLTSWC	Large Probe Thermal Switch Status	LP	LDESC,LBLKD	3(2)	BL	ACTV	NACT	-	-	-	-	-	

APPENDIX B: TELEMETRY REFERENCE DATA
(MNEMONICS IN ALPHABETICAL ORDER)

MNEMONIC	DESCRIPTION	APPL S/C	FORMATS	WORD (BIT)	DATA TYPE	STATUS			E.V. CONVERSION		RANGE	SAMPLE TIME	REMARKS/REF
						1BIT =	"0" BIT =	MULT	UNIT	QUANTIZ			
CMEM1	Command Memory 1 Readout	B,O	BCMRO,OCMRO	9-35	SD	-	-	-	-	-	-	-	
CMEM2	Command Memory 2 Readout	B,O	BCMRO,OCMRO	37-63	SD	-	-	-	-	-	-	-	
CMEM1C	Command Memory 1 Address Pointer	B,O	BCMRO,OCMRO	8(107)	SD	-	-	-	CMDS PROCESSED		0-127 CMDS	-	7-Bit Binary
CMEM2C	Command Memory 2 Address Pointer	B,O	BCMRO,OCMRO	36(1-7)	SD	-	-	-	CMDS PROCESSED		0-127 CMDS	-	Counters
CPWR1S	CMD Processor 1 Power Status	B,O	BSUBA,OSUBA	10(5)	BL	ON	OFF	-	-	-	-	-	
CPWR2S	CMD Processor 2 Power Status	B,O	BSUBA,OSUBA	10(6)	BL	ON	OFF	-	-	-	-	-	
CREJ1S	Command Reject 1 Status	B,O	BENG,OENG	13(0)	SD	REJT	ACPT	-	-	-	-	-	Real-Time Cmnds.
			BSUBA,OSUBA	13(0)	SD	REJT	ACPT	-	-	-	-	-	
			OLACR	13(0)	SD	REJT	ACPT	-	-	-	-	-	
			BACS,OACS	29(0)	SD	REJT	ACPT	-	-	-	-	-	
CREJ2S	Command Reject 2 Status	B,O	BENG,OENG	45(0)	SD	REJT	ACPT	-	-	-	-	-	Real-Time Cmnds
			BSUBA,OSUBA	16(0)	SD	REJT	ACPT	-	-	-	-	-	
			OLACR	45(0)	SD	REJT	ACPT	-	-	-	-	-	
			BACS,OACS	61(0)	SD	REJT	ACPT	-	-	-	-	-	
CREV1S	CMD Processor 1 Receiver Reverse Status	B,O	BSUBA,OSUBA	10(2)	BL	NORM	REV	-	-	-	-	-	
CREV2S	CMD Processor 2 Receiver Reverse Status	B,O	BSUBA,OSUBA	10(3)	BL	NORM	REV	-	-	-	-	-	
CSCL3S	Stored Command Logic 1 Status	B,O	BENG,OENG	14(4)	BL	ON	OFF	-	-	-	-	-	
			BACS,OACS	30(4)	BL	ON	OFF	-	-	-	-	-	
			BSUBA,OSUBA	14(4)	BL	ON	OFF	-	-	-	-	-	
			OLACR	14(4)	BL	ON	OFF	-	-	-	-	-	
CSCL2S	Stored Command Logic 2 Status	B,O	BENG,OENG	46(4)	BL	ON	OFF	-	-	-	-	-	
			BACS,OACS	62(4)	BL	ON	OFF	-	-	-	-	-	
			BSUBA,OSUBA	17(4)	BL	ON	OFF	-	-	-	-	-	
			OLACR	46(4)	BL	ON	OFF	-	-	-	-	-	
CXASWS	Acceleration Switch Status	SP1,2,3	SBLKD	3(2)	BL	ACTV	NACT	-	-	-	-	-	X-SP1,2,&3

APPENDIX B: TELEMETRY REFERENCE DATA
(MNEMONICS IN ALPHABETICAL ORDER)

MNEMONIC	DESCRIPTION	APPL S/C	FORMATS	WORD (BIT)	DATA TYPE	STATUS			E.V. CONVERSION		RANGE	SAMPLE TIME	REMARKS/REF
						1BIT =	"0" BIT =	MULT	UNIT	QUANTIZ			
PRRATE	Bit Rate	B,O	BETY,	3(4-7)	SD	-	8 BPS	= 0000	-	-	-	-	
			BENG,OENG,		SD	-	16 BPS	= 0001	-	-	-	-	
							32 BPS	= 0010	-	-	-	-	
			BACS,OACS,		SD	-	64 BPS	= 0011	-	-	-	-	
			BOMBO,OCMBO,		SD	-	128 BPS	= 0100	-	-	-	-	
			OPERA,OPERB,		SD	-	170 2/3	= 0101	-	-	-	-	
			OPERC,OPERD		SD	-	256	= 0110	-	-	-	-	
			OPERE,OAPOA,		SD	-	341 1/3	= 0111	-	-	-	-	
			OAPOB,OLACR,		SD	-	512	= 1000	-	-	-	-	
			OCMBO,OPBK		SD	-	682 2/3	= 1001	-	-	-	-	
					SD	-	1024	= 1010	-	-	-	-	
					SD	-	2048	= 1011	-	-	-	-	
					SD	-	4096	= 1100,1101,1110,1111			-	-	
		LP	LDESC,LBLKD	3(7)	SD	256	128	-	-	-	-	-	
		SP	SUDES,SBLKD,	3(7)	SD	64	16	-	-	-	-	-	
		SP	SLDES	3(7)	SD	64	16	-	-	-	-	-	
ECLOK	Spacecraft Clock Magnitude	B,O	BOMBO,OCMBO	7	SD	-	-	-	SEC	.125 SEC		-	7(0) = MSB, 2^{23}
1CLOK	Spacecraft Clock Magnitude	B,O	BENG,OENG	50	SD	-	-	-	SEC	.125 SEC		-	50(0) = MSB, 2^{15}
2CLOK	Spacecraft Clock Magnitude	B,O	BENG,OENG	51	SD	-	-	-	SEC	.125 SEC		-	51(0) = MSB, 2^{7}
3CLOK	Spacecraft Clock Magnitude	B,O	BACS,OACS	54	SD	-	-	-	SEC	.125 SEC		-	54(0) = MSB, 2^{7}
4CLOK	Spacecraft Clock Magnitude	B,O	BSUBA,OSUBA	0-2	SD	-	-	-	SEC	.125 SEC		-	0(0) = MSB, 2^{23}
1TCONT	Command Data Unit Temp	SP1	SSUBA,	32	SD	-	-	-	°C			-	
2TCONT	" " " "	SP2	SSUBA	32	SD	-	-	-	°C			-	
3TCONT	" " " "	SP3	SSUBA	32	SD	-	-	-	°C			-	
4TCONT	" " " "	LP	LSUBA	58	SD	-	-	-	°C			-	
BLSTRS	La. Probe Data Storage Status	LP	LDESC,LBLKD	3(0)	BL	STOR	MSTO	-	-	-	-	-	
DPCHIS	PCM Encoder 1 Power Status	B,O	BSUBA,OSUBA	4(6)	BL	INLK	OUTL	-	-	-	-	-	
DPCHPS	PCM Encoder 2 Power Status	B,O	BSUBA,OSUBA	4(7)	BL	INLK	OUTL	-	-	-	-	-	
DPCH1V	PCM Encoder 1 Calibrate	B,O	BSUBA,OSUBA	11	SD	-	-	-	VOLTS	-	0 to 5.1	-	
DPCH2V	PCM Encoder 2 Calibrate	B,O	BSUBA,OSUBA	12	SD	-	-	-	VOLTS	-	0 to 5.1	-	
DRSHIS	Data Storage Unit 1 Power Status	O	OPERA,OPERB,OPERC OPERD,OPERE,APOA	8(2)	BL	ON	OFF	-	-	-	-	-	FORMATS APPLY TO
DRSHPS	Data Storage Unit 2 Power Status	O	VOLT,OENG APOB,OACR,OCMBO	8(3)	BL	ON	OFF	-	-	-	-	-	BOTH DSU STATUS
DRSWCS	Info on p. 9												

APPENDIX B: TELEMETRY REFERENCE DATA
(MNEMONICS IN ALPHABETICAL ORDER)

MNEMONIC	DESCRIPTION	APPL S/C	FORMATS	WORD (BIT)	DATA TYPE	STATUS			E.V. CONVERSION		RANGE	SAMPLE TIME	REMARKS/REF
						1BIT =	"0" BIT =	MULT	UNIT	QUANTIZ			
DTLM1S	TLM PROCESSOR 1 Power Status	B,O	BSUBA,OSUBA	6(0)	BL	ON	OFF	-	-	-	-	-	
DTLM2S	TLM Processor 2 Power Status	B,O	BSUBA,OSUBA	6(1)	BL	ON	OFF	-	-	-	-	-	
DESTRS	SBand Probe(s) Data Storage Status	SP1,2,3	VDES,RLKD,LDES,	3(0)	BL	STOR	MSTD	-	-	-	-	-	
EBIMSS	BIMS Power Status	B	BSUBC	1(0)	BL	ON	OFF	-	-	-	-	-	
EBMRSS	BMRS " "	B	BSUBC	1(1)	BL	ON	OFF	-	-	-	-	-	
ELASS	LAS " "	LP	LSUBA,LSUBB	8(0)	BL	ON	OFF	-	-	-	-	-	
ELCPSS	LCPS " "	LP	LSUBA,LSUBB	8(2)	BL	ON	OFF	-	-	-	-	-	
ELEVTR	Time of HGA Elevation Angle Sample (in HT)	O	N/A	N/A	CALC	ON	OFF	-	HH:MM:SS.T			-	PC-454.04/3.6.4.9.3
ELGCS	LGC Power Status	LP	LSUBA,LSUBB	8(5)	BL	ON	OFF	-	-	-	-	-	
ELIRS	LIR " "	LP	LSUBA,LSUBB	8(3)	BL	ON	OFF	-	-	-	-	-	
ELMRSS	LMRS " "	LP	LSUBA,LSUBB	8(4)	BL	ON	OFF	-	-	-	-	-	
ELRS	LR " "	LP	LSUBA,LSUBB	8(1)	BL	ON	OFF	-	-	-	-	-	
ELSFRS	LSFR " "	LP	LSUBA,LSUBB	8(6)	BL	ON	OFF	-	-	-	-	-	
EQCPPS	QCPP Power Status	O	QSUBD	1(6)	BL	ON	OFF	-	-	-	-	-	
EQEFDS	QEFD " "	O	QSUBD	0(4)	BL	ON	OFF	-	-	-	-	-	
EQETPS	QETP " "	O	QSUBD	1(1)	BL	ON	OFF	-	-	-	-	-	
EQGBGS	QGBG " "	O	QSUBD	0(5)	BL	ON	OFF	-	-	-	-	-	
EQIMSS	QIMS " "	O	QSUBD	1(0)	BL	ON	OFF	-	-	-	-	-	
EQIRSS	QIR Power Status	O	QSUBD	1(7)	BL	ON	OFF	-	-	-	-	-	
EQMAGS	QMAG " "	O	QSUBD	0(2)	BL	ON	OFF	-	-	-	-	-	
EQMRSS	QMRS " "	O	QSUBD	1(5)	BL	ON	OFF	-	-	-	-	-	
EQPASS	QPA " "	O	QSUBD	0(3)	BL	ON	OFF	-	-	-	-	-	
EQRAGS	QRAG " "	O	QSUBD	2(0)	BL	ON	OFF	-	-	-	-	-	
EQRPAS	QRPA Power Status	O	QSUBD	0(0)	BL	ON	OFF	-	-	-	-	-	
EQRYSS	QRYS " "	O	QSUBD	1(4)	BL	ON	OFF	-	-	-	-	-	
ESASS	SAS " "	SP1,2,3	SUBA,SUBB,SUBC	56(0)	BL	ON	OFF	-	-	-	-	-	
ESHS	SH " "	SP1,2,3	SUBA,SUBB,SUBC	56(1)	BL	ON	OFF	-	-	-	-	-	
ESHFRS	SHFR " "	SP1,2,3	SUBA,SUBB,SUBC	56(2)	BL	ON	OFF	-	-	-	-	-	

APPENDIX B: TELEMETRY REFERENCE DATA
(MNEMONICS IN ALPHABETICAL ORDER)

MNEMONIC	DESCRIPTION	APPL S/C	FORMATS	WORD (BIT)	DATA TYPE	STATUS			E.V. CONVERSION		RANGE	SAMPLE TIME	REMARKS/REF
						IBIT -	'C' BIT -	MULT	UNIT	QUANTIZ			
FORMAT	Spacecraft Telemetry Format ID	O	OENG	3 (0-3)	SD	ORBITER ENGRG.		0000	-	-	-	-	
		B	BENG	3 (0-3)	SD	BUS ENGRG.		0001	-	-	-	-	
		B	BETY	3 (0-3)	SD	BUS ENTRY		0010	-	-	-	-	
		O	PROGRAMMABLE	3 (0-3)	SD			0011	-	-	-	-	
		O,B	OACS,BACS	3 (0-3)	SD	ACS		0100	-	-	-	-	
		O,B	OCMRO,BCMRO	3 (0-3)	SD	CMD MEM READOUT		0101	-	-	-	-	
		O,B	OPBK,BPBK	3 (0-3)	SD	PLAYBACK		0110	-	-	-	-	
		O	OMRO,BMRO	3 (0-3)	SD	DATA MEM. READOUT		0111	-	-	-	-	
		O	OLACR	3 (0-3)	SD	LAUNCH/CRUISE		1000	-	-	-	-	
		O	OPERA	3 (0-3)	SD	PERIAPSIS A		1001	-	-	-	-	
		O	OPERB	3 (0-3)	SD	PERIAPSIS B		1010	-	-	-	-	
		O	OPERC	3 (0-3)	SD	PERIAPSIS C		1011	-	-	-	-	
		O	OPERD	3 (0-3)	SD	PERIAPSIS D		1100	-	-	-	-	
		O	OPERE	3 (0-3)	SD	PERIAPSIS E		1101	-	-	-	-	
		O	OAPOA	3 (0-3)	SD	APOAPSIS A		1110	-	-	-	-	
		O	OAPOB	3 (0-3)	SD	APOAPSIS B		1111	-	-	-	-	
		LP	LDESC	3 (5-6)	SD	DESCENT		00	-	-	-	-	
		LP	LBLKO	3 (5-6)	SD	BLACKOUT		11	-	-	-	-	
		SP	SUDES	3 (5-6)	SD	UPPER DESCENT		00	-	-	-	-	
		SP	SBLKO	3 (5-6)	SD	LOWER DESCENT		11	-	-	-	-	
		SP	SLDES	3 (5-6)	SD				-	-	-	-	
OOSWES	OSU Control Status	O	OPERA,OPERB,OPERC	8 (5-7)	SD	STBY(STANDBY)		000	-	-	-	-	
			OPERD,OPERE,OAPOA	8 (5-7)	SD	STBY(STANDBY)		001	-	-	-	-	
			OAPOB,OLACR,OOMRO	8 (5-7)	SD	READ1(READMEM1)		010	-	-	-	-	
			OPBK,OENG	8 (5-7)	SD	READ2(READMEM2)		011	-	-	-	-	
				8 (5-7)	SD	STORE1(STOREMEM1)		100	-	-	-	-	
				8 (5-7)	SD	STORE2(STOREMEM2)		101	-	-	-	-	
				8 (5-7)	SD	STBY-R(STANDBY-RESET)		110	-	-	-	-	
				8 (5-7)	SD	STBY (STANDBY)		111	-	-	-	-	

APPENDIX B: TELEMETRY REFERENCE DATA
(MNEMONICS IN ALPHABETICAL ORDER)

MNEMONIC	DESCRIPTION	APPL S/C	FORMATS	WORD (BIT)	DATA TYPE	STATUS			E.V. CONVERSION		RANGE	SAMPLE TIME	REMARKS/REF
						1BIT =	"0" BIT =	MULT.	UNIT	QUANTIZ.			
BASUTR	Time of S/C Transmission of 1st Bit of Major Frame	B,O	N/A	N/A	CALC	-	-	-	HH:MM:SS.T		-	-	SCID = O, IN UT
BIPUTR	Time of S/C Transmission of 1st Bit of Master Index Pulse	B,O	N/A	N/A	CALC	-	-	-	HH:MM:SS.T		-	-	IN UT
BISUTR	Time of S/C Transmission of 1st Bit of Minor Frame	B,O	N/A	N/A	CALC	-	-	-	HH:MM:SS.T		-	-	IN UT
F1BATT	Small Probe 1 Battery Temp	B	BSUBC	28	SD	-	-	-	°C			-	
F2BATT	" " 2 " "	B	BSUBC,BSUBA	44,26	SD	-	-	-	°C			-	
F3BATT	" " 3 " "	B	BSUBC	60	SD	-	-	-	°C			-	
P1BUSI	" " 1 Bus Current	SP	SSUBA,SSUBB,SSUBC	7,23,39,55	SD	-	-	-	AMPS	-	0.0 to 4.0	-	
P2BUSI	" " 2 " "	SP	SSUBA,B,AC	7,23,39,55	SD	-	-	-	AMPS	-	0.0 to 4.0	-	
P3BUSI	Small Probe 3 Bus Current	SP	SSUBA,B,AC	7,23,39,55	SD	-	-	-	AMPS	-	0.0 to 4.0	-	
P1BUSV	" " 1 " Voltage	SP	SSUBA,B,AC	6,22,38,54	SD	-	-	-	VOLTS		0.0 to 33.0	-	
P2BUSV	" " 2 " "	SP	SSUBA,B,AC	6,22,38,54	SD	-	-	-	VOLTS		0.0 to 33.0	-	
P3BUSV	" " 3 " "	SP	SSUBA,B,AC	6,22,38,54	SD	-	-	-	VOLTS		0.0 to 33.0	-	
P1CD1S	" " ; CDU and RF Power Relay 1 Status	SP	SSUBA,B,AC	8(2)	BL	ON	OFF	-	-	-	-	-	
P1CD2S	Small Probe 1 CDU and RF Power Relay 2 Status	SP	SSUBA,B,AC	8(3)	BL	ON	OFF	-	-		-	-	
P2CD1S	" " 2 " " " " 1 "	SP	SSUBA,B,AC	8(2)	BL	ON	OFF	-	-	-	-	-	
P2CD2S	" " 2 " " " " 2 "	SP	SSUBA,B,AC	8(3)	BL	ON	OFF	-	-	-	-	-	
P3CD1S	" " 3 " " " " 1 "	SP	SSUBA,B,AC	8(2)	BL	ON	OFF	-	-	-	-	-	
P3CD2S	" " 3 " " " " 2 "	SP	SSUBA,B,AC	8(3)	BL	ON	OFF	-	-	-	-	-	
P1HTRS	Small Probe 1 Battery Htr. Relay Status	B	BSUBC	1(4)	BL	ON	OFF	-	-	-	-	-	
P2HTRS	" " 2 " " " "	B	BSUBC	1(5)	BL	ON	OFF	-	-	-	-	-	
P3HTRS	" " 3 " " " "	B	BSUBC	1(6)	BL	ON	OFF	-	-	-	-	-	
P1MTRS	Small Probe 1 Heater Power Status	B	BENG,BSUBC	8(4) 9(4)	BL	ON	OFF	-	-	-	-	-	
P2MTRS	" " 2 " " "	B	BENG,BSUBC	8(5) 9(5)	BL	ON	OFF	-	-	-	-	-	
P3MTRS	" " 3 " " "	B	BENG,BSUBC	8(6) 9(6)	BL	ON	OFF	-	-	-	-	-	
P1WTMS	Small Probe 1 SA Window Htr. Power Status	SP	SSUBA,SUBB,SUBC	8(4)	BL	ON	OFF	-	-	-	-	-	
P2WTMS	" " 2 " " " " "	SP	SSUBA,SUBB,SUBC	8(4)	BL	ON	OFF	-	-	-	-	-	
P3WTMS	" " 3 " " " " "	SP	SSUBA,SUBB,SUBC	8(4)	BL	ON	OFF	-	-	-	-	-	

APPENDIX B: TELEMETRY REFERENCE DATA
(MNEMONICS IN ALPHABETICAL ORDER)

MNEMONIC	DESCRIPTION	APPL S/C	FORMATS	WORD (BIT)	DATA TYPE	STATUS			E.V. CONVERSION		RANGE	SAMPLE TIME	REMARKS/REF
						"1" BIT =	"0" BIT =	MULT	UNIT	QUANTIZ			
P1PR1S	Small Probe 1 Interval Power Relay 1 Status	B	BSUBC	2(0)	BL	ON	OFF	-	-	-	-	-	
P1PR2S	" " 1 " " " 2 "	B	BSUBC	2(1)	BL	ON	OFF	-	-	-	-	-	
P2PR1S	" " 2 " " " 1 "	B	BSUBC	2(2)	BL	ON	OFF	-	-	-	-	-	
P2PR2S	" " 2 " " " 2 "	B	BSUBC	2(3)	BL	ON	OFF	-	-	-	-	-	
P3PR1S	" " 3 " " " 1 "	B	BSUBC	2(4)	BL	ON	OFF	-	-	-	-	-	
P3PR2S	" " 3 " " " 2 "	B	BSUBC	2(5)	BL	ON	OFF	-	-	-	-	-	
P1SP1S	Small Probe 1 Science Power Relay 1 Status	SP	SSUBA, B, & C	B(5)	BL	ON	OFF	-	-	-	-	-	
P1SP2S	" " 1 " " " 2 "	SP	SSUBA, B, & C	B(6)	BL	ON	OFF	-	-	-	-	-	
P2SP1S	" " 2 " " " 1 "	SP	SSUBA, B, & C	B(5)	BL	ON	OFF	-	-	-	-	-	
P2SP2S	" " 2 " " " 2 "	SP	SSUBA, B, & C	B(6)	BL	ON	OFF	-	-	-	-	-	
P3SP1S	" " 3 " " " 1 "	SP	SSUBA, B, & C	B(5)	BL	ON	OFF	-	-	-	-	-	
P3SP2S	" " 3 " " " 2 "	SP	SSUBA, B, & C	B(6)	BL	ON	OFF	-	-	-	-	-	
P1T1I	Battery 1, Pack 1 Temp	B	BENG,BSUBA	56 55	AC	-	-	-	°C	-	30.0 to 0.0	-	
P1T2I	" 2 " 1 "	B	BENG,BSUBA	57 57	AC	-	-	-	°C	-	"	-	
P1T1T	" 1 " 2 "	B	BENG,BSUBA	58 56	AC	-	-	-	°C	-	"	-	
P1T2T	" 2 " 2 "	B	BENG,BSUBA	59 58	AC	-	-	-	°C	-	"	-	
P1R1S	Battery 1 Relay 1 Status	B	BSUBA	4(0)	BL	HI	LO	-	-	-	-	-	
P1R2S	" 1 " 2 "	B	BSUBA	4(1)	BL	HI	LO	-	-	-	-	-	
P2R1S	" 2 " 1 "	B	BSUBA	4(3)	BL	HI	LO	-	-	-	-	-	
P2R2S	" 2 " 2 "	B	BSUBA	4(4)	BL	HI	LO	-	-	-	-	-	
P1B1V	Battery 1 Terminal Voltage	B,O	BENG,OLACR	22	AN	-	-	-	VOLTS	-	0.0 to 40.0	-	
		O	OENG,	22	AN	-	-	-	VOLTS	-	0.0 to 40.0	-	
		B,O	BSUBA,OSUBA	49	AN	-	-	-	VOLTS	-	"	-	
P2B1V	Battery 2 Terminal Voltage	B,O	BENG,OLACR	54	AN	-	-	-	VOLTS	-	"	-	
		O	OENG	54	AN	-	-	-	VOLTS	-	"	-	
		B,O	BSUBA,OSUBA	52	AN	-	-	-	VOLTS	-	"	-	
P3BUS1	Spacecraft Loads Current	B	BETY	43	AN	-	-	-	AMPS	-	0.0 to 18.0	-	
		B,O	BENG,OENG,OLACR	18	AN	-	-	-	AMPS	-	"	-	
		B,O	DACS,ORCS	23&55	AN	-	-	-	AMPS	-	"	-	
		B,O	BSUBA,OSUBA	61	AN	-	-	-	AMPS	-	"	-	

APPENDIX B: TELEMETRY REFERENCE DATA
(MNEMONICS IN ALPHABETICAL ORDER)

MNEMONIC	DESCRIPTION	APPL S/C	FORMATS	WORD (BIT)	DATA TYPE	STATUS			E.V. CONVERSION		RANGE	SAMPLE TIME	REMARKS/REF
						1BIT =	"0" BIT =	MULT	UNIT	QUANTIZ			
PCHEKS	Probe Checkout Power Status	B	BSUBC	3	BL	ON	OFF	-	-	-	-	-	
PCHG1I	Battery 1 Charge Current	B,0	BENG,OLACR,OENG	20	AN	-	-	-	AMPS	-	0.0 to 1.0	-	
		B,0	BSUBA,OSUBA	47	AN	-	-	-	AMPS	-	0.0 to 1.0	-	
PCHG2I	Battery 2 Charge Current	B,0	BENG,OLACR,OENG	52	AN	-	-	-	AMPS	-	0.0 to 1.0	-	
		B,0	BSUBA,OSUBA	50	AN	-	-	-	AMPS	-	0.0 to 1.0	-	
PDIS1I	Battery 1 Discharge Current	B,0	BENG,OLACR,OENG	21	AN	-	-	-	AMPS	-	0.0 to 12.0	-	
		B,0	BSUBA,OSUBA	48	AN	-	-	-	AMPS	-	0.0 to 12.0	-	
PDIS2I	Battery 2 Discharge Current	B,0	BENG,OLACR,OENG	53	AN	-	-	-	AMPS		0.0 to 12.0	-	
		B,0	BSUBA,OSUBA	51	AN	-	-	-	AMPS		0.0 to 12.0	-	
PEBUSV	Essential Bus Voltage	B	BETY	42	AN	-	-	-	VOLTS	-	0.0 to 32.0	-	
		B,0	BENG,OLACR,OENG	16 & 48	AN	-	-	-	VOLTS	-	"	-	
		B,0	BACS,OACS	22	AN	-	-	-	VOLTS	-	"	-	
		B,0	BSUBA,CSUBA	11 & 63	AN	-	-	-	VOLTS	-	"	-	
PHT1MS	OIM Heater 1 Power Status	0	OSUBA	6(6)	BL	ON	OFF	-	-	-	-	-	
PHT2MS	OIM Heater 2 Power Status	0	OSUBA	6(7)	BL	ON	OFF	-	-	-	-	-	
PHTAJS	AFT Axial Jet Heaters Power Status	B	BSUBA	6(4)	BL	ON	OFF	-	-	-	-	-	
PHTFJS	Forward Axial Jet Heaters Power Status	B	BSUBA	6(5)	BL	ON	OFF	-	-	-	-	-	
PHTJ1S	Forward Jets 1 & 2 Heater Status	0	OSUBA	6(4)	BL	ON	OFF	-	-	-	-	-	
PHTJ3S	AFT Jets 3 & 4 Heater Status	0	OSUBA	6(5)	BL	ON	OFF	-	-	-	-	-	
PHTTKS	Propellant Tank Heaters Select Status	B,0	BENG,OENG	9(6)	BL	PRIM	SECD	-	-			-	
		B,0	BSUBA,OSUBA	5(6)	BL	PRIM	SECD	-	-		-	-	
PLBATT	Large Probe Battery Temp	B	BSUBC	44	AC	-	-	-	°C			-	
		B	BSUBC	12	AC	-	-	-	°C			-	
PLBUSI	Large Probe Bus Current	LP	LSUBA,LSUBB	7, 23	AN	-	-	-	AMPS	-	0.0 to 15.0	-	
		LP	LSUBA,LSUBB	39, 55	AN	-	-	-	AMPS	-	0.0 to 15.0	-	
PLBUSV	Large Probe Bus Voltage	LP	LSUBA,LSUBB	6, 22	AN	-	-	-	VOLTS	-	0.0 to 33.0	-	
		LP	LSUBA,LSUBB	38, 54	AN	-	-	-	VOLTS	-	0.0 to 33.0	-	
PLCB1S	Lg. Pr. CDU Power Relay 1 Status	LP	LSUBA,LSUBB	0(2)	BL	ON	OFF	-	-	-	-	-	
PLCB2S	Lg. Pr. CDU Power Relay 2 Status	LP	LSUBA,LSUBB	0(3)	BL	ON	OFF	-	-	-	-	-	
PLBRKS	Switched Loads Bus Protection Relay Status	B,0	BSUBA,OSUBA	8(7)	BL	ON	OFF	-	-	-	-	-	

APPENDIX B: TELEMETRY REFERENCE DATA
(MNEMONICS IN ALPHABETICAL ORDER)

MNEMONIC	DESCRIPTION	APPL S/C	FORMATS	WORD (BIT)	DATA TYPE	STATUS			E.V. CONVERSION		RANGE	SAMPLE TIME	REMARKS/REF
						1BIT -	"0" BIT -	MULT	UNIT	QUANTIZ			
PLHT1S	LIR Window Heater 1st Stage Status	LP	LSUBA,LSUBB	0(4)	BL	ON	OFF	-	-	-	-	-	
PLHT2S	LIR Window Heater 2nd Stage Status	LP	LSUBA,LSUBB	9(5)	BL	ON	OFF	-	-	-	-	-	
PLHTBS	Large Probe Battery Heater Relay Status	B	BSUBC	1(7)	BL	ON	OFF	-	-	-	-	-	
PLHTRS	Large Probe Heater Power Status	B	BENG	8(7)	BL	ON	OFF	-	-	-	-	-	
		B	BSUBC	9(7)	BL	ON	OFF	-	-	-	-	-	
PLHTWS	Large Probe Window and Prism Heaters Power Status	LP	LSUBA,LSUBB	16(6)	BL	ON	OFF	-	-	-	-	-	
PLIMTI	Bus Voltage Limiter Current	B,O	BENG,OLACR,OENG	19	AN	-	-	-	AMPS	-	0.0 to 12.0	-	
		B	BSUBA,OSUBA	60	AN	-	-	-	AMPS	-	0.0 to 12.0	-	
PLIM1S	Bus Limiter 1 Enable Status	B,O	BSUBA,OSUBA	8(9)	BL	ENAB	DSAB	-	-	-	-	-	
PLIM2S	" " 2 " "	B,O	BSUBA,OSUBA	8(1)	BL	ENAB	DSAB	-	-	-	-	-	
PLIM3S	" " 3 " "	B,O	BSUBA,OSUBA	8(2)	BL	ENAB	DSAB	-	-	-	-	-	
PLIM4S	" " 4 " "	B,O	BSUBA,OSUBA	8(3)	BL	ENAB	DSAB	-	-	-	-	-	
PLIM5S	" " 5 " "	B,O	BSUBA,OSUBA	8(4)	BL	ENAB	DSAB	-	-	-	-	-	
PLIM6S	" " 6 " "	O	OSUBA	9(5)	BL	ENAB	DSAB	-	-	-	-	-	
PLIM7S	" " 7 " "	O	OSUBA	8(6)	BL	ENAB	DSAB	-	-	-	-	-	
PLPR1S	Large Probe Internal Power Relay 1 Status	B	BSUBC	2(6)	BL	ON	OFF	-	-	-	-	-	
PLPR2S	Large Probe Internal Power Relay 2 Status	B	BSUBC	2(7)	BL	ON	OFF	-	-	-	-	-	
PLRCVS	Large Probe Receiver Power Status	LP	LSUBA,LSUBB	16(7)	BL	ON	OFF	-	-	-	-	-	
PLRFRS	Large Probe RF Power Relay Status	LP	LSUBA,LSUBB	16(3)	BL	ON	OFF	-	-	-	-	-	
PLSP1S	Large Probe Science Power Relay 1 Status	LP	LSUBA,LSUBB	16(4)	BL	ON	OFF	-	-	-	-	-	
PLSP2S	Large Probe Science Power Relay 2 Status	LP	LSUBA,LSUBB	16(5)	BL	ON	OFF	-	-	-	-	-	
PPAB1T	Solar Panel 1 Temp	B	BETT.	45	AC	-	-	-	°C	-	-85 to +100	-	
		B,O	BSUBA,OSUBA	53	AC	-	-	-	°C	-	"	-	
PPAB2T	Solar Panel 2 Temp	B,O	BSUBA,OSUBA	54	AC	-	-	-	°C	-	"	-	
		O	OENG	58	AC	-	-	-	°C	-	"	-	
PPABLI	Solar Panel Current	B,O	BENG,OLACR,OENG	17	AN	-	-	-	AMPS	-	0.0 to 18.0	-	
		B,O	BACS,OACS	43	AN	-	-	-	AMPS	-	"	-	
		B,O	BSUBA,OSUBA	62	AN	-	-	-	AMPS	-	"	-	
PRCHGS	Precharge Status	B,O	BSUBA,OSUBA	10(4)	BL	ON	OFF	-	-	-	-	-	

APPENDIX B: TELEMETRY REFERENCE DATA
(MNEMONICS IN ALPHABETICAL ORDER)

MNEMONIC	DESCRIPTION	APPL S/C	FORMATS	WORD (BIT)	DATA TYPE	STATUS			E.V. CONVERSION		RANGE	SAMPLE TIME	REMARKS/REF
						1BIT =	"0" BIT =	MULT	UNIT	QUANTIZ			
PREG1S	Discharge Regulator 1 Select Status	B,O	BSUBA,OSUBA	4(2)	BL	PRI	RED	-	-	-	-	-	
PREG2S	Discharge Regulator 2 Select Status	B,O	BSUBA,OSUBA	4(5)	BL	PRI	RED	-	-	-	-	-	
PRFBUS	RF Restr. Bus Reset Relay Status	B,O	BSUBA,OSUBA	7(3)	BL	ON	OFF	-	-	-	-	-	
PSCBUS	Science Bus Reset Relay Status	B,O	BSUBA,OSUBA	7(2)	BL	ON	OFF	-	-	-	-	-	
PSCICI	Science Current	B,O	BENG,OLACR,OENG	49	AR	-	-	-	AMPS	-	0.0 to 4.7	-	
		B	SSUBA,OSUBA	53	AR	-	-	-	AMPS	-	0.0 to 4.7	-	
PSPBUS	Power System Protection Status	B,O	BENG,OENG	9(7)	BL	ON	OFF	-	-	-	-	-	
		B,O	BSUBA,OSUBA	5(7) 9(0)	BL	ON	OFF	-	-	-	-	-	
PVBIBD	Spacecraft ID	B,O	BCTT,OPERA	4(0-1)	SD		BUS +	00	-	-	-	-	
			BENG,OPERB	4(0-1)	SD		ORBITER +	01	-	-	-	-	
			BACS,OPERC	4(0-1)	SD				-	-	-	-	
			BCMRG,OPERD	4(0-1)	SD				-	-	-	-	
			PROG OPERE	4(0-1)	SD				-	-	-	-	
			SAPSA	4(0-1)	SD				-	-	-	-	
			OAPSB	4(0-1)	SD				-	-	-	-	
			OLACR	4(0-1)	SD				-	-	-	-	
			OLMRD	4(0-1)	SD				-	-	-	-	
			OCMRD	4(0-1)	SD				-	-	-	-	
			OFSK	4(0-1)	SD				-	-	-	-	
			OENG	4(0-1)	SD				-	-	-	-	
			OACS	4(0-1)	SD				-	-	-	-	
PVBIBD	Spacecraft ID	PROBES	LDESC,SUDES	3(3-4)	SD	SMALL	PROBE 1	= 00	-	-	-	-	
			LBLRD,SBLKD	3(3-4)	SD	SMALL	PROBE 2	= 01	-	-	-	-	
			SLDES	3(3-4)	SD	SMALL	PROBE 3	= 10	-	-	-	-	
					SD	LARGE	PROBE	= 11	-	-	-	-	
PVSCID	Subcommutator ID (Minor Frame Count)	B,O	Same as B,O, above	4(2-7)	SD	-	-	-	COUNTS	BIT 7 = LSB	0 - 63	-	Binary Counter
	"	PROBES	Same as above	4(4@7)	SD	-	-	-	COUNTS	BIT 7 = LSB	0 - 15	-	Binary Counter
RSNTON	Stable Oscillator Heater Voltage (SR. Probe 1)	PROBES	SSUBA,SSUBB,SSUBC	15,31,47		-	-	-	VOLTS	-	0.0 to +5.1	-	
RSNTON	" " " " (SR. Probe 2)			63		-	-	-	VOLTS	-	0.0 to +5.1	-	
RSNTON	" " " " (SR. Probe 3)					-	-	-	VOLTS	-		-	

APPENDIX B: TELEMETRY REFERENCE DATA
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MNEMONIC	DESCRIPTION	APPL SIC	FORMATS	WORD (BIT)	DATA TYPE	STATUS			E.V. CONVERSION		RANGE	SAMPLE TIME	REMARKS/REF
						1BIT -	"0" BIT -	MULT	UNIT	QUANTIZ			
R1OSC5	Stable Oscillator Power Status, Sp. Probe 1	SP	SUBA, B, & C	8(7)	BL	ON	OFF	-	-	-	-	-	
R2OSC5	" " " " Sp. Probe 2	SP	SUBA, B, & C	8(7)	BL	ON	OFF	-	-	-	-	-	
R3OSC5	" " " " Sp. Probe 3	SP	SUBA, B, & C	8(7)	BL	ON	OFF	-	-	-	-	-	
RAGC1V	Receiver 1 AGC Voltage	B,D	BETY,DLACB	29	AR	-	-	-	dBm	-	-152 to -70	-	
		B,D	BENG,DENG	29	AR	-	-	-	"	-	"	-	
		B,D	BACS,DACS	7	AR	-	-	-	"	-	"	-	
		B,D	SUBA,DSUBA	23	AR	-	-	-	"	-	"	-	
RAGC2V	Receiver 2 AGC Voltage	B,D	BETH,DLACB	30	AR	-	-	-	"	-	"	-	
		B,D	BENG,DENG	30	AR	-	-	-	"	-	"	-	
		B,D	BACS,DACS	35	AR	-	-	-	"	-	"	-	
		B,D	BSUBA,DSUBA	24	AR	-	-	-	"	-	"	-	
RAMP1S	Amplifier 1/2 Power Status	B,D	BSUBA,DSUBA	9(3)	BL	A1LA2H	A1LA1H	-	-	-	-	-	
RAMP2S	Amplifier 3/4 Switch Position Status	B,D	BSUBA,DSUBA	9(4)	BL	3	4	-	-	-	-	-	
RAMP1T	RF Power Amplifier 1 Temp	B,D	BSUBA,DSUBA	27	AC	-	-	-	°C	-	-185.0 to +100.0	-	
RAMP2T	" " " 2 "	B,D	BSUBA,DSUBA	28	AC	-	-	-	°C	-	"	-	
RAMP3T	" " " 3 "	B,D	BSUBA,DSUBA	29	AC	-	-	-	°C	-	"	-	
RAMP4T	" " " 4 "	B,D	BSUBA,DSUBA	30	AC	-	-	-	°C	-	"	-	
RAMP1V	RF Power Amplifier 1 Output	B,D	BENG,DLACB,DENG	24	AR	-	-	-	WATTS	-	0.0 to 11.5	-	
		B,D	BSUBA,DSUBA	33	AR	-	-	-	"	-	"	-	
RAMP2V	" " " 2 "	B,D	BENG,DLACB,DENG	25	AR	-	-	-	"	-	0.0 to 11.8	-	
		B,D	BSUBA,DSUBA	34	AR	-	-	-	"	-	"	-	
RAMP3V	" " " 3 "	B,D	BENG,DLACB,DENG	26	AR	-	-	-	"	-	0.0 to 10.4	-	
		B,D	BSUBA,DSUBA	35	AR	-	-	-	"	-	"	-	
RAMP4V	" " 4 "	B,D	BENG,DLACB,DENG	27	AR	-	-	-	"	-	0.0 to 12.0	-	
		B,D	BSUBA,DSUBA	36	AR	-	-	-	"	-	"	-	
RANT1S	Fwd. Horn/Aft OMNI Power Switch Status	B	BSUBA	9(6)	BL	LFHA	LAHF	-	-	-	-	-	
RANT1S	Aft OMNI/BAPTA Power Switch Status	D	DSUBA	9(6)	BL	HALA	LAHB	-	-	-	-	-	
RANT2S	Horn/Fwd. OMNI Switch Position Status	B	BSUBA	9(5)	BL	OMNI	HORN	-	-	-	-	-	
RANT2S	Hi Gain Dish/Lo Gain Ant. Switch Position Status	D	DSUBA	9(7)	BL	HGDH	LGDM	-	-	-	-	-	
RANT3S	Hi Gain Backup/Fwd. OMNI Switch Position Status	D	DSUBA	9(5)	BL	HGHU	FWDM	-	-	-	-	-	

APPENDIX B: TELEMETRY REFERENCE DATA
(MNEMONICS IN ALPHABETICAL ORDER)

MNEMONIC	DESCRIPTION	APPL. S/C	FORMATS	WORD (BIT)	DATA TYPE	STATUS			E.V. CONVERSION		RANGE	SAMPLE TIME	REMARKS/REF
						IBIT =	"0" BIT =	MULT	UNIT	QUANTIZ			
RCON1S	Exciter 1 Coherent Mode Status	B,O	BENG,OENG	9(2)	BL	INHB	REST	-	-	-	-	-	
		B,O	BSUBA,OSUBA	5(2)	BL	INHB	REST	-	-	-	-	-	
RCON2S	Exciter 2 Coherent Mode Status	B,O	BENG,OENG	9(3)	BL	INHB	REST	-	-	-	-	-	
		B,O	BSUBA,OSUBA	5(3)	BL	INHB	REST	-	-	-	-	-	
RLAGCV	Receiver AGC Voltage	LP	LSUBA,LSUBB	14,30,	AN	-	-	-	dBm	-	-154 to -70	-	
				46,62	AN	-	-	-	dBm	-	-154 to -70	-	
RLOK1S	Receiver 1 Phase Lock Status	B,O	BENG,OENG	9(0)	BL	INLK	OUTL	-	-	-		-	
		B,O	BSUBA,OSUBA	5(0)	BL	INLK	OUTL	-	-	-	-	-	
RLOK2S	Receiver 2 Phase Lock Status	B,O	BENG,OENG	9(1)	BL	INLK	OUTL	-	-	-	-	-	
		B,O	BSUBA,OSUBA	5(1)	BL	INLK	OUTL	-	-	-	-	-	
RLOSCT	Transmitter Aux. Oscillator Temp	LP	LSUBA,LSUBB	29	AC	-	-	-	°C	-	-30.0 to +80.0	-	
RLSPEV	Receiver Static Phase Error	LP	LSUBA,LSUBB	13	AN	-	-	-	KHZ	-	±200.0	-	
RLVCOT	Receiver VCO Temp	LP	LSUBA,LSUBB	21	AC	-	-	-	°C	-	-30.0 to +80.0	-	
ROSC1T	Xmtr. Aux. Oscillator 1 Temp	B,O	BENG,OENG	40	AC	-	-	-	°C	-	-30.0 to +80.0	-	
		B,O	BSUBA,OSUBA	21	AC	-	-	-	°C	-	-30.0 to +80.0	-	
ROSC2T	Xmtr. Aux. Oscillator 2 Temp	B,O	BENG,OENG	41	AC	-	-	-	°C	-	-30.0 to +80.0	-	
		B,O	BSUBA,OSUBB	22	AC	-	-	-	°C	-	-30.0 to +80.0	-	
RRCVRS	Receiver Switch Position Status	B,O	BSUBA,OSUBA	9(1)	BL	F2A1	F1A2	-	-	-	-	-	
RREV1C	Receiver 1 Reverse Timer	B,O	BENG,OENG,OLACR	14(6-7)	SD	(WHEN BIT 6=1=13.2 HRS)			(WHEN BIT 7=1=9.1 HRS)			-	MSB BITS
		B,O	BACS,OACS	30(6-7)	SD		"			"		-	" "
		B,O	BSVBA,OSUBA	14(6-7)	SD		"			"		-	" "
RREV2C	Receiver 2 Reverse Timer	B,O	BENG,OENG,OLACR	46(6-7)	SD		"			"		-	" "
		B,O	BACS,OACS	62(6-7)	SD		"			"		-	" "
		B,O	BSUBA,OSUBA	17(6-7)	SD		"			"		-	" "
RSPE1V	Receiver 1 Static Phase Error Voltage	B,O	BENG,OENG,OLACR	37	AN	-	-	-	KHZ	-	±200.0	-	
		B,O	BSUBA,OSUBA	25	AN	-	-	-	KHZ	-	±200.0	-	
RSPE2V	Receiver 2 Static Phase Error Voltage	B,O	BENG,OENG,OLACR	38	AN	-	-	-	KHZ	-	±200.0	-	
		B,O	BSUBA,OSUBA	26	AN	-	-	-	KHZ	-	±200.0	-	
XVCO1S	Exciter 1 Transfer Select Status	B,O	BSUBA,OSUBA	6(2)	BL	VCO	RCVR	-	-	-	-	-	
XVCO2S	Exciter 2 Transfer Select Status	B,O	BSUBA,OSUBA	6(3)	BL	VCO	RCVR	-	-	-	-	-	

APPENDIX B: TELEMETRY REFERENCE DATA
(MNEMONICS IN ALPHABETICAL ORDER)

MNEMONIC	DESCRIPTION	APPL S/C	FORMATS	WORD (BIT)	DATA TYPE	STATUS			E.V. CONVERSION		RANGE	SAMPLE TIME	REMARKS/REF
						1BIT =	"0" BIT =	MULT	UNIT	QUANTIZ			
RVCO1T	Receiver 1 VCO Temp	B,O	BENG,OENG	31	AC	-	-	-	°C	-	-30 to +80	-	
		B,O	BSUBA,OSUBA	19	AC	-	-	-	°C	-	-30 to +80	-	
RVCO2T	Receiver 2 VCO Temp	B,O	BENG,OENG	39	AC	-	-	-	°C	-	-30 to +80	-	
		B,O	BSUBA,OSUBA	20	AC	-	-	-	°C	-	-30 to +80	-	
RXCT1S	Exciter 1 Power Status	B,O	BENG,OENG	9(4)	BL	ON	OFF	-	-	-	-	-	
		B,O	BSUBA,OSUBA	5(4)	BL	ON	OFF	-	-	-	-	-	
RXCT2S	Exciter 2 Power Status	B,O	BENG,OENG	9(5)	BL	ON	OFF	-	-	-	-	-	
		B,O	BSUBA,OSUBA	5(5)	BL	ON	OFF	-	-	-	-	-	
RXCTRS	Exciter Switch Position Status	B,O	BSUBA,OSUBA	9(2)	BL	1	2	-	-	-	-	-	
RXMTRT	X-Band Transmitter Temp	O	OENG,	36		-	-	-	°C	-	3.0 to 50.0	-	
		O	OSUBA	32		-	-	-	°C	-	3.0 to 50.0	-	
RXMTRW	X-Band Transmitter RF Output	O	OENG	28		-	-	-	WATTS		0.0 to 12.5	-	
		O	OSUBA	37		-	-	-	WATTS		0.0 to 12.5	-	
S1AF1T	Small Probe 1 Aft Shelf Temp	B	BENG	61	AC	-	-	-	°C			-	From Sus
		B	BSUBA	41	AC	-	-	-	°C			-	
		B	BSUBC	29	AC	-	-	-	°C			-	
S1AF2T	Small Probe 1 Aft Shelf Temp	SP	SSUBA, B & C	13,45	AC	-	-	-	°C			-	From Probes
		SP	SSUBC	24	AC	-	-	-	°C			-	
S1ASBS	Small Probe 1 SAS Boom Deploy Status	SP	SSUBA, B, & C	8(0)	BL	DEPL	STOW	-	-	-	-	-	
S1FRBS	Small Probe 1 SRF9 Boom Deploy Status	SP	SSUBA, B, & C	8(1)	BL	DEPL	STOW	-	-	-	-	-	
S1FW1T	Small Probe 1 Forward Shelf Temp	B	BSUBA	44	AC	-	-	-	°C			-	
		B	BSUBC	30	AC	-	-	-	°C			-	
S1FW2T	Small Probe 1 Forward Shelf Temp	SP	SSUBA	29,61	AC	-	-	-	°C			-	
		SP	SSUBB	21,53	AC	-	-	-	°C			-	
		SP	SSUBC	29,40,61	AC	-	-	-	°C			-	
S1INTP	Small Probe 1 Internal Pressure	SP	SSUBA,SSUBB	14,30,	AM	-	-	-	PSI			-	
		SP	SSUBA,SSUBB	46,61	AM	-	-	-	PSI			-	
		SP	SSUBC	2,14,18	AM	-	-	-	PSI			-	
		SP	SSUBC	30,34,46	AM	-	-	-	PSI			-	
		SP	SSUBC	50,62	AM	-	-	-	PSI			-	

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APPENDIX B: TELEMETRY REFERENCE DATA
(MNEMONICS IN ALPHABETICAL ORDER)

MNEMONIC	DESCRIPTION	APPL S/C	FORMATS	WORD (BIT)	DATA TYPE	STATUS 1BIT	STATUS '0' BIT	STATUS MULT	E.U. CONVERSION UNIT	E.U. CONVERSION QUANTIZ	RANGE	SAMPLE TIME	REMARKS/REF
S1RELS	Small Probe 1 Release Status	B	BLRG	8(0)	BL	STOW	RELS	-	-	-	-	-	
		B	BSUBC	0(0)	BL	STOW	RELS	-	-	-	-	-	
S2AFIT	Small Probe 2 Aft Shelf Temp	B	BSUBB	24	AC	-	-	-	°C			-	FROM BUS
		B	BSUBC	45	AC	-	-	-	°C			-	
S2AF2T	Small Probe 2 Aft Shelf Temp	SP	SSUBA, B, & C	13,45	AC	-	-	-	°C			-	FROM PROBE
		SP	SSUBC	24	AC	-	-	-	°C			-	
S2ASBS	Small Probe 2 SAS Boom Deploy Status	SP	SSUBA, B, & C	8(0)	BL	DEPL	STOW	-	-	-	-	-	
S2FRBS	Small Probe 2 SMFR Boom Deploy Status	SP	SSUBA, B, & C	8(1)	BL	DEPL	STOW	-	-	-	-	-	
S2FWIT	Small Probe 2 Forward Shelf Temp	B	BSUBA	20	AC	-	-	-	°C			-	
		B	BSUBC	46	AC	-	-	-	°C			-	
S2FW2T	Small Probe 2 Forward Shelf Temp	SP	SSUBA	29,61	AC	-	-	-	°C			-	
		SP	SSUBA	27,53	AC	-	-	-	°C			-	
		SP	SSUBC	29,40,61	AC	-	-	-	°C			-	
S2INTP	Small Probe 2 Internal Pressure	SP	SSUBA,SSUBB	14,30	AR	-	-	-	PSI			-	
		SP	SSUBA,SSUBB	46,61	AR	-	-	-	PSI			-	
		SP	SSUBC	2,14,18	AR	-	-	-	PSI			-	
		SP	SSUBC	30,34,46	AR	-	-	-	PSI			-	
		SP	SSUBC	50,62	AR	-	-	-	PSI			-	
S2RELS	Small Probe 2 Release Status	B	BLRG	8(1)	BL	STOW	RELS	-	-	-	-	-	
		B	BSUBC	0(1)	BL	STOW	RELS	-	-	-	-	-	
S3AFIT	Small Probe 3 Aft Shelf Temp	B	BSUBC	61	AC	-	-	-	°C			-	
S3AF2T	Small Probe 3 Aft Shelf Temp	SP	SSUBA, B, & C	13,45	AC	-	-	-	°C			-	
		SP	SSUBC	24	AC	-	-	-	°C			-	
S3ASBS	Small Probe 3 SAS Boom Deploy Status	SP	SSUBA, B, & C	8(0)	BL	DEPL	STOW	-	-	-	-	-	
S3FRBS	Small Probe 3 SMFR Boom Deploy Status	SP	SSUBA, B, & C	8(1)	BL	DEPL	STOW	-	-	-	-	-	
S3FWIT	Small Probe 3 Forward Shelf Temp	B	BSUBA	62	AC	-	-	-	°C			-	

APPENDIX B: TELEMETRY REFERENCE DATA
(MNEMONICS IN ALPHABETICAL ORDER)

MNEMONIC	DESCRIPTION	APPL S/C	FORMATS	WORD (BIT)	DATA TYPE	STATUS			E.V. CONVERSION		RANGE	SAMPLE TIME	REMARKS/REF
						1BIT -	"0" BIT -	MULT	UNIT	QUANTIZ			
S3FW2T	Small Probe 3 Forward Shelf Temp	SP	SSUBA	29,61	AC	-	-	-	°C			-	
		SP	SSUBB	21,53	AC	-	-	-	°C			-	
		SP	SSUBC	29,40,61	AC	-	-	-	°C			-	
S3INTP	Small Probe 3 Internal Pressure	SP	SSUBA,SSUBB	14,30	AN	-	-	-	PSI			-	
		SP	SSUBA,SSUBB	46,61	AN	-	-	-	PSI			-	
		SP	SSUBC	2,14,18	AN	-	-	-	PSI			-	
		SP	SSUBC	30,34,46	AN	-	-	-	PSI			-	
		SP	SSUBC	50,62	AN	-	-	-	PSI			-	
S3REL5	Small Probe 3 Release Status	B	BEMS	8(2)	BL	STOW	RELS	-	-	-	-	-	
		B	BSUBC	0(2)	BL	STOW	RELS	-	-	-	-	-	
SBIMST	BIMS Temp	B	BSUBA	43	AC	-	-	-	°C			-	
		B	BSUBC	40	AC	-	-	-	°C			-	
SBMPST	BMPS Temp	B	BSUBA	40	AC	-	-	-	°C			-	
		B	BUSBC	8	AC	-	-	-	°C			-	
SBOOMS	Magnetometer Boom Deploy Status	0	OSUBA	7(6)	BL	DEPL	STOW	-	-	-	-	-	
SHL01T	Shelf Temperature 1	B,0	BSUBB,OSUBB	52	AC	-	-	-	°C	-	-70.0 to +130.0	-	
SHL02T	" " 2	B,0	"	53	AC	-	-	-	°C	-	"	-	
SHL03T	" " 3	B,0	"	54	AC	-	-	-	°C	-	"	-	
SHL04T	" " 4	B,0	"	55	AC	-	-	-	°C	-	"	-	
SHL05T	" " 5	B,0	"	56	AC	-	-	-	°C	-	"	-	
SHL06T	" " 6	B,0	"	57	AC	-	-	-	°C	-	"	-	
SHL07T	" " 7	B,0	"	58	AC	-	-	-	°C	-	"	-	
SHL08T	" " 8	B,0	"	59	AC	-	-	-	°C	-	"	-	
SHL09T	" " 9	B,0	"	60	AC	-	-	-	°C	-	"	-	
SHL10T	" " 10	B,0	"	61	AC	-	-	-	°C	-	"	-	
SHL11T	" " 11	B,0	"	62	AC	-	-	-	°C	-	"	-	
SHL12T	" " 12	B,0	"	63	AC	-	-	-	°C	-	"	-	
SIMUTR	Time of S/C Change to Simulated SPR	B,0	N/A	N/A	CALC	CALCULATED					-	-	IN UT
									-	-	-	-	
									-	-	-	-	

APPENDIX B: TELEMETRY REFERENCE DATA
(MNEMONICS IN ALPHABETICAL ORDER)

MNEMONIC	DESCRIPTION	APPL S/C	FORMATS	WORD (BIT)	DATA TYPE	STATUS			E.V CONVERSION		RANGE	SAMPLE TIME	REMARKS/REF
						1BIT =	"0" BIT =	MULT	UNIT	QUANTIZ			
SLAF1T	Large Probe Aft Shelf Temp	B	BSUBA	39	AC	-	-	-	°C			-	
		B	BSUBC	13	AC	-	-	-	°C			-	
SLAF2T	Large Probe Aft Shelf Temp	LP	LSUBA,LSUBB	45	AC	-	-	-	°C			-	
SLFW1T	Large Probe Forward Shelf Temp	B	BENG	60	AC	-	-	-	°C			-	
		B	BSUBA	25	AC	-	-	-	°C			-	
		B	BSUBC	14	AC	-	-	-	°C			-	
SLFW2T	Large Probe Forward Shelf Temp	LP	LSUBA,LSUBB	53	AC	-	-	-	°C			-	
SLINTP	Large Probe Internal Pressure	LP	LSUBA,LSU_B	15,31	AM	-	-	-	PSI			-	
		LP	LSUBA,LSUBB	47,63	AM	-	-	-	PSI			-	
SLIRWT	LIR Window Temp	LP	LSUBA,LSUBB	61	AC	-	-	-				-	
SLRELS	Large Probe Release Status	B	BENG	8(3)	BL	STOW	RELS	-	-	-	-	-	
		B	BSUBC	0(3)	BL	STOW	RELS	-	-	-	-	-	
SOETPT	OETP Temp	O	OSUBA	41	AC	-	-	-	°C	-	-70.0 to +130.0	-	
SOIMST	OIMS Temp	O	OSUBA	40	AC	-	-	-	°C	-	"	-	
SOMMST	OMMS Temp	O	OSUBA	42	AC	-	-	-	°C	-	"	-	
SOPAIT	OPA Temp	O	OSUBA	43	AC	-	-	-	°C	-	"	-	
SORADT	ORAD Temp	O	OSUBA	44	AC	-	-	-	°C	-	"	-	
SORPAT	ORPA Temp	O	OSUBA	39	AC	-	-	-	°C	-	"	-	
SSEP1S	S/C Separation Status	B,O	BCMBO,OCMBO	8(0)	BL	SEP	NSEP	-	-	-	-	-	
		B,O	BACS,OACS	31(0)	BL	SEP	NSEP	-	-	-	-	-	
		B,O	BENG,OENG,OLACR	15(0)	BL	SEP	NSEP	-	-	-	-	-	
		O	OSUBA	15(0)	BL	SEP	NSEP	-	-	-	-	-	
SSEP2S	S/C Separation Status	B,O	BCMBO,OCMBO	36(0)	BL	SEP	NSEP	-	-	-	-	-	
		B,O	BACS,OACS	63(0)	BL	SEP	NSEP	-	-	-	-	-	
		B,O	BENG,OENG	47(0)	BL	SEP	NSEP	-	-	-	-	-	
		O	OLACR	47(0)	BL	SEP	NSEP	-	-	-	-	-	
		B,O	BSUBA,OSUBA	18(0)	BL	SEP	NSEP	-	-	-	-	-	
VALV1S	Latch Valve 1 Position Status	B,O	BSUBA,OSUBA	7(4)	BL	OPEN	CLOS	-	-	-	-	-	
VALV2S	Latch Valve 2 Position Status	B,O	BSUBA,OSUBA	7(5)	BL	OPEN	CLOS	-	-	-	-	-	
WCASET	DIM Case Temp	O	OSUBA	25	AC	-	-	-				-	

APPENDIX B: TELEMETRY REFERENCE DATA
(MNEMONICS IN ALPHABETICAL ORDER)

MNEMONIC	DESCRIPTION	APPL S/C	FORMATS	WORD (BIT)	DATA TYPE	STATUS			E.V. CONVERSION		RANGE	SAMPLE TIME	REMARKS/REF
						MBT -	"0" BIT -	MULT.	UNIT	QUANTIZ.			
WFDVHS	Fill and Drain Valve Prt. Heater Status	B	BSUBB	50	AR	-	-	-	AMPS	-	0.0 to 0.14	-	
		D	DSUBD	62	AR	-	-	-	AMPS	-	0.0 to 0.14	-	
WJ12HS	Radial Jets 1/2 Line Heaters Status	B	BSUBB	13	AR	-	-	-	AMPS	-	0.0 to 0.30	-	
		D	DSUBD	30	AR	-	-	-	AMPS	-	0.0 to 0.30	-	
WJ34HS	Radial Jets 3/4 Line Heaters Status	B	BSUBB	45	AR	-	-	-	AMPS	-	0.0 to 0.22	-	
		B	DSUBB	35	AR	-	-	-	AMPS	-	0.0 to 0.22	-	

APPENDIX B: TELEMETRY REFERENCE DATA
(MNEMONICS IN ALPHABETICAL ORDER)

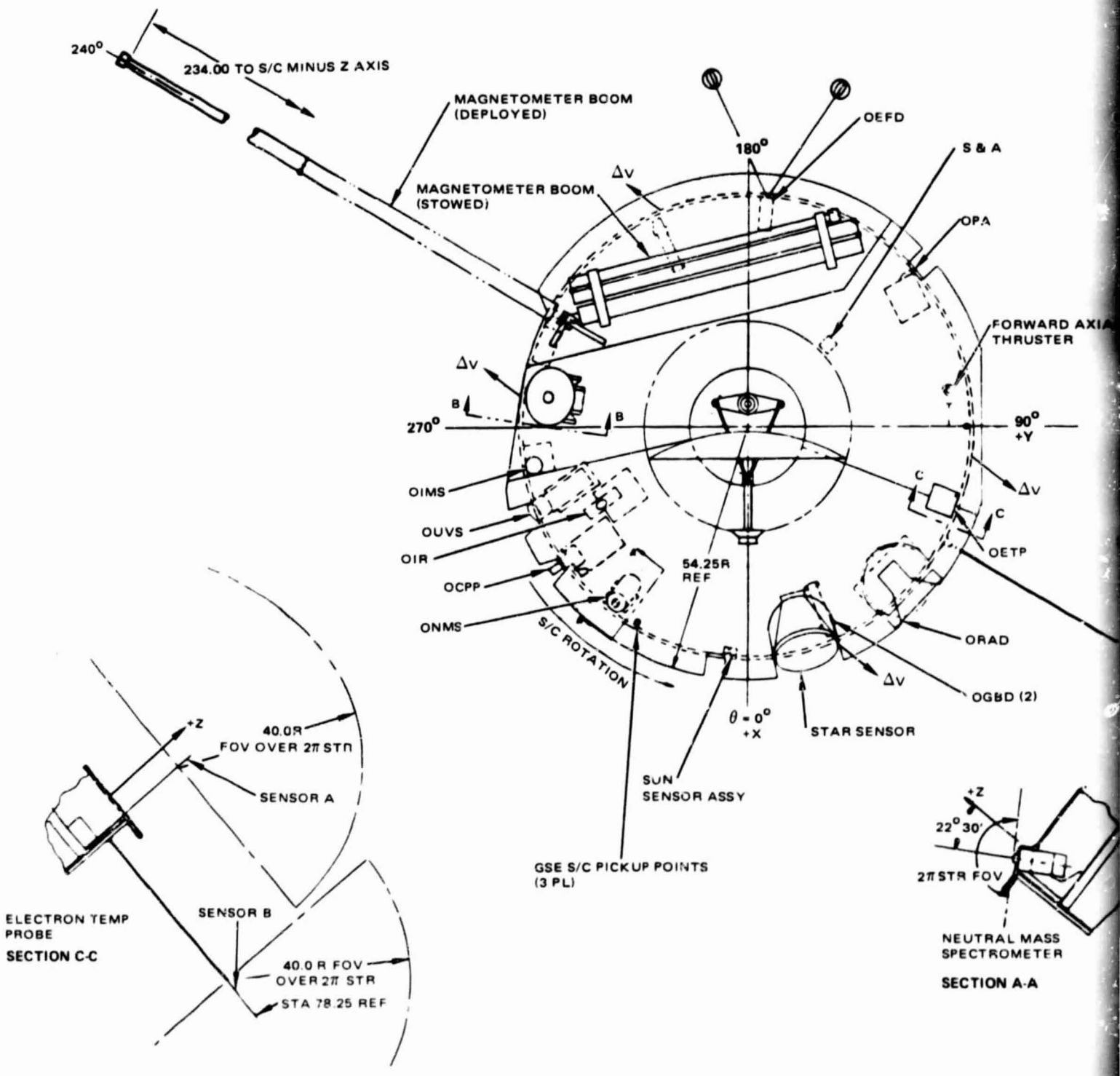
MNEMONIC	DESCRIPTION	APPL S/C	FORMATS	WORD (BIT)	DATA TYPE	STATUS BIT -	STATUS "0" BIT -	STATUS MULT	E.V. CONVERSION UNIT	E.V. CONVERSION QUANTIZ.	RANGE	SAMPLE TIME	REMARKS/REF
WJET1T	Radial Jet 1 Temp	B,D	BACS,BACS,	32	AC	-	-	-	°C	-	-180.0 to +180.0	-	
		B,D	DSABA,DSABA	38	AC	-	-	-	°C	-	"	-	
		B	DSABC	24	AC	-	-	-	°C	-	"	-	
WJET2T	Radial Jet 2 Temp	BB	BACS,BACS	33	AC	-	-	-	°C	-	"	-	
		B,D	DSABA,DSABA	45	AC	-	-	-	°C	-	"	-	
		B	DSABC	25	AC	-	-	-	°C	-	"	-	
WJET3T	Radial Jet 3 Temp	B,D	BACS,BACS	34	AC	-	-	-	°C	-	-180.0 to +180.0	-	
		B,D	DSABA,DSABA	48	AC	-	-	-	°C	-	"	-	
		B	DSABC	26	AC	-	-	-	°C	-	"	-	
WJET4T	Radial Jet 4 Temp	B,D	BACS,BACS	35	AC	-	-	-	°C	-	"	-	
		B	DSABC	32	AC	-	-	-	°C	-	"	-	
WJET5T	Forward Axial Jet 5 Temp	B,D	BACS,BACS	36	AC	-	-	-	°C	-	-180.0 to +180.0	-	
		B,D	DSABA,DSABA	31	AC	-	-	-	°C	-	"	-	
		B	DSABC	33	AC	-	-	-	°C	-	"	-	
WJET6T	Forward Axial Jet 6 Temp	B,D	BACS,BACS	37	AC	-	-	-	°C	-	"	-	
		B,D	DSABA,DSABA	39	AC	-	-	-	°C	-	"	-	
		B	DSABC	34	AC	-	-	-	°C	-	"	-	
WJET7T	Forward Axial Jet 7 Temp	D	DACS	38	AC	-	-	-	°C	-	-180.0 to +180.0	-	
		D	DSABB	51	AC	-	-	-	°C	-	"	-	
WJET8T	Propellant Line 1 Temp	B,D	DSABA,DSABA	35	AC	-	-	-	°C	-	-10.0 to +80.0	-	
WJET9T	" " 2 "	B,D	DSABA,DSABA	36	AC	-	-	-	°C	-	"	-	
WJET0T	" " 3 "	B,D	DSABA,DSABA	37	AC	-	-	-	°C	-	"	-	
WJET1T	" " 4 "	B,D	DSABA,DSABA	38	AC	-	-	-	°C	-	"	-	
WLVPRS	Latch Valve Primary Monitor Status	B	DSABA	49	AM	-	-	-	AMPS	-	0.0 to 0.6	-	
		D	DSABD	57	AM	-	-	-	AMPS	-	0.0 to 0.6	-	
WSAFET	SAM Safe and Arm Temp	D	DSABB	26	AC	-	-	-	°C	-		-	
WTANKP	Tank Pressure	B,D	BACS,BACS	11	AC	-	-	-	PSI	-	0.0 to 360	-	
		B,D	BENG,BENG	63	AC	-	-	-	PSI	-	0.0 to 360	-	
		B,D	DSABA,DSABA	21	AC	-	-	-	PSI	-	0.0 to 360	-	
		BB	BETT	56	AC	-	-	-	PSI	-	0.0 to 360	-	

APPENDIX B: TELEMETRY REFERENCE DATA
(MNEMONICS IN ALPHABETICAL ORDER)

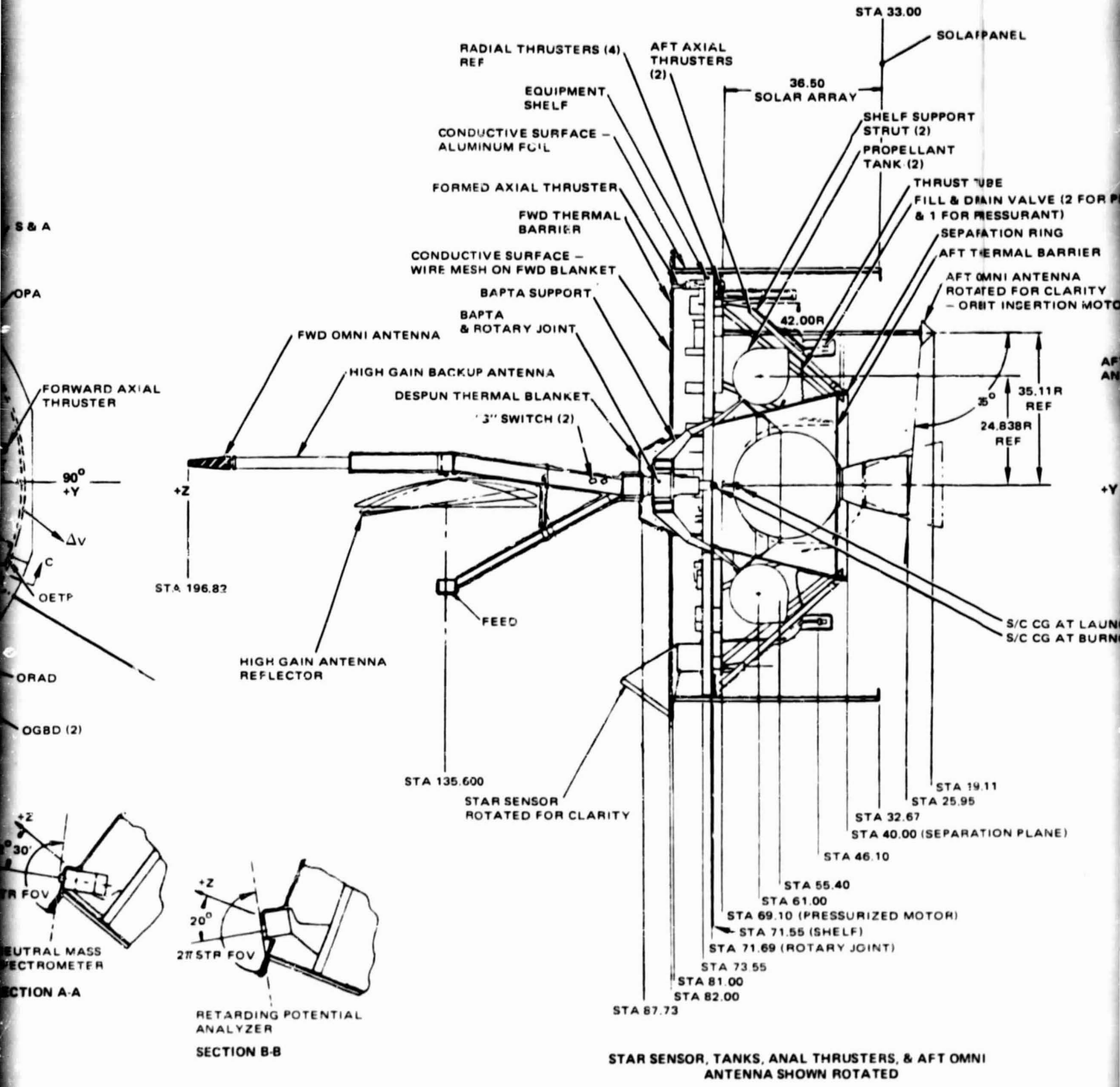
MNEMONIC	DESCRIPTION	APPL. S/C	FORMATS	WORD (BIT)	DATA TYPE	STATUS			E.V. CONVERSION		RANGE	SAMPLE TIME	REMARKS/REF
						1BIT =	"0" BIT =	MULT.	UNIT	QUANTIZ.			
VTHRST	OIM Thrust Temp	0	RSUB8	24	AC	-	-	-	°C			-	
VTNK1T	Tank 1 Temp	8.0	RSUB8,OSUB8	22	AC	-	-	-	°C	-	-10.0 to +80.0	-	
VTNK2T	Tank 2 Temp	8.0	RSUB8,OSUB8	23	AC	-	-	-	°C	-	-10.0 to +80.0	-	

APPENDIX C

FOLD-OUT DIAGRAMS



FOLDOUT FRAME

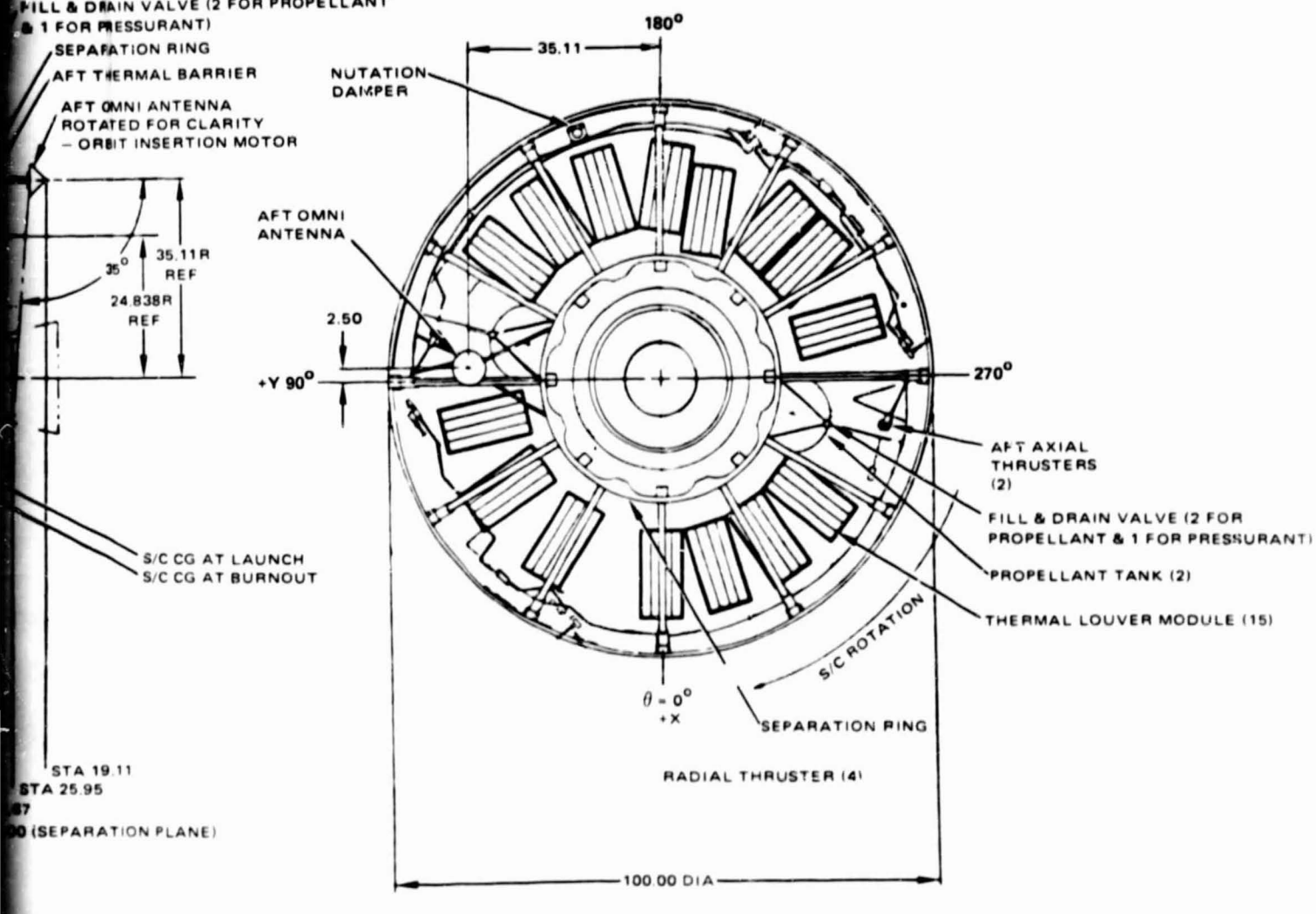


STAR SENSOR, TANKS, ANAL THRUSTERS, & AFT OMNI ANTENNA SHOWN ROTATED

FOLDOUT FRAME 2

NOTES:

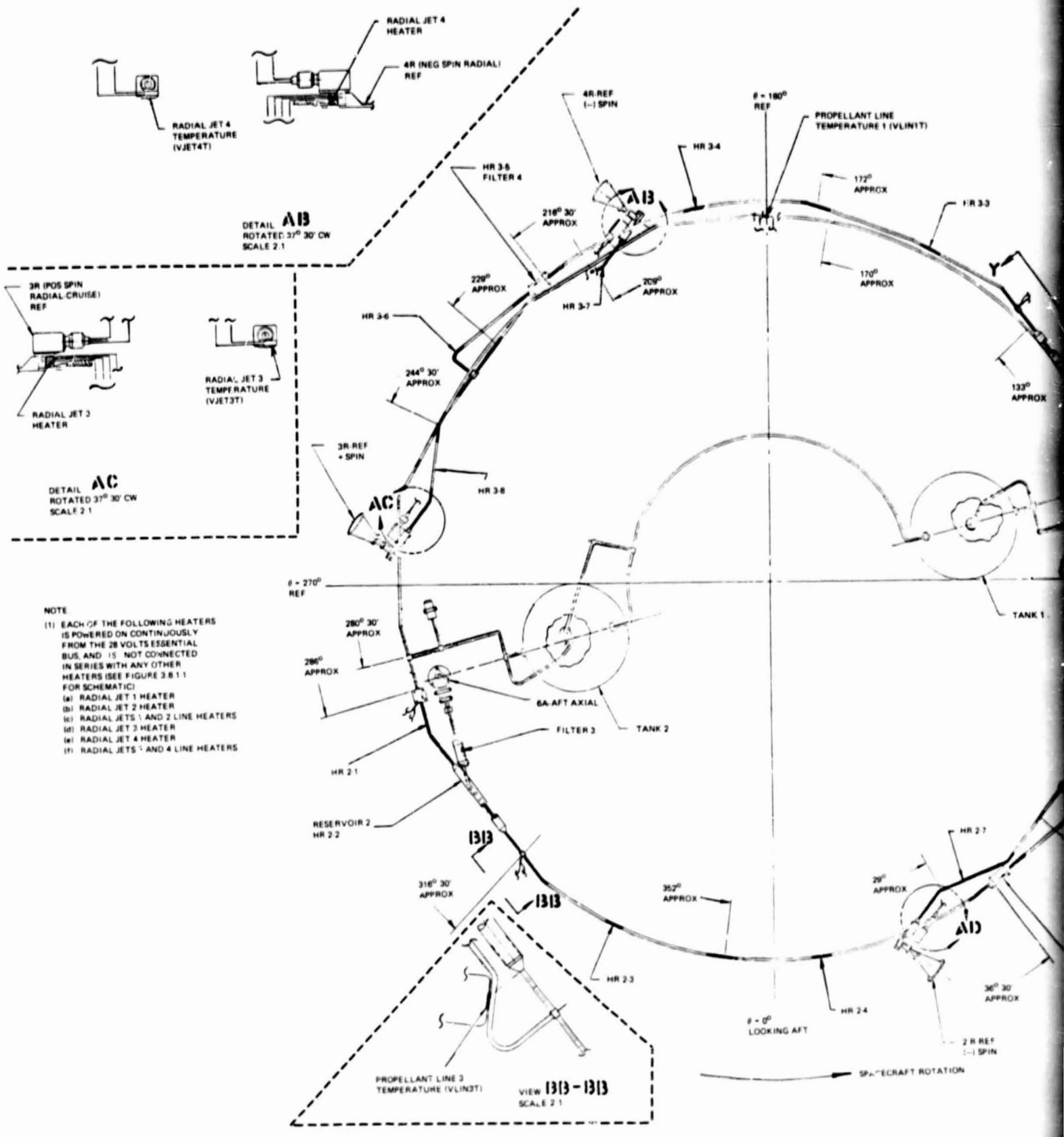
(1) ALL DIMENSIONS SHOWN ARE IN INCHES.



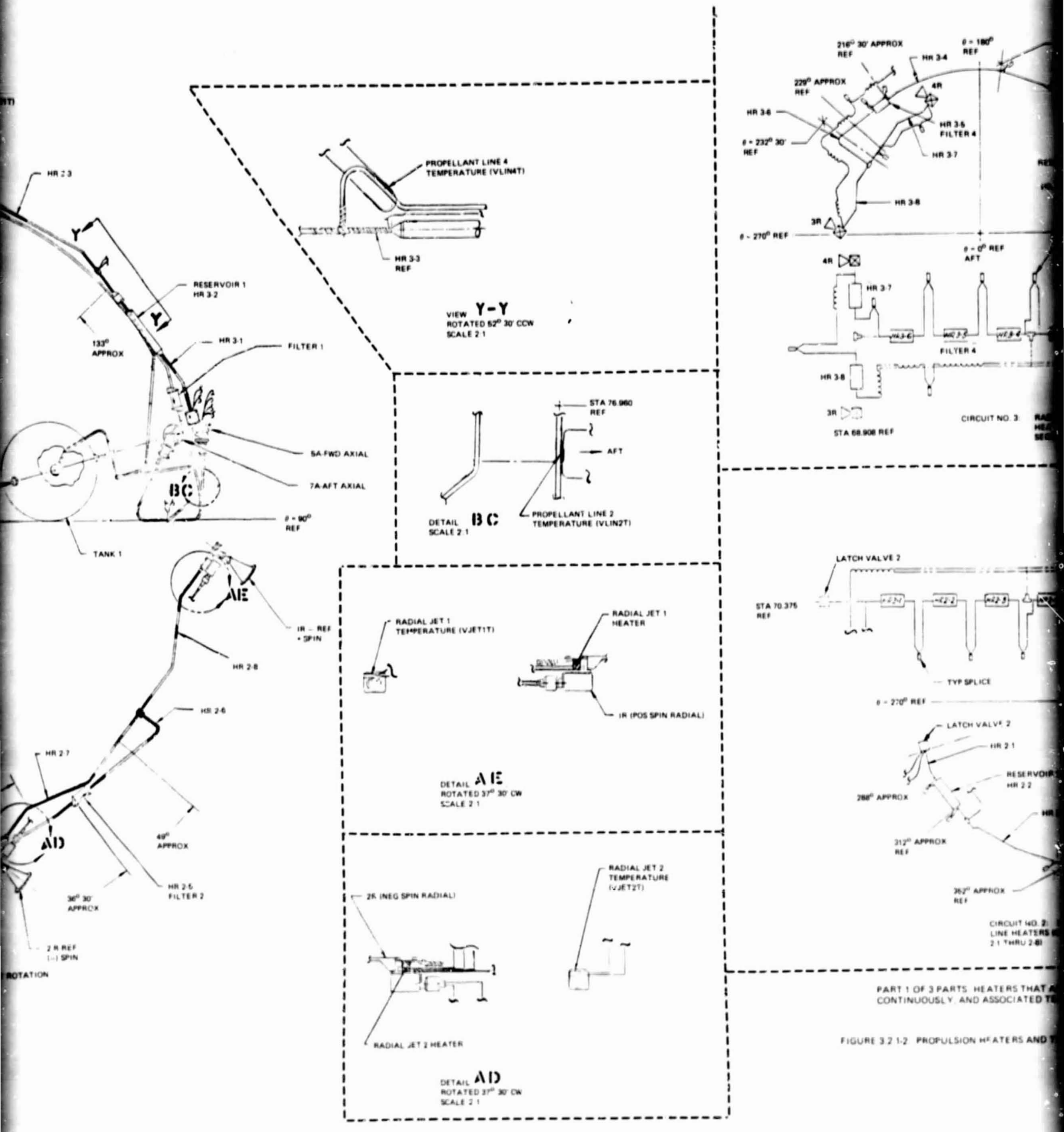
THERMAL BLANKETS OMITTED FOR CLARITY

FIGURE 3.1.1.1-1. ORBITER GENERAL ARRANGEMENT

FOLDOUT FRAME 3



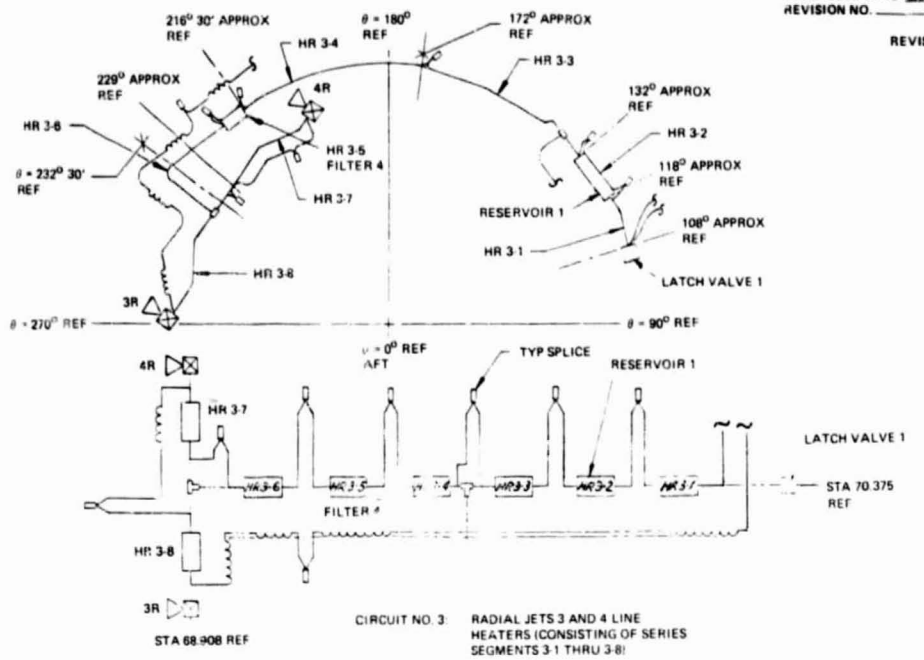
FOLDOUT FRAME



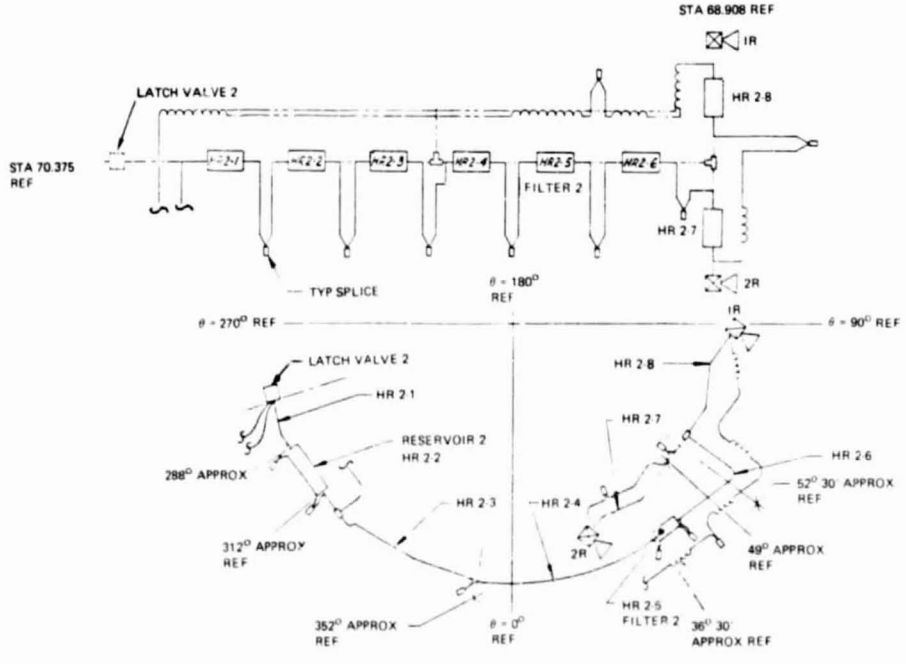
PART 1 OF 3 PARTS. HEATERS THAT A
CONTINUOUSLY. AND ASSOCIATED TE

FIGURE 3.2.1-2. PROPULSION HEATERS AND T

FOLDOUT FRAME



CIRCUIT NO. 3: RADIAL JETS 3 AND 4 LINE HEATERS (CONSISTING OF SERIES SEGMENTS 3-1 THRU 3-8)

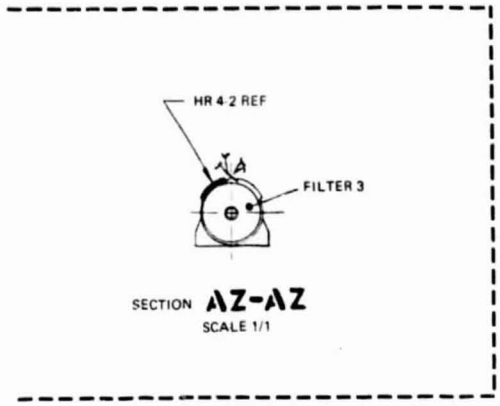


CIRCUIT NO. 2: RADIAL JETS 1 AND 2 LINE HEATERS (CONSISTING OF SERIES SEGMENTS 2-1 THRU 2-8)

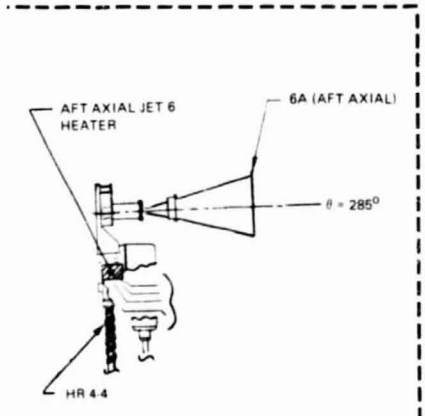
PART 1 OF 3 PARTS. HEATERS THAT ARE POWERED ON CONTINUOUSLY AND ASSOCIATED TELEMETERED SENSORS

FIGURE 3.2.1-2. PROPULSION HEATERS AND TEMPERATURE SENSORS

BOILDOUT FRAME 3

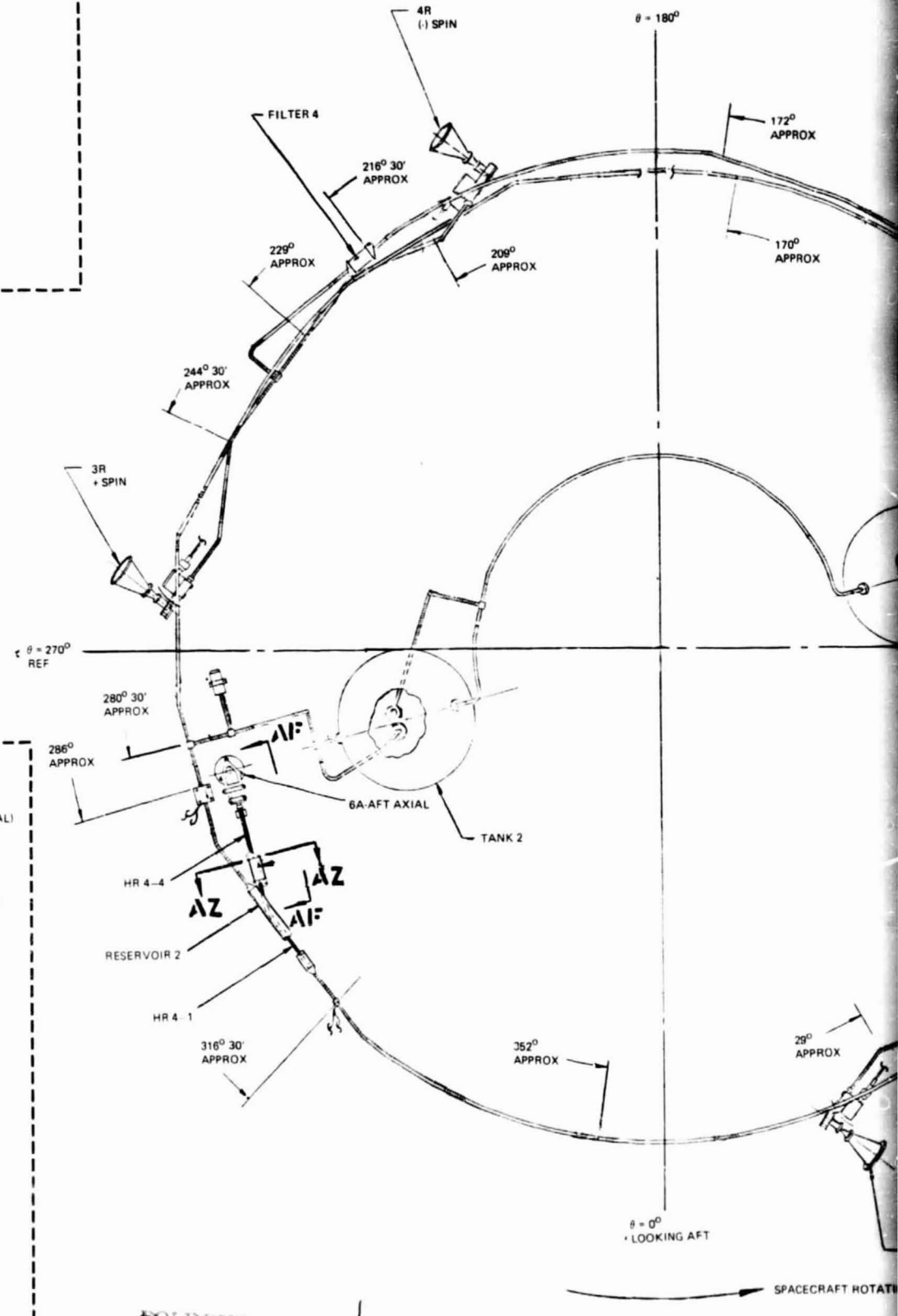


SECTION **AZ-AZ**
SCALE 1/1





VIEW **AF-AF**
SCALE 2 1



FOLDOUT FRAME

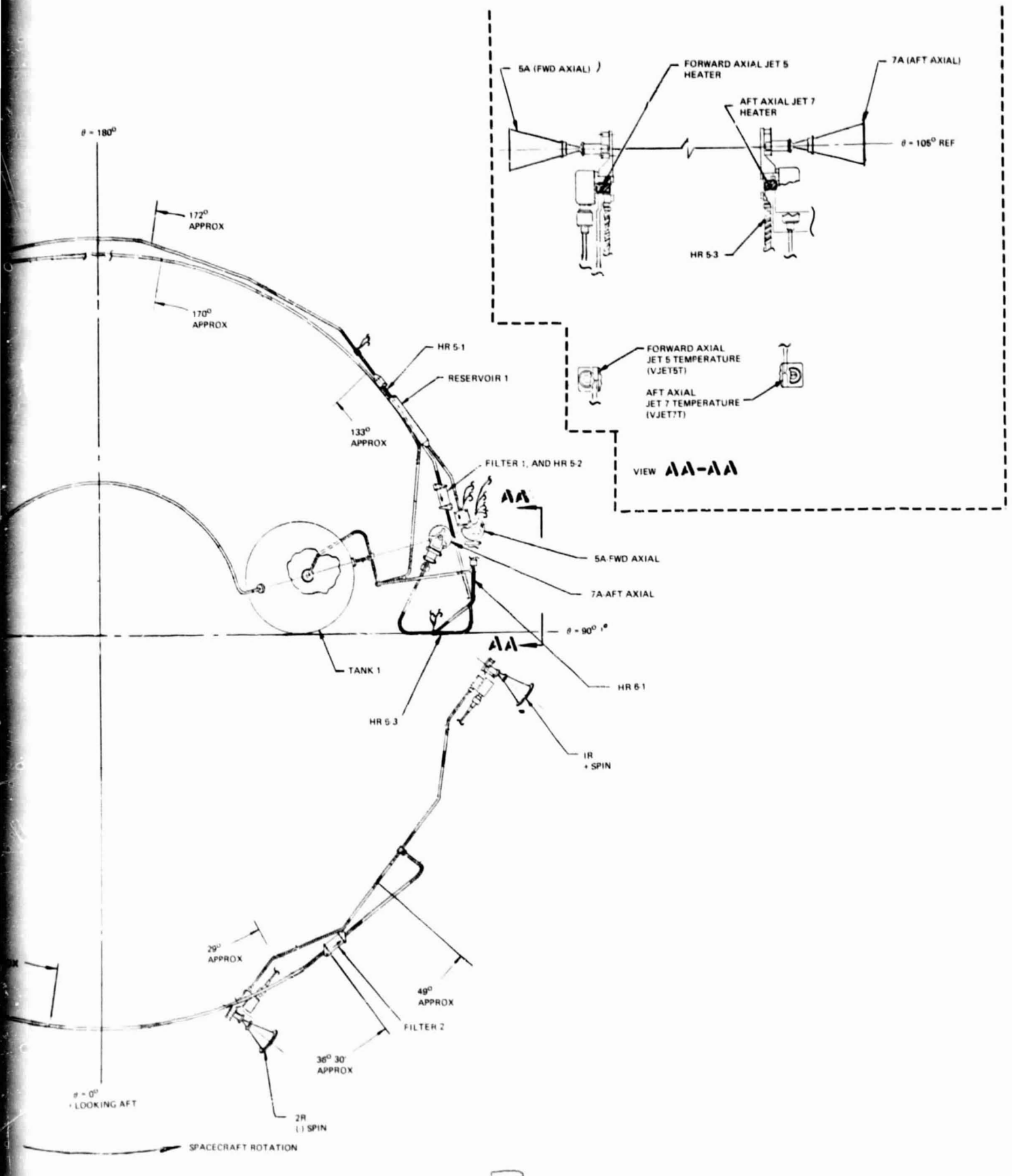
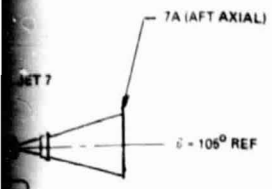
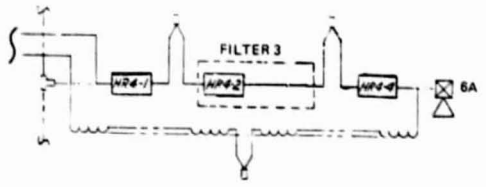


FIGURE 3.2.1

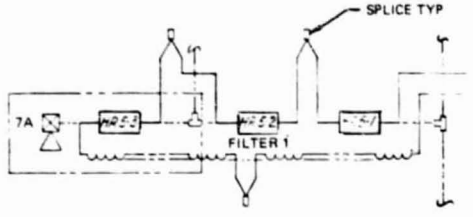
FOLDOUT FRAME 2



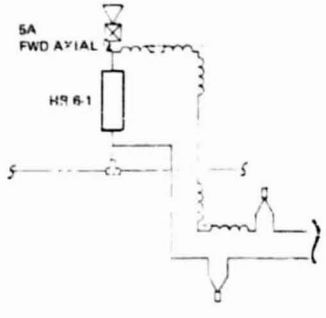
REVISION



CIRCUIT NO. 4: AFT AXIAL JET 6 LINE HEATERS (CONSISTING OF SERIES SEGMENTS 4-1, 4-2, AND 4-4)



CIRCUIT NO. 5: AFT AXIAL JET 7 LINE HEATERS (CONSISTING OF SERIES SEGMENTS 5-1 THRU 5-3)

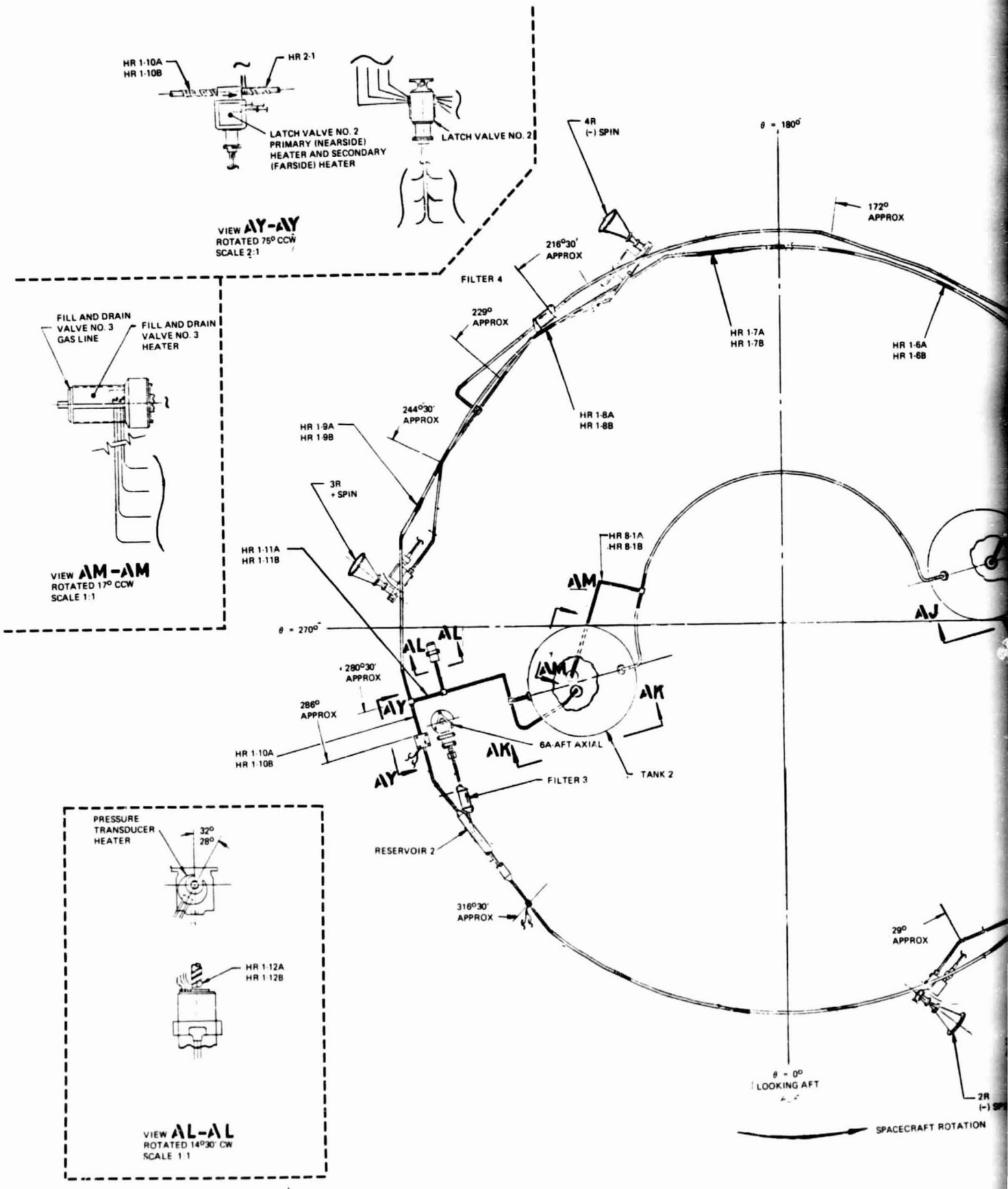


CIRCUIT NO. 6: FORWARD AXIAL JET 5 LINE HEATER (CONSISTING ONLY OF ELEMENT 6-1)

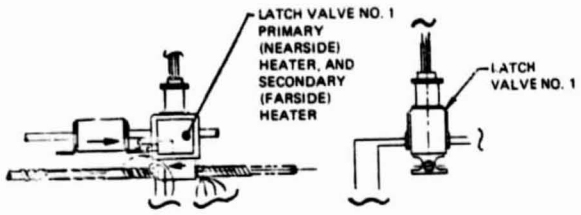
PART 2 OF 3 PARTS: AXIAL JETS HEATERS, ASSOCIATED LINES HEATERS, AND ASSOCIATED TELEMETERED TEMPERATURE SENSORS

FIGURE 3.2.1-2 PROPULSION HEATERS AND TEMPERATURE SENSORS

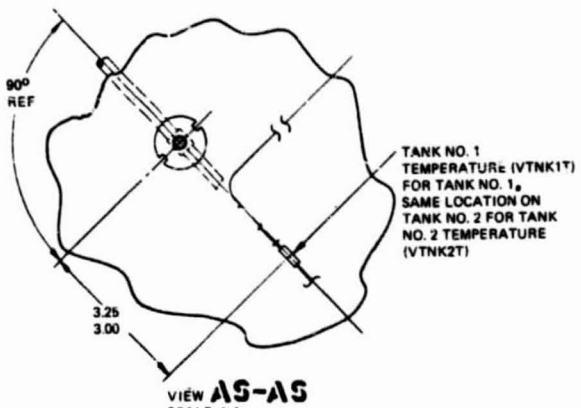
FOLDOUT FRAME 3



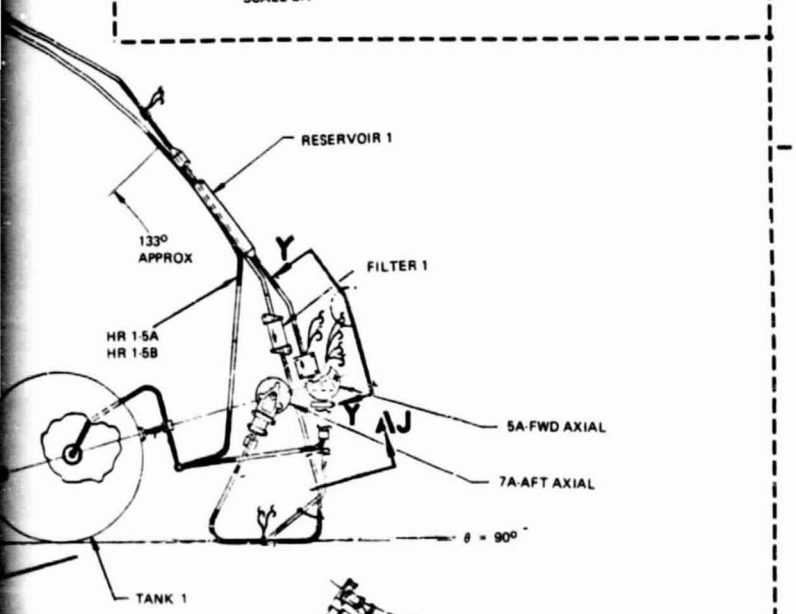
FOLDOUT FRAME

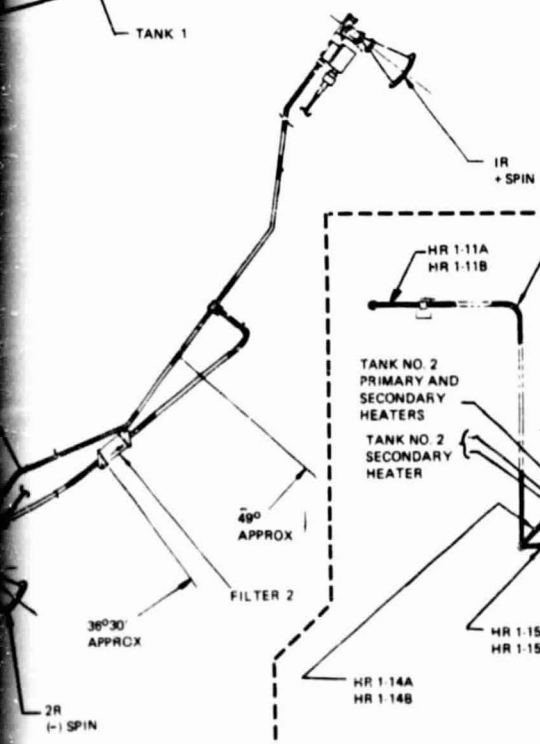


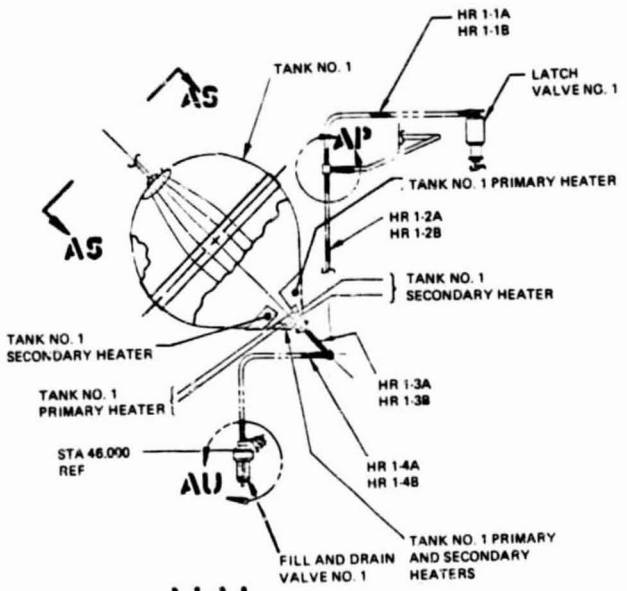
VIEW **Y-Y**
ROTATED 52°30' CCW
SCALE 2:1



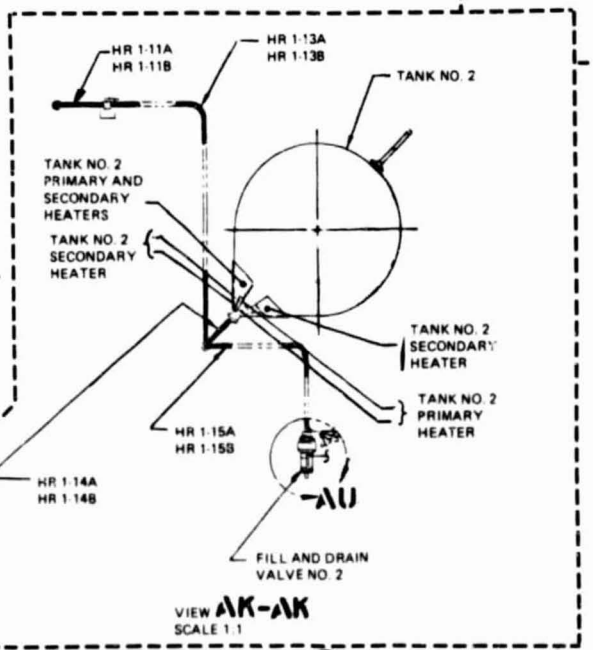
VIEW **AS-AS**
SCALE 1:1



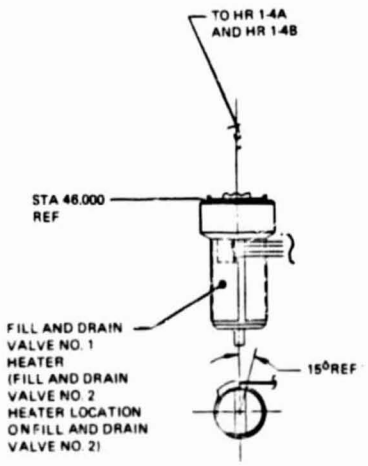




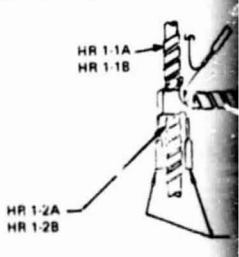
VIEW **AJ-AJ**
SCALE 1:1



VIEW **AK-AK**
SCALE 1:1

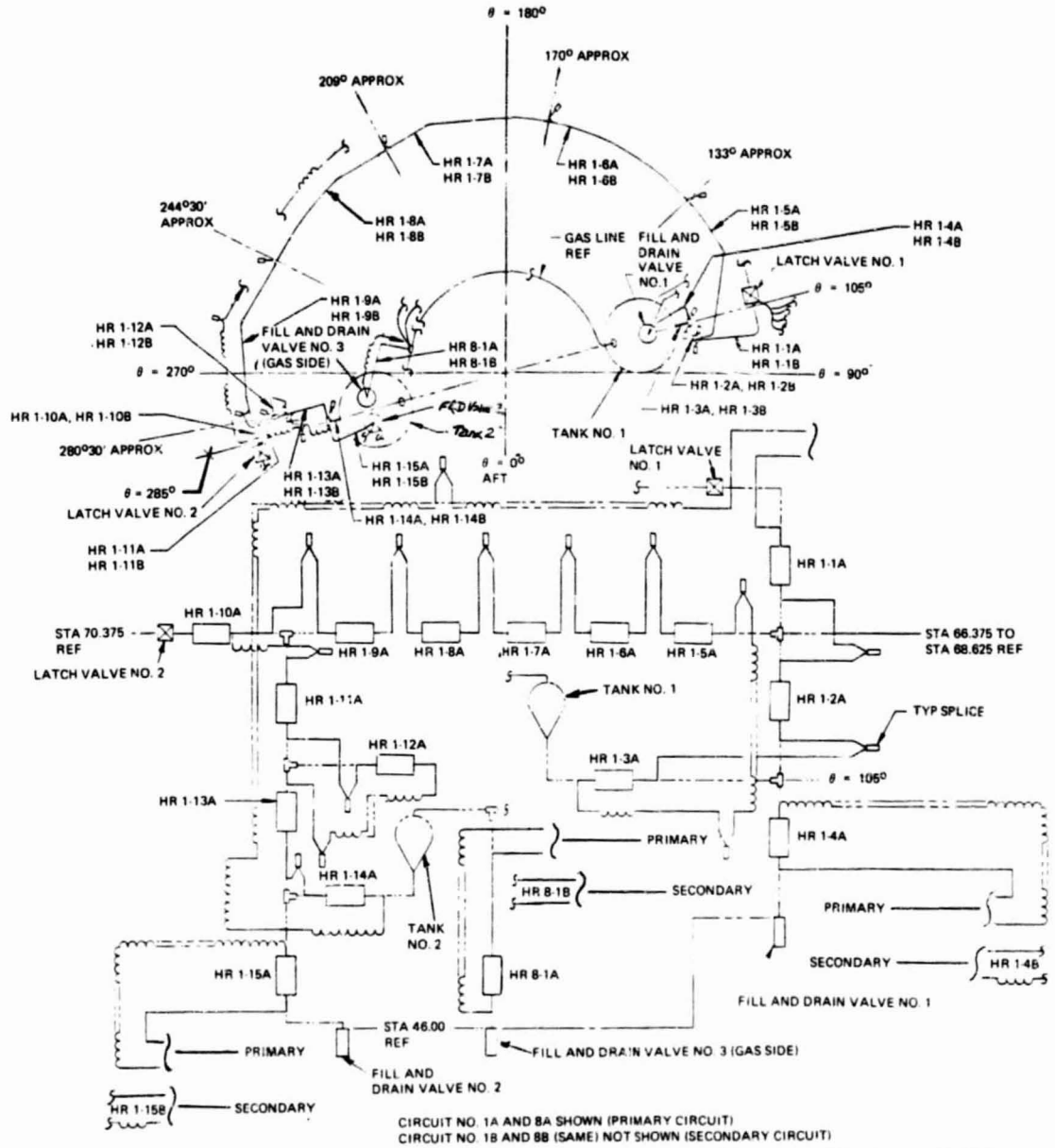


DETAIL **AU**
SCALE 1:1



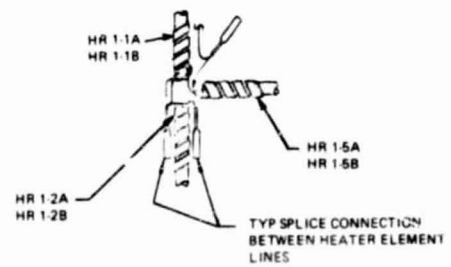
DETAIL **AP**
SCALE 1:1

FOLDOUT FRAME



CIRCUIT NO. 1A AND 8A SHOWN (PRIMARY CIRCUIT)
CIRCUIT NO. 1B AND 8B (SAME) NOT SHOWN (SECONDARY CIRCUIT)

TANK LINES HEATERS AND FILL AND DRAIN VALVES (NO. 1, NO. 2 AND NO. 3) LINES HEATERS

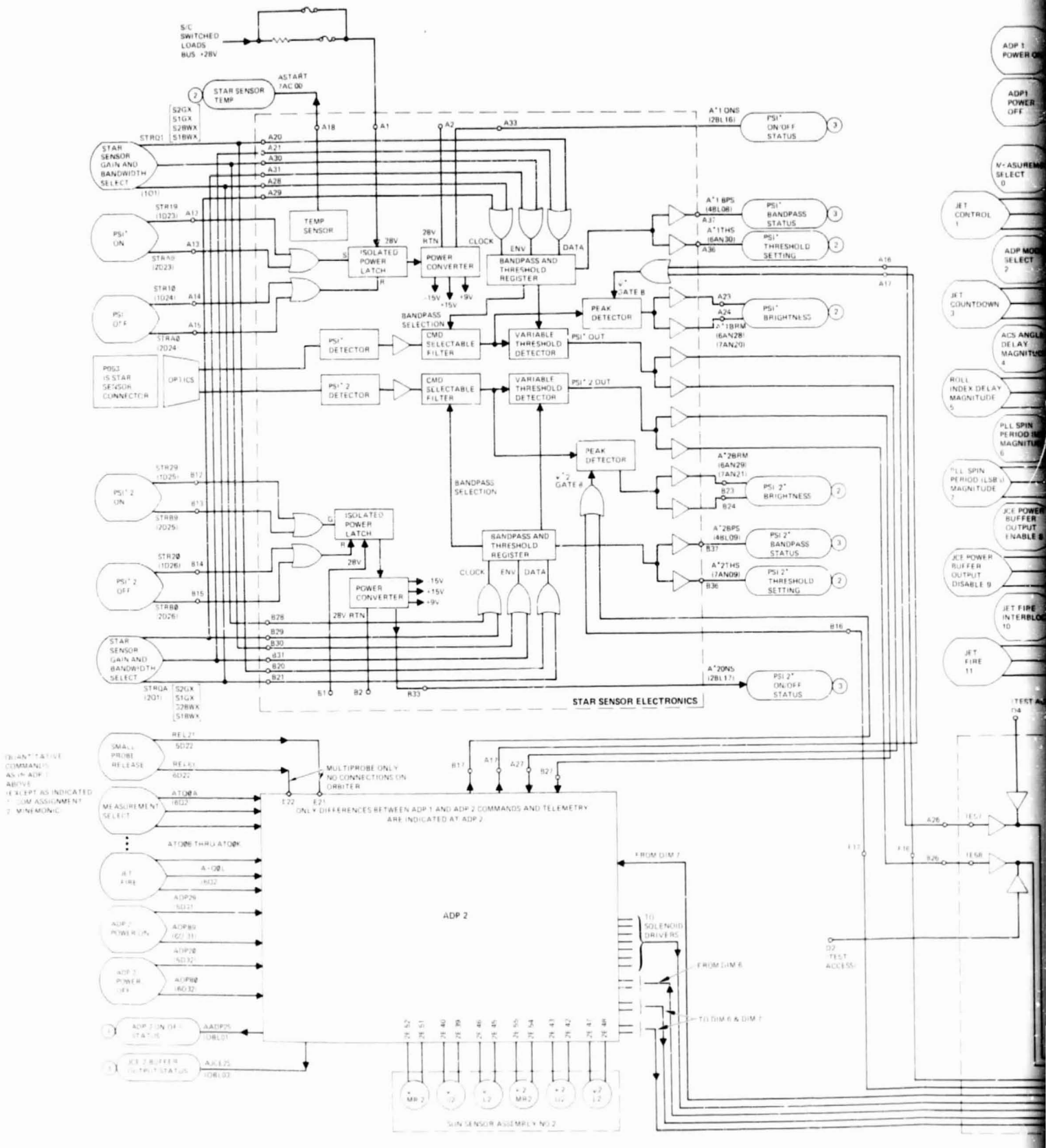


DETAIL AP
SCALE 1:1

PART 3 OF 3 PARTS: ALL HEATERS CONTROLLED BY THE COMMAND "PRIMARY PROPELLANT TANK HEATERS SELECT" (HTT19 OR HTTA9), AND THE COMMAND "SECONDARY PROPELLANT TANK HEATERS SELECT" (HTT29 OR HTT89); AND TELEMETERED TANKS TEMPERATURE SENSORS

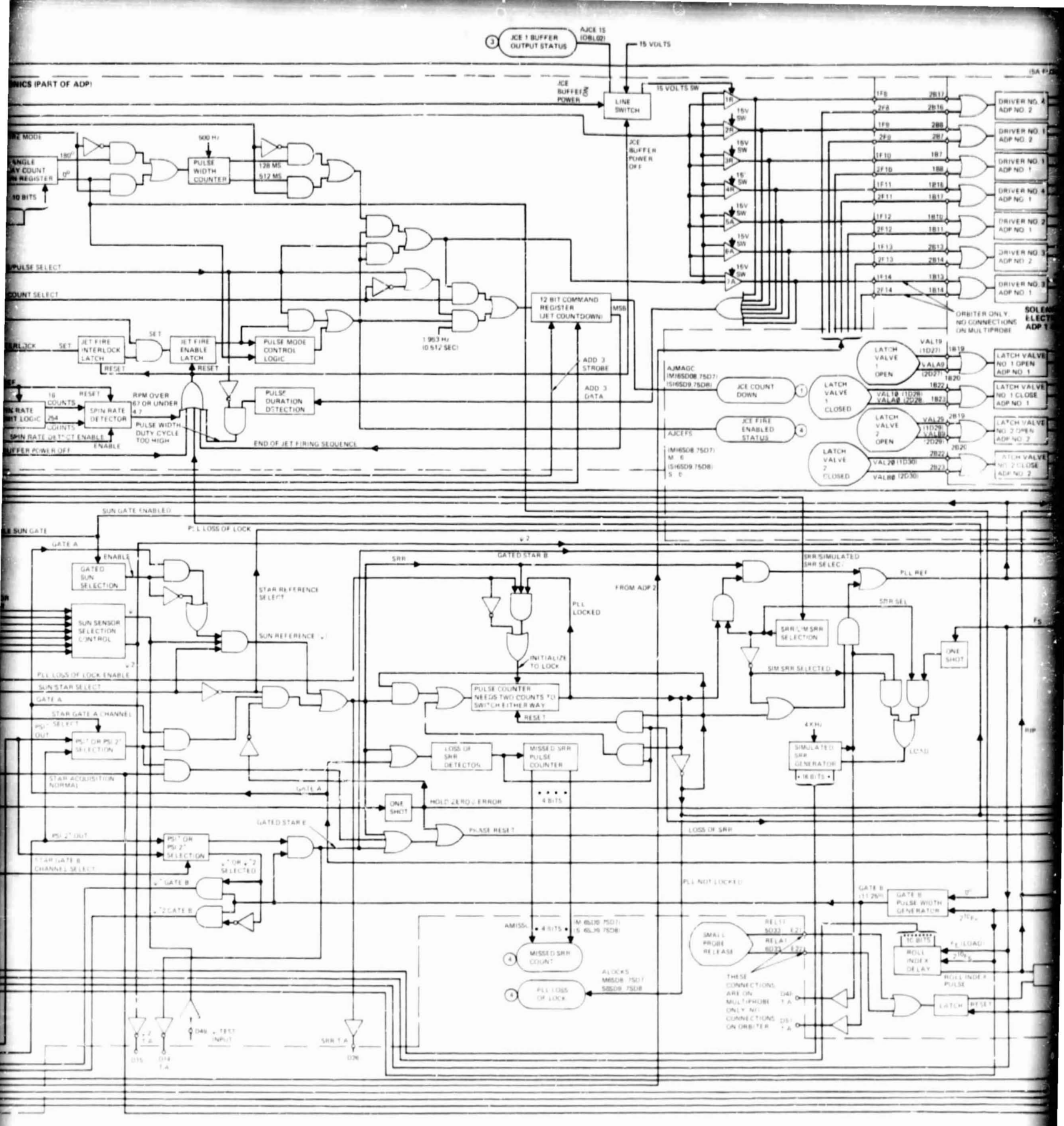
FIGURE 3.2.1-2. PROPULSION HEATERS AND TEMPERATURE SENSORS

FOLDOUT FRAME

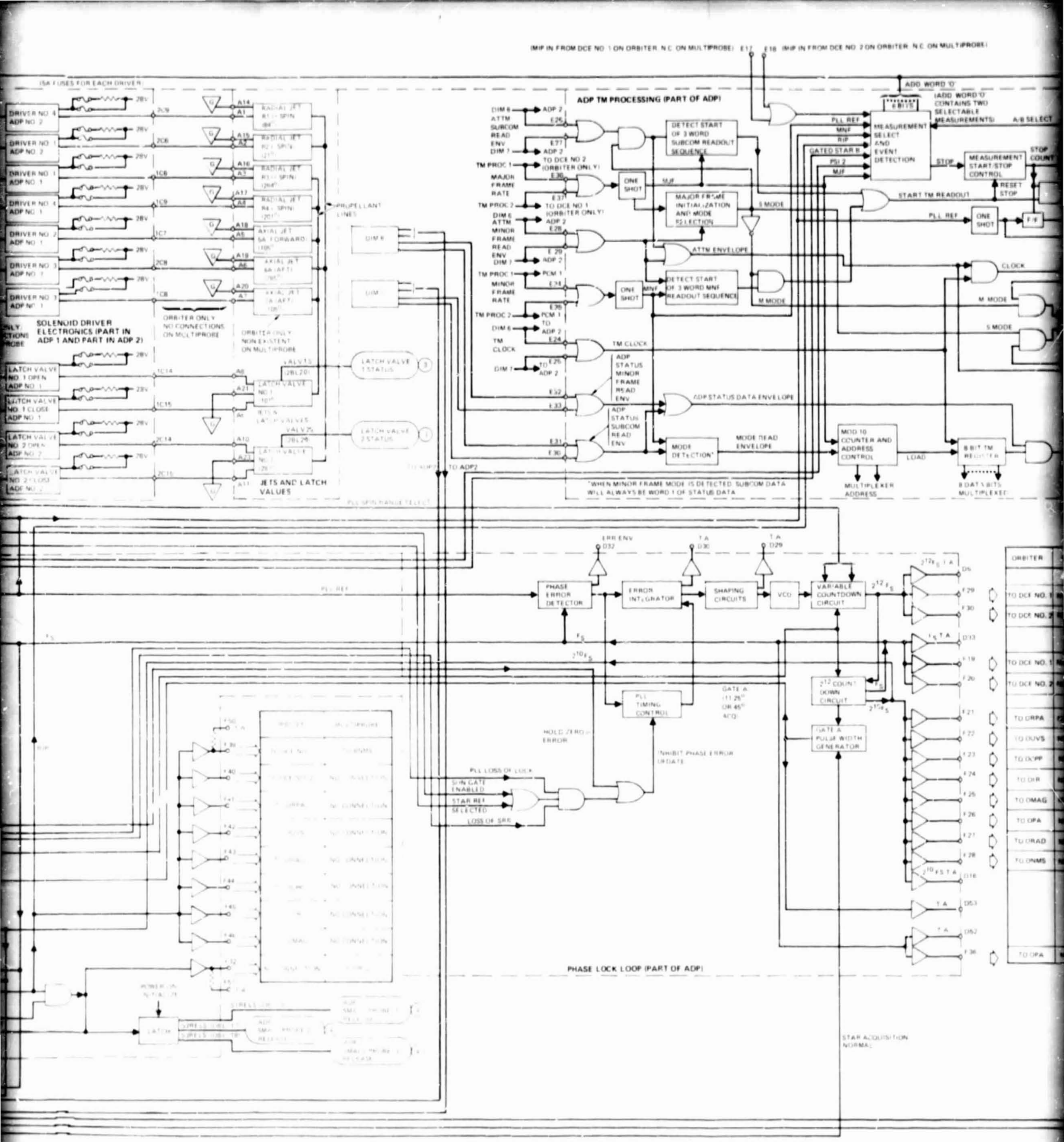


FOLDOUT FRAME

FOLDOUT FRAME 2



FOLDOUT FRAME 3



FOLDOUT FRAME 4

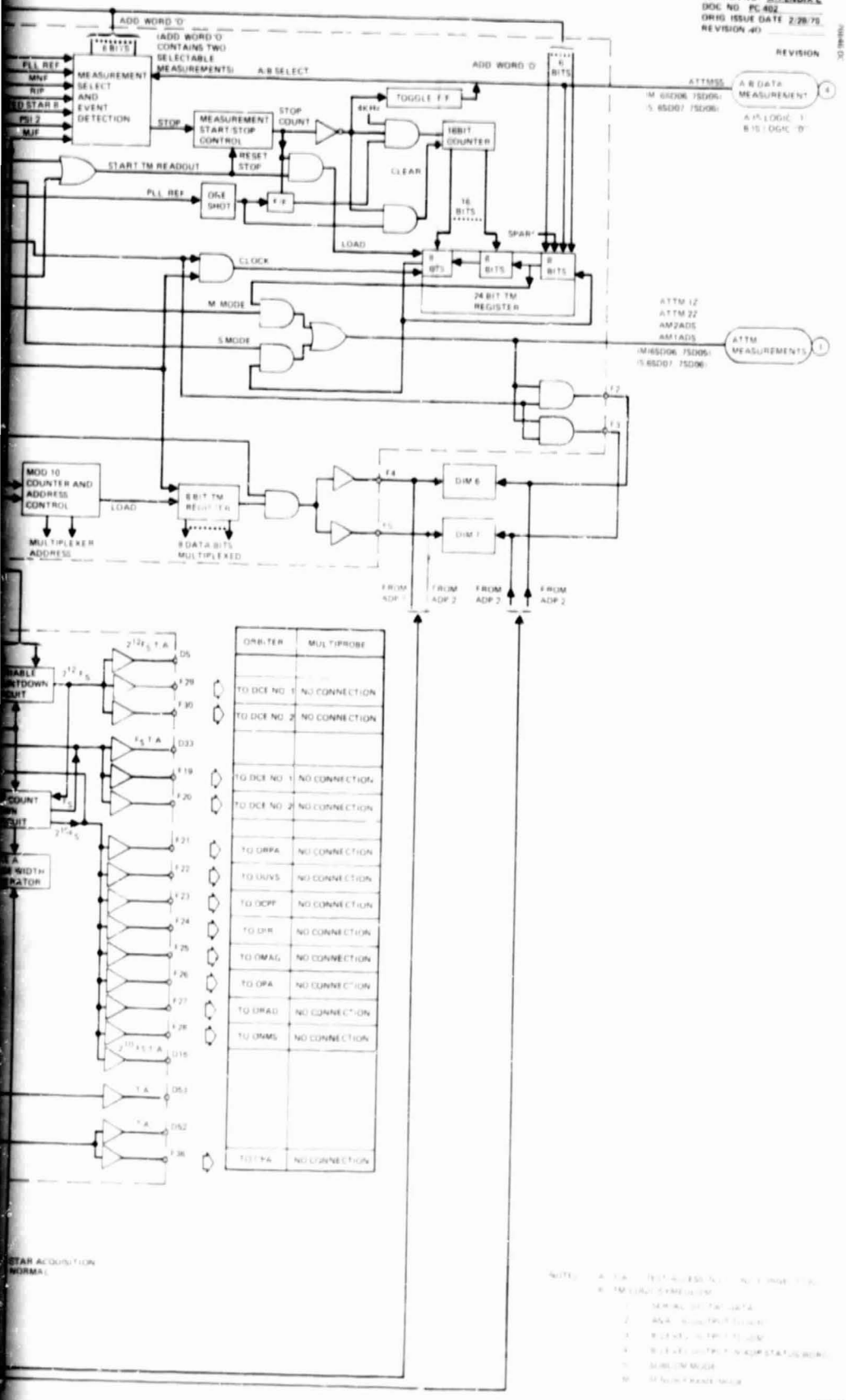
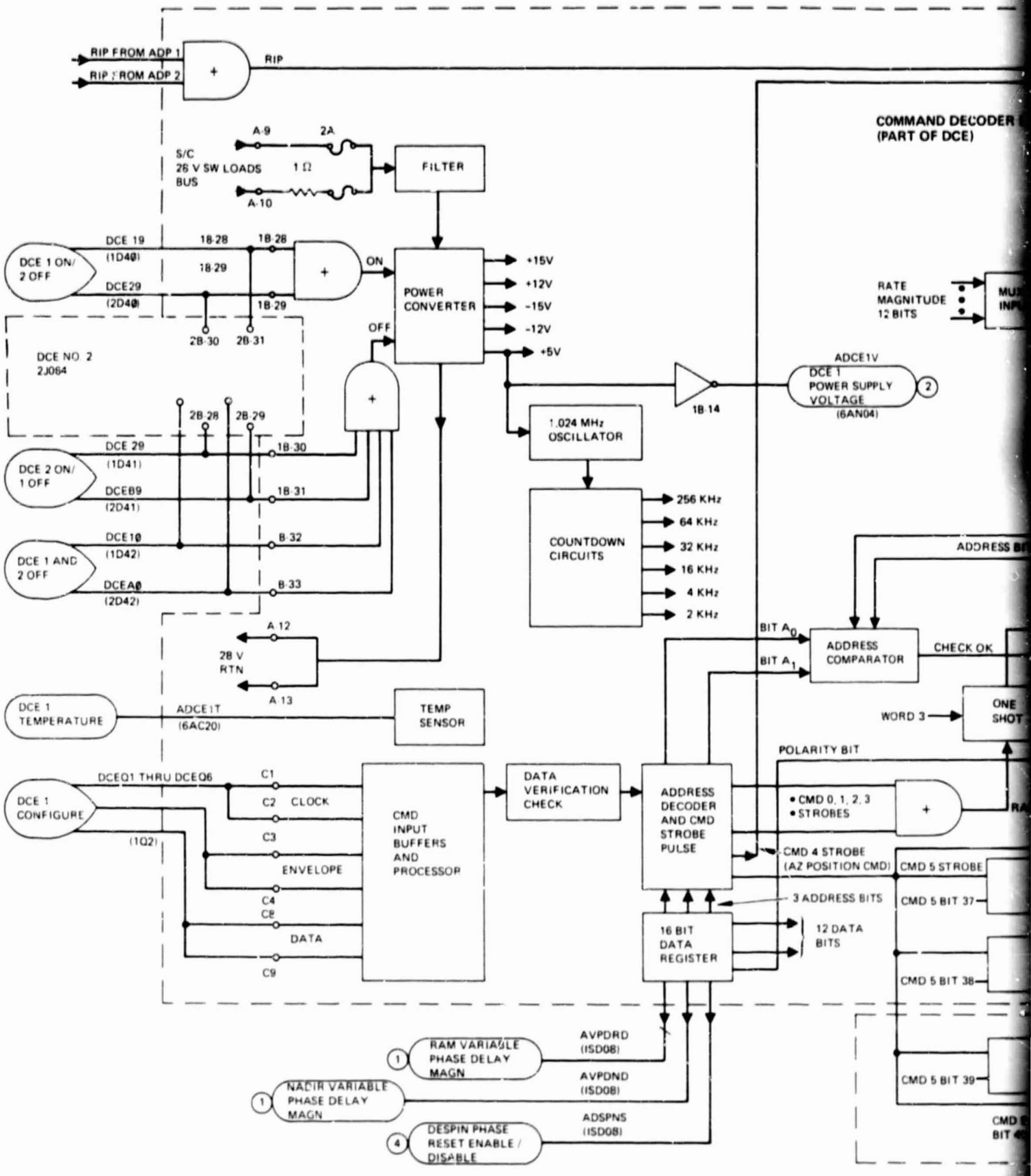


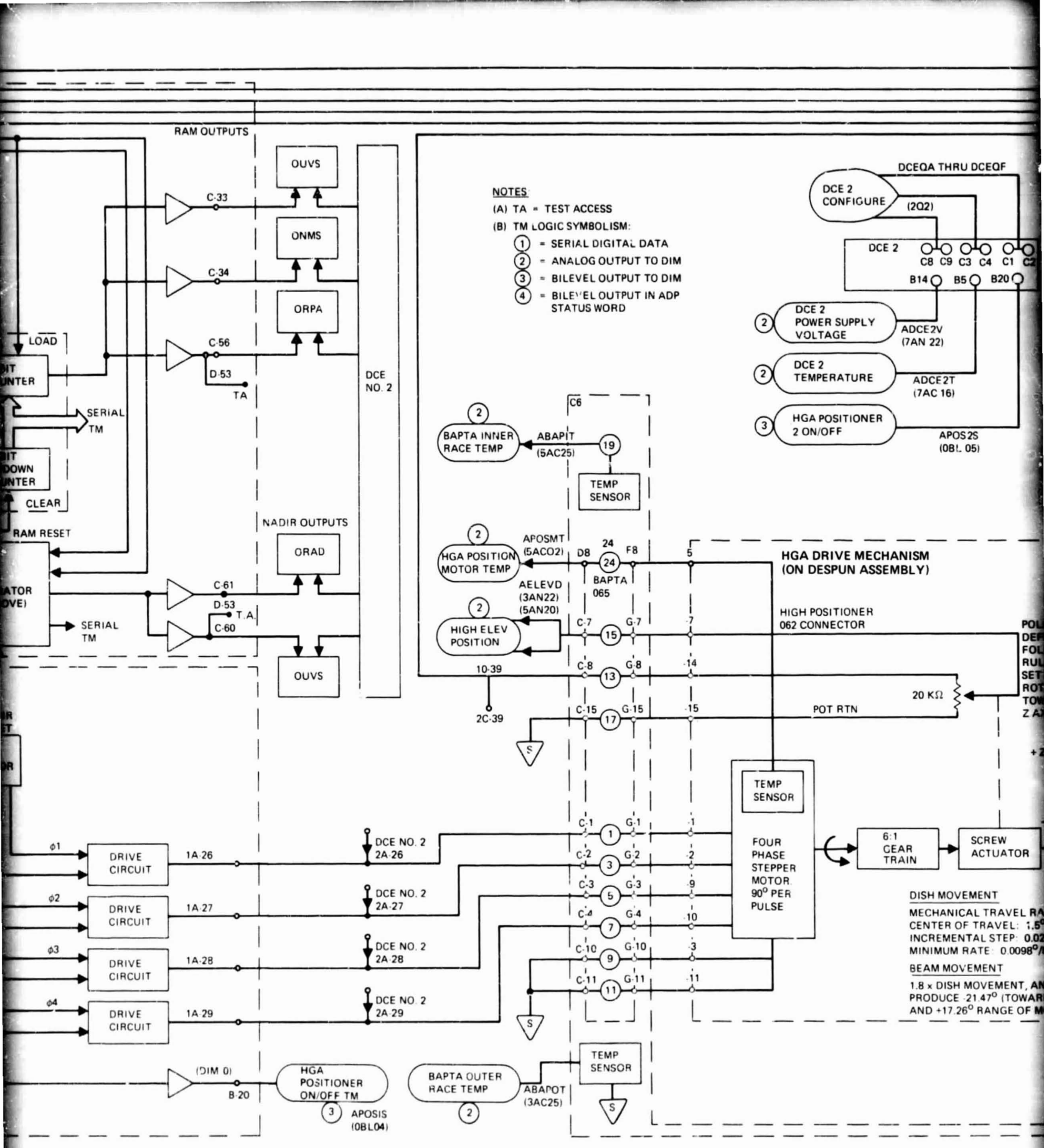
FIGURE 3.2.3.1 SUN SENSORS STAR SENSOR ELECTRONICS ADP JETS AND LATCH VALVES COMMAND AND TELEMETRY LOGIC FLOW DIAGRAM

FOLDOUT FRAME 5

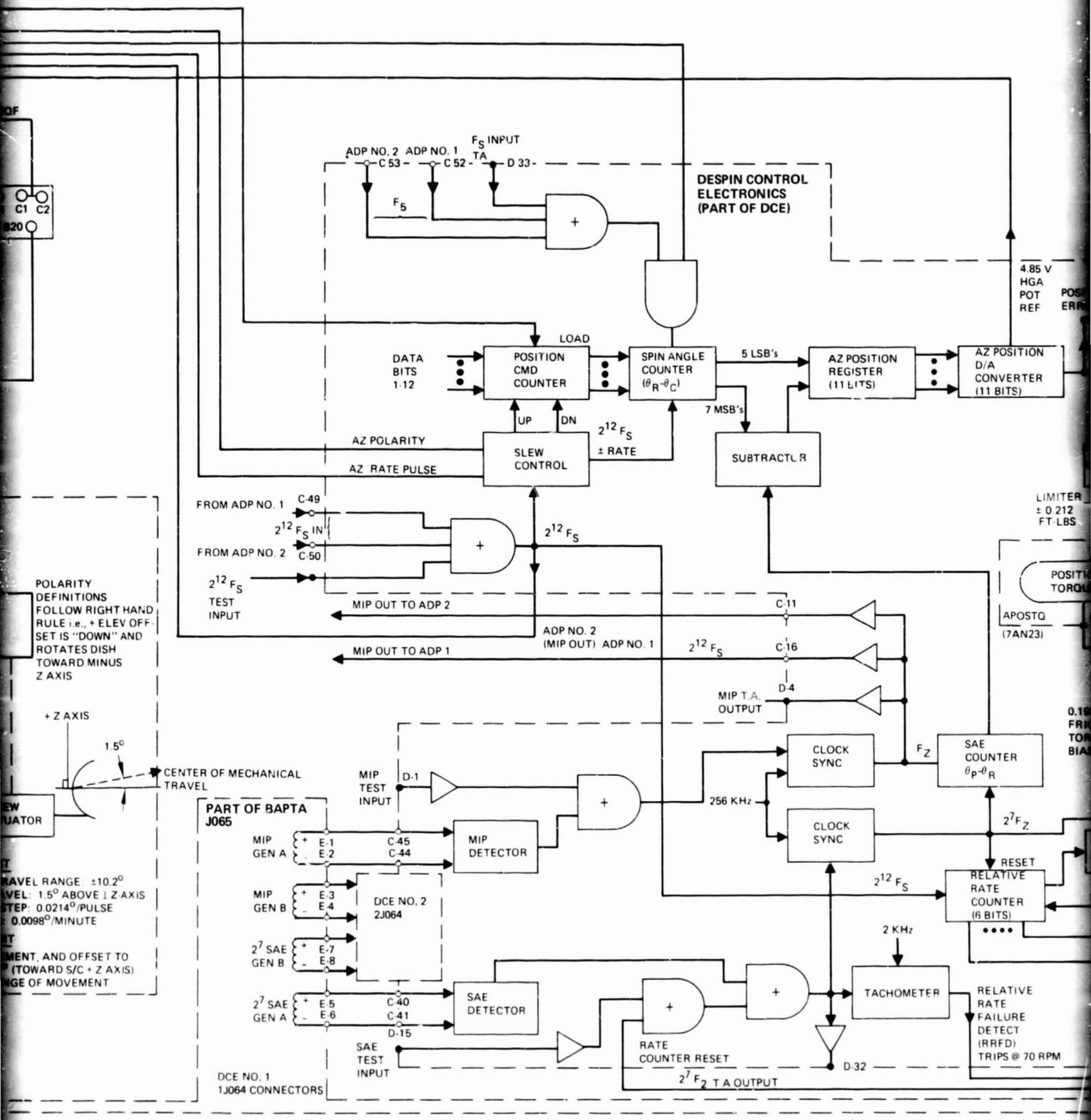


FOLDOUT FRAME

FOLDOUT FRAME 2



MOLDOUT FRAME 3



FOLDOUT FRAME 4

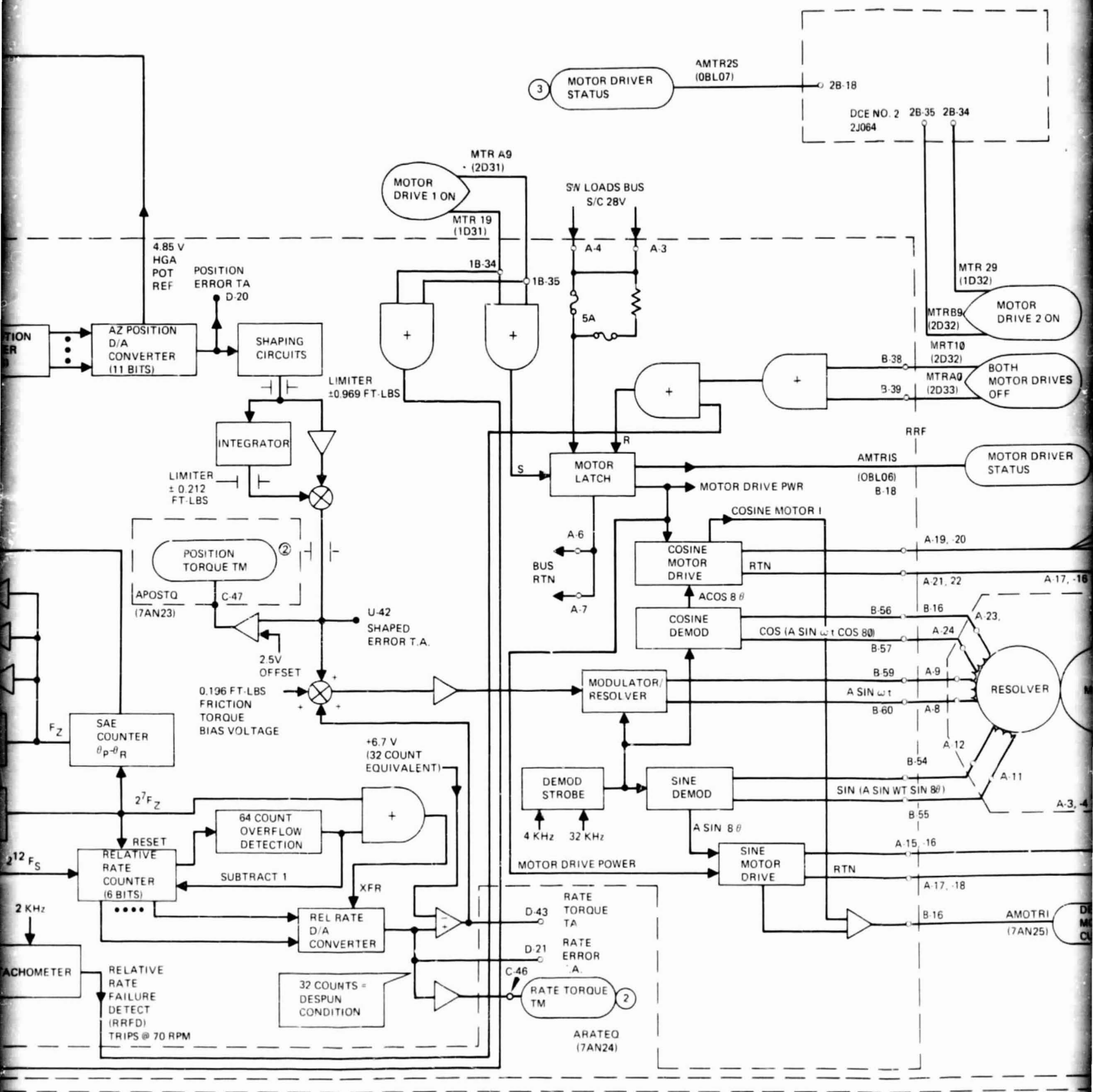


FIGURE 3.3.2.4-1. ORBITER DCE COMMAND TELEMETRY LOGIC FLOW DIAGRAM

FOLDOUT FRAME 5

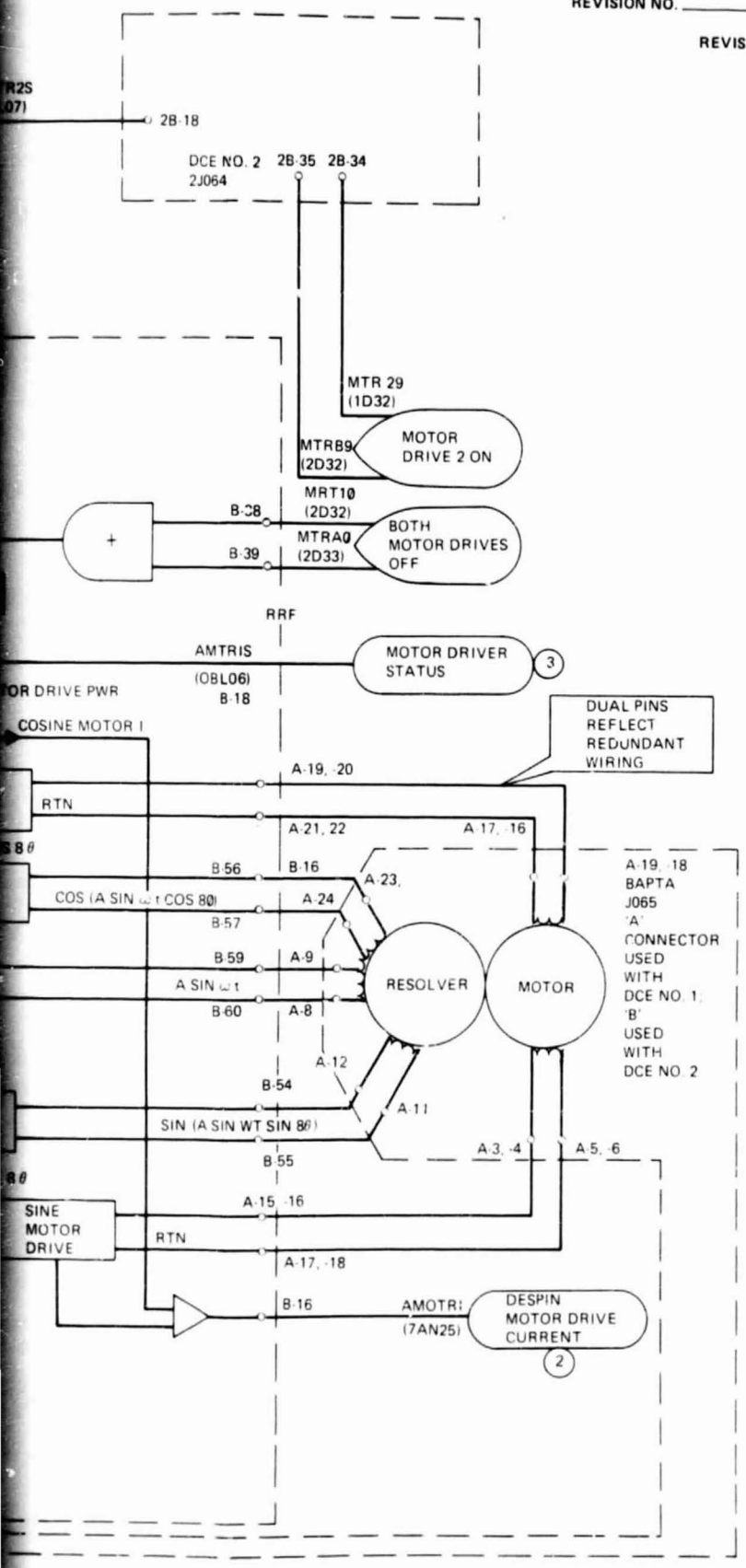
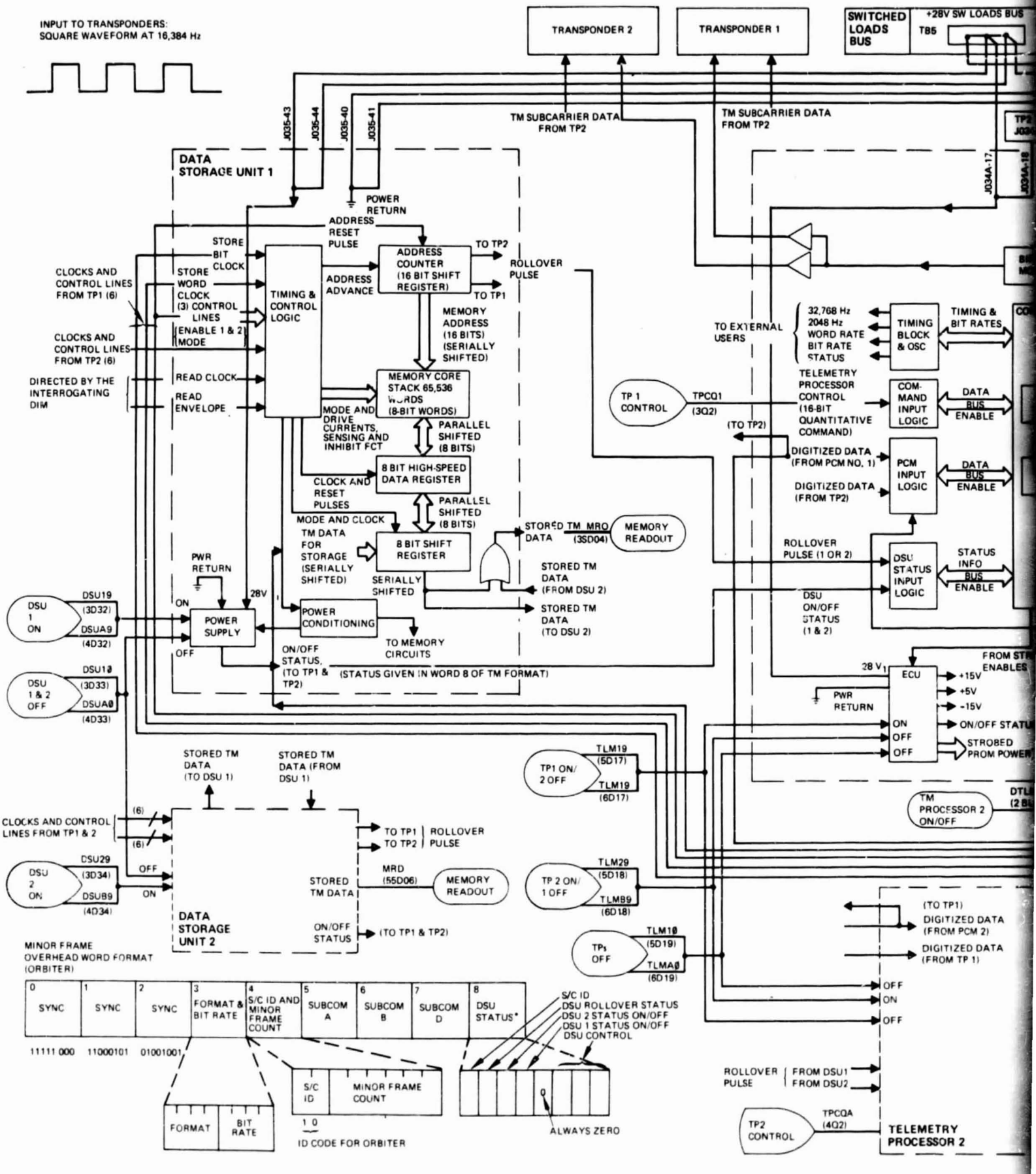


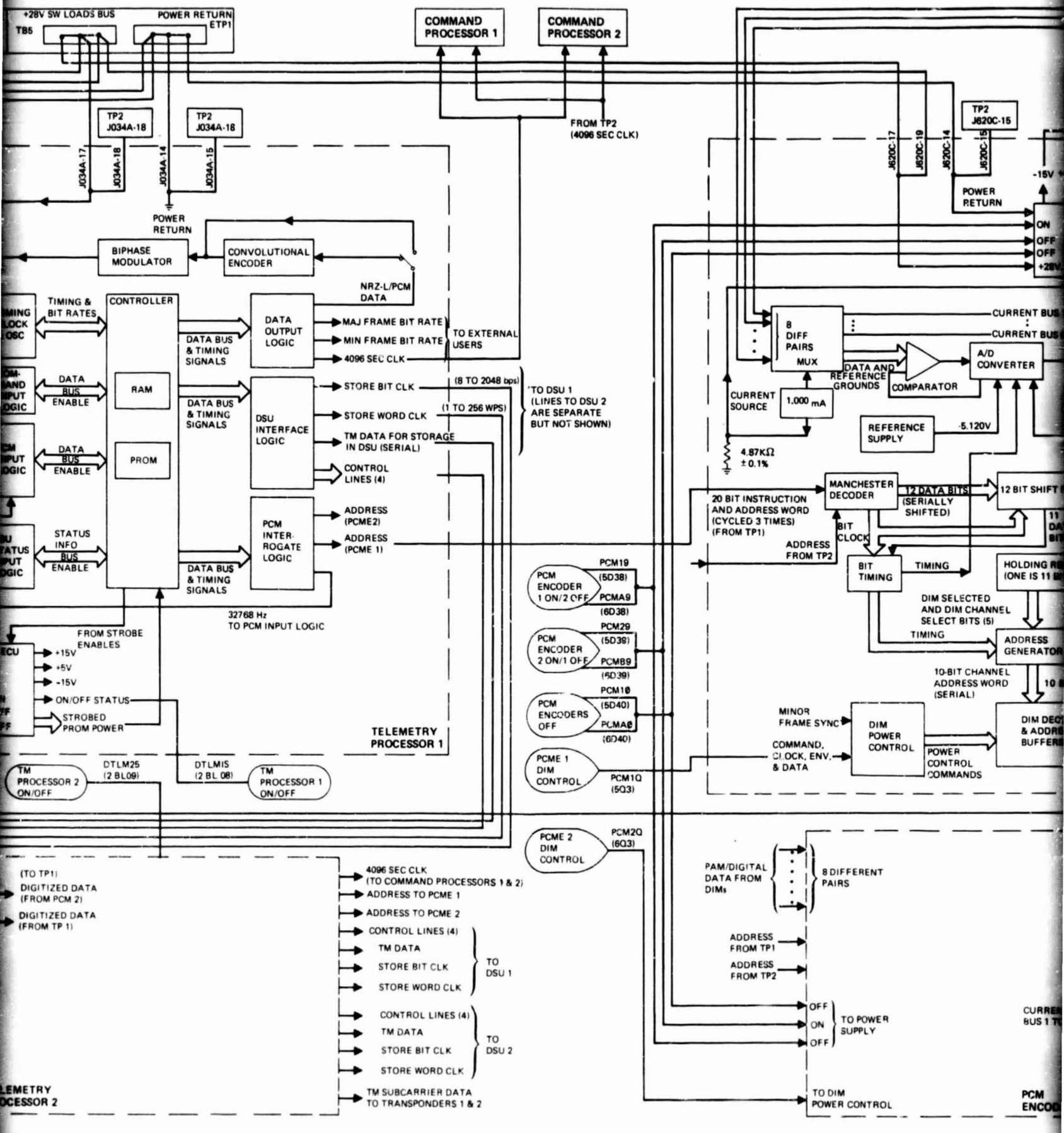
FIGURE 3.3.2.4-1. ORBITER DCE COMMAND AND TELEMETRY LOGIC FLOW DIAGRAM

SHEET NO. C-13/C-14

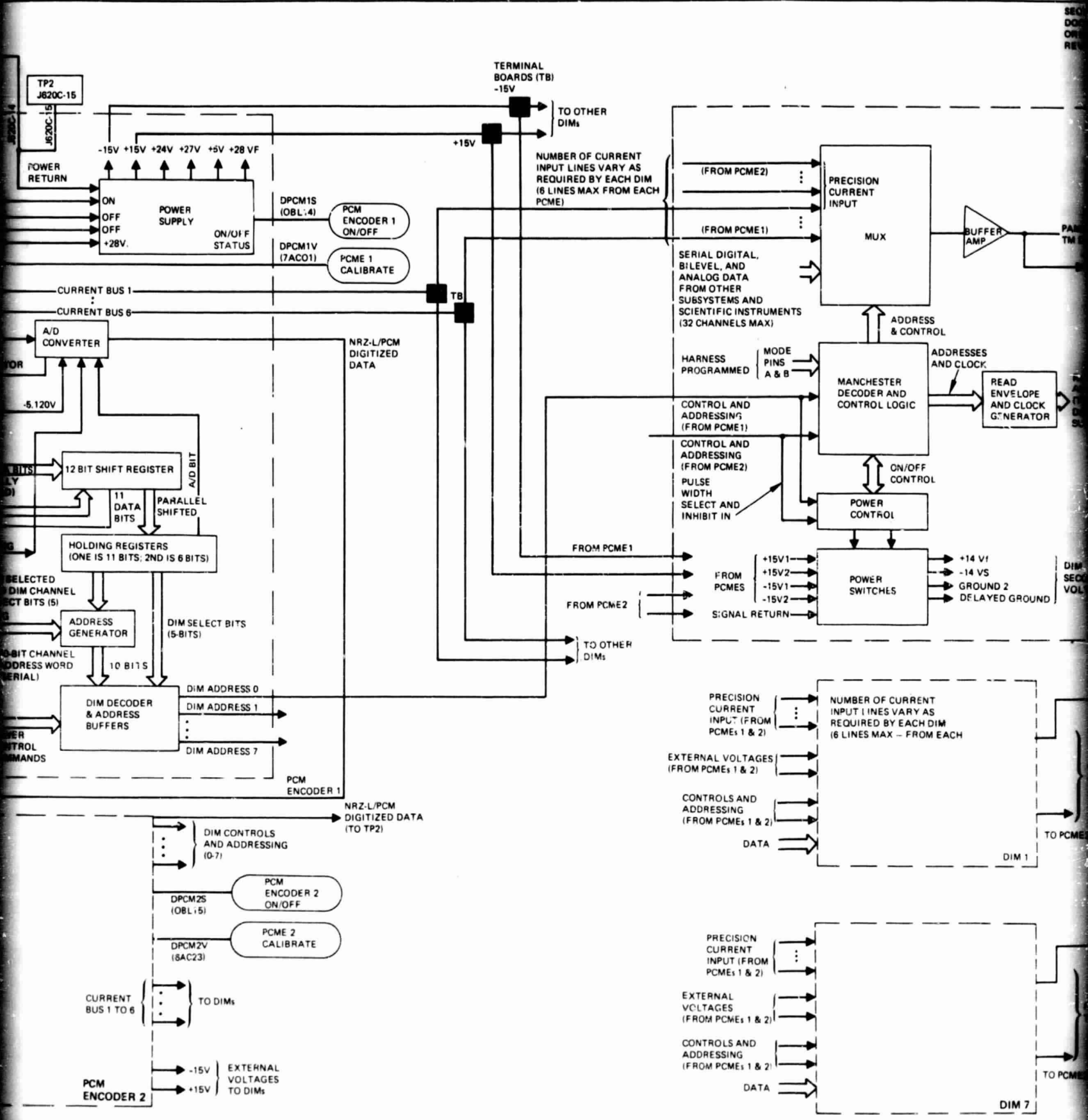
FOLDOUT FRAME 6



FOLDOUT FRAME



FOLDOUT FRAME 2



FOLDOUT FRAME 3

FIGURE 3.5.1-1. ORBITER DATA HANDLING SUBSYSTEM COMMAND AND TELEMETRY LOGIC FLOW DIAGRAM

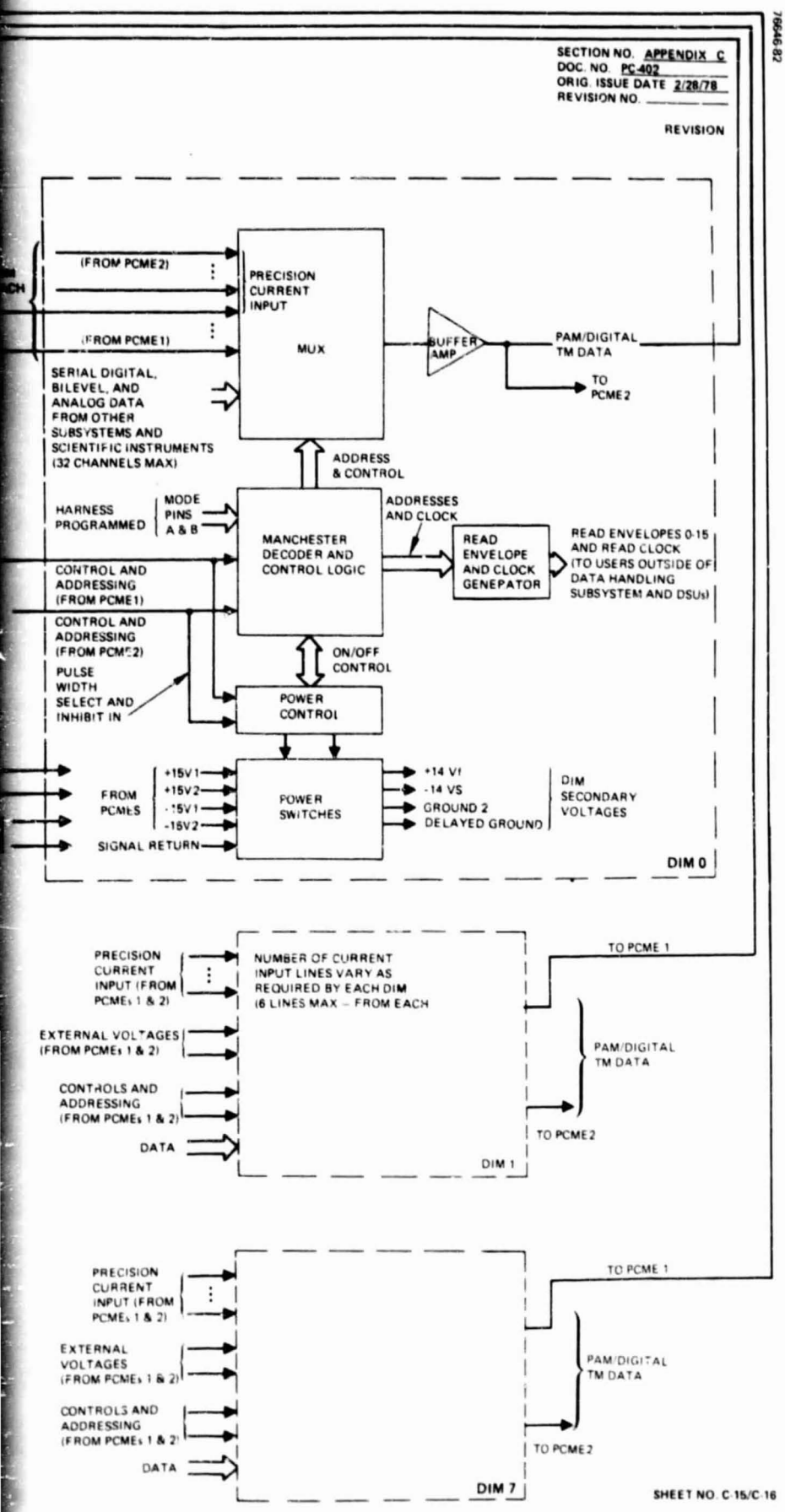
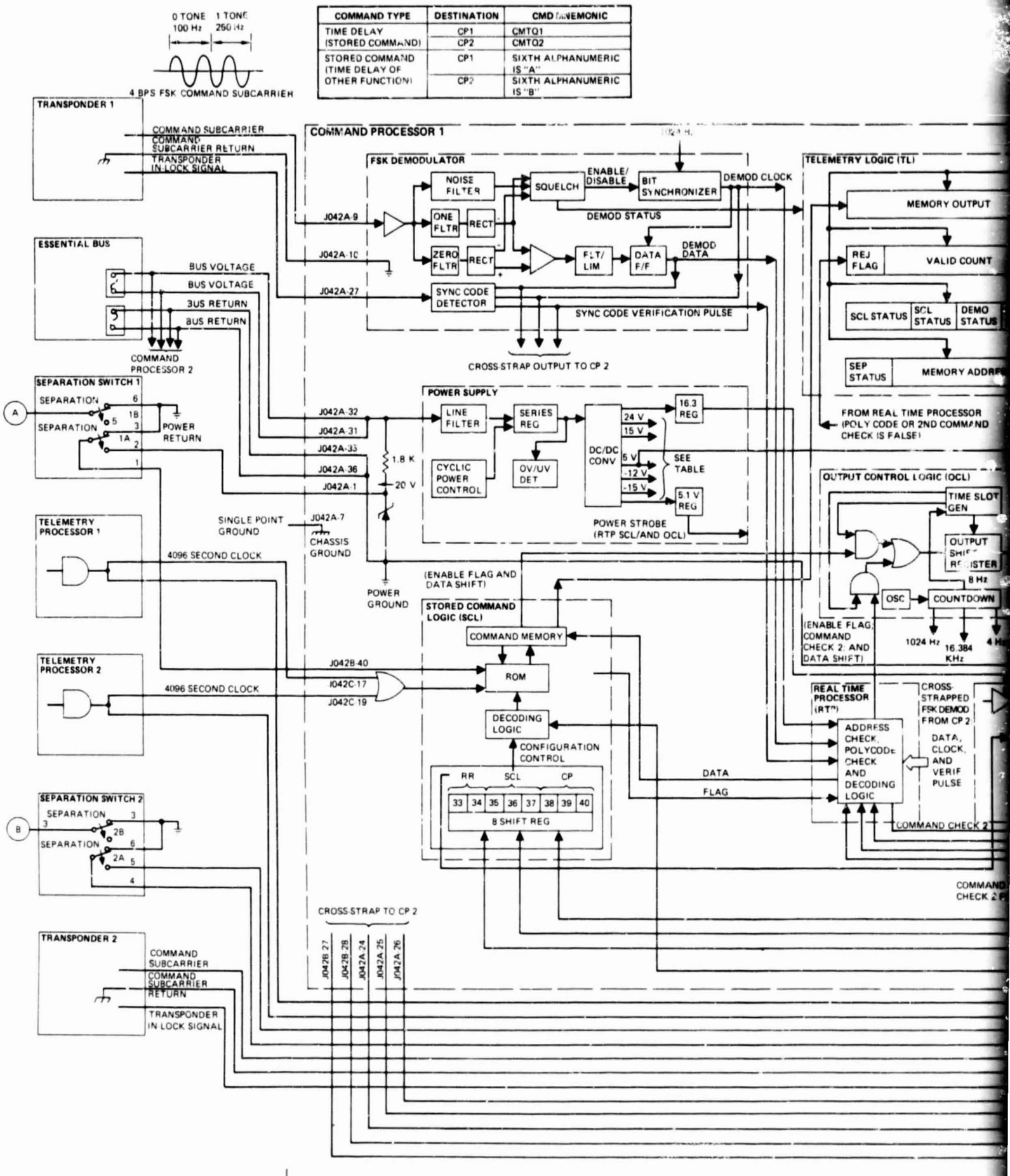
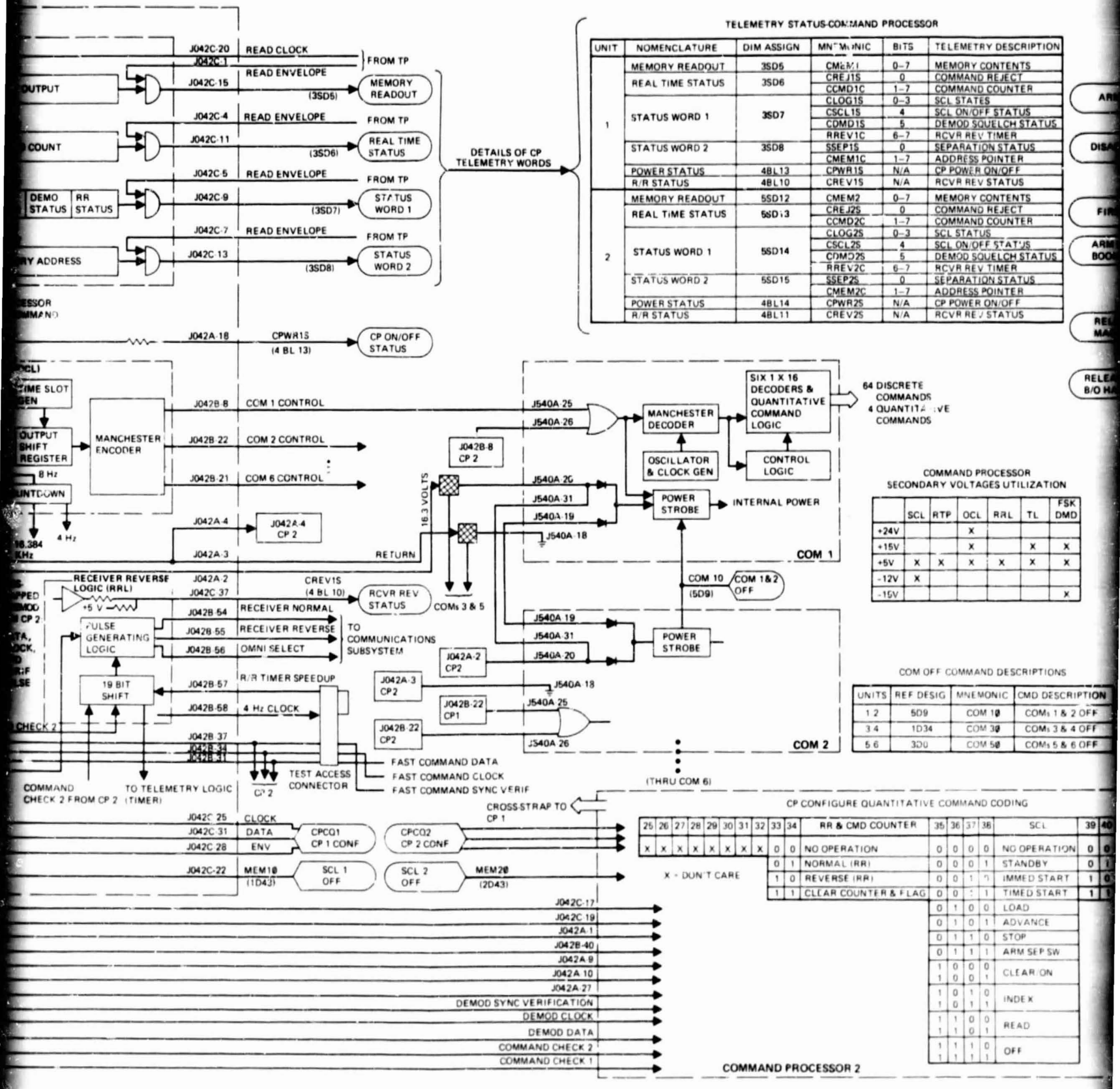


FIGURE 3.5.1-1. ORBITER DATA HANDLING SUBSYSTEM COMMAND AND TELEMETRY LOGIC FLOW DIAGRAM

FOLDOUT FRAME 4



FOLDOUT FRAME



TELEMETRY STATUS-COMMAND PROCESSOR

UNIT	NOMENCLATURE	DIM ASSIGN	MNEMONIC	BITS	TELEMETRY DESCRIPTION
1	MEMORY READOUT	3SD5	CMEM1	0–7	MEMORY CONTENTS
	REAL TIME STATUS	3SD6	CREJ1S	0	COMMAND REJECT
			CCMD1C	1–7	COMMAND COUNTER
	STATUS WORD 1	3SD7	CLOG1S	0–3	SCL STATES
			CSCL1S	4	SCL ON/OFF STATUS
			CDMD1S	5	DEMOD SQUELCH STATUS
			RREV1C	6–7	RCVR REV TIMER
	STATUS WORD 2	3SD8	SSEP1S	0	SEPARATION STATUS
			CMEM1C	1–7	ADDRESS POINTER
	POWER STATUS	4BL13	CPWR1S	N/A	CP POWER ON/OFF
	R/R STATUS	4BL10	CREV1S	N/A	RCVR REV STATUS
2	MEMORY READOUT	5SD12	CMEM2	0–7	MEMORY CONTENTS
	REAL TIME STATUS	5SD13	CREJ2S	0	COMMAND REJECT
			CCMD2C	1–7	COMMAND COUNTER
	STATUS WORD 1	5SD14	CLOG2S	0–3	SCL STATUS
			CSCL2S	4	SCL ON/OFF STATUS
			CDMD2S	5	DEMOD SQUELCH STATUS
			RREV2C	6–7	RCVR REV TIMER
	STATUS WORD 2	5SD15	SSEP2S	0	SEPARATION STATUS
			CMEM2C	1–7	ADDRESS POINTER
	POWER STATUS	4BL14	CPWR2S	N/A	CP POWER ON/OFF
	R/R STATUS	4BL11	CREV2S	N/A	RCVR REV STATUS

COMMAND PROCESSOR SECONDARY VOLTAGES UTILIZATION

	SCL	RTP	OCL	RRL	TL	FSK DMD
+24V			X			
+15V			X		X	X
+5V	X	X	X	X	X	X
-12V	X					
-15V						X

COM OFF COMMAND DESCRIPTIONS

UNITS	REF DESIG	MNEMONIC	CMD DESCRIPTION
1 2	5D9	COM 1Ø	COMs 1 & 2 OFF
3 4	1D34	COM 3Ø	COMs 3 & 4 OFF
5 6	3D0	COM 5Ø	COMs 5 & 6 OFF

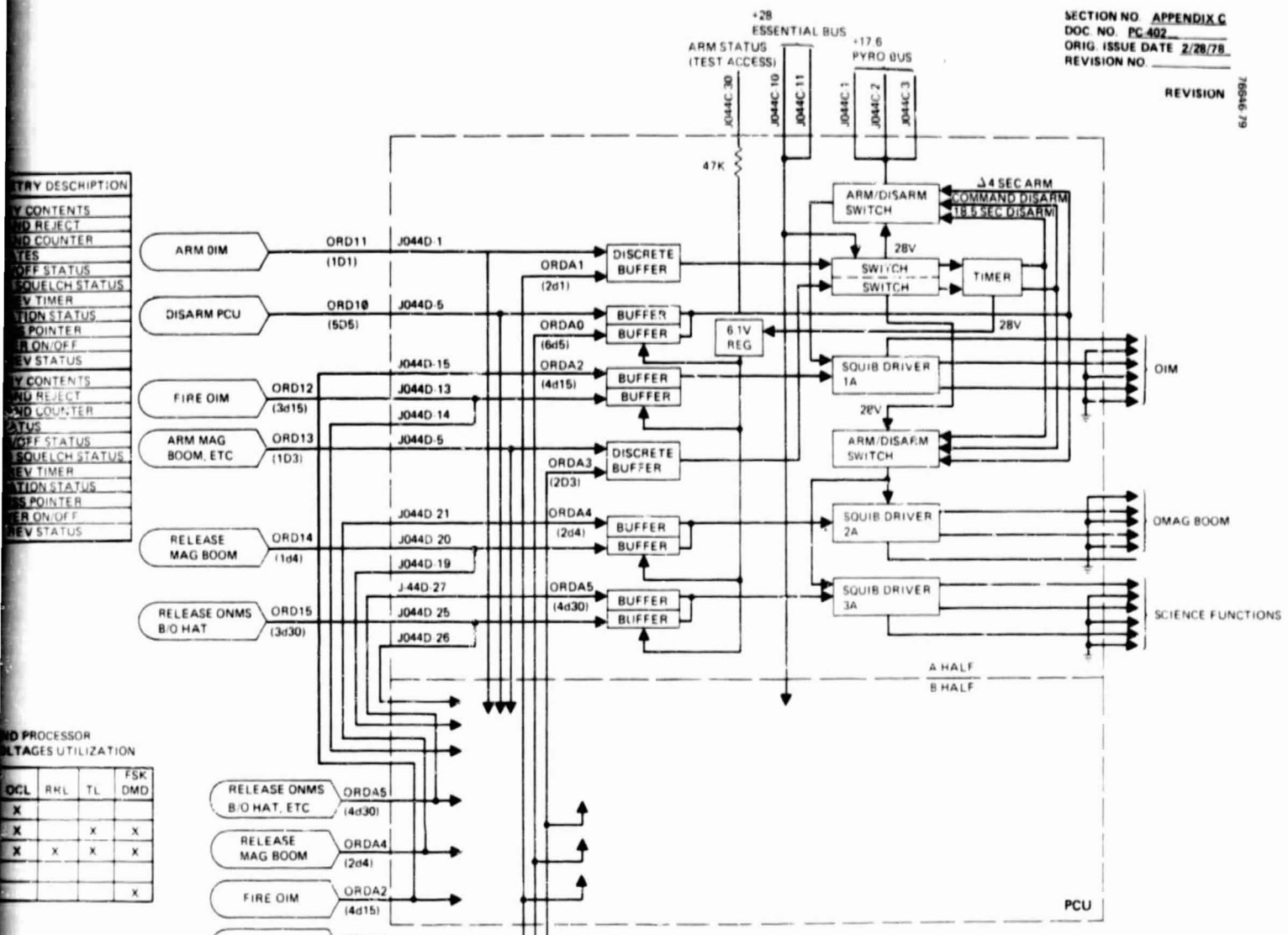
CP CONFIGURE QUANTITATIVE COMMAND CODING

25	26	27	28	29	30	31	32	33	34	RR & CMD COUNTER	35	36	37	38	SCL	39	40
X	X	X	X	X	X	X	X	0	0	NO OPERATION	0	0	0	0	NO OPERATION	0	0
								0	1	NORMAL (RR)	0	0	0	1	STANDBY	0	1
								1	0	REVERSE (RR)	0	0	1	0	IMMED START	1	0
								1	1	CLEAR COUNTER & FLAG	0	0	1	1	TIMED START	1	1
											0	1	0	0	LOAD		
											0	1	0	1	ADVANCE		
											0	1	1	0	STOP		
											0	1	1	1	ARM SEP SW		
											1 1	0 0	0 0	0 1	CLEAR ON		
											1 1	0 0	1 1	0 1	INDEX		
											1 1	1 1	0 0	0 1	READ		
											1 1	1 1	1 1	0 1	OFF		

X = DON'T CARE

FOLDOUT FRAME 2

SECTION NO. APPENDIX C
DOC. NO. PC-402
ORIG. ISSUE DATE 2/28/78
REVISION NO. _____
REVISION



TRY DESCHIPTION
Y CONTENTS
ND REJECT
ND COUNTER
TES
OFF STATUS
SQUELCH STATUS
EV TIMER
TION STATUS
S POINTER
R ON/OFF
EV STATUS
Y CONTENTS
ND REJECT
ND COUNTER
ATUS
OFF STATUS
SQUELCH STATUS
EV TIMER
TION STATUS
S POINTER
ER ON/OFF
EV STATUS

ND PROCESSOR
LTAGES UTILIZATION

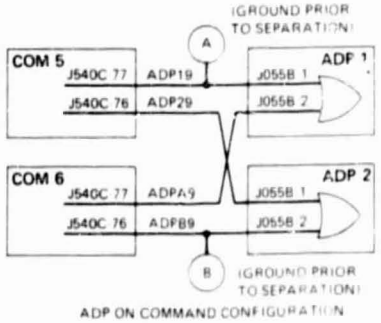
OCL	RHL	TL	FSK DMD
X			
X		X	X
X	X	X	X
			X

MAND DESCRIPTIONS

NEMONIC	CMD DESCRIPTION
COM 1Ø	COMs 1 & 2 OFF
COM 3Ø	COMs 3 & 4 OFF
COM 5Ø	COMs 5 & 6 OFF

MAND CODING

37	38	SCI	39	40	SELECT CP
0	0	NO OPERATION	0	0	NEITHER
0	1	STANDBY	0	1	CP 1
1	0	IMMED START	1	0	CP 2
1	1	TIMED START	1	1	EITHER
0	0	LOAD			
0	1	ADVANCE			
1	0	STOP			
1	1	ARM SEP SW			
0	0	CLEAR ON			
0	1				
1	0	INDEX			
1	1				
0	0	READ			
0	1				
1	0	OFF			
1	1				



ADP ON COMMAND CONFIGURATION

ADP ON COMMAND

UNIT	REF DESIG	MNEMONIC	STATUS
1	5D28	ADP19	GROUNDED BEFORE SEP
	6D28	ADPA9	UNGROUNDED
2	5D31	ADP29	UNGROUNDED
	6D31	ADPB9	GROUNDED BEFORE SEP

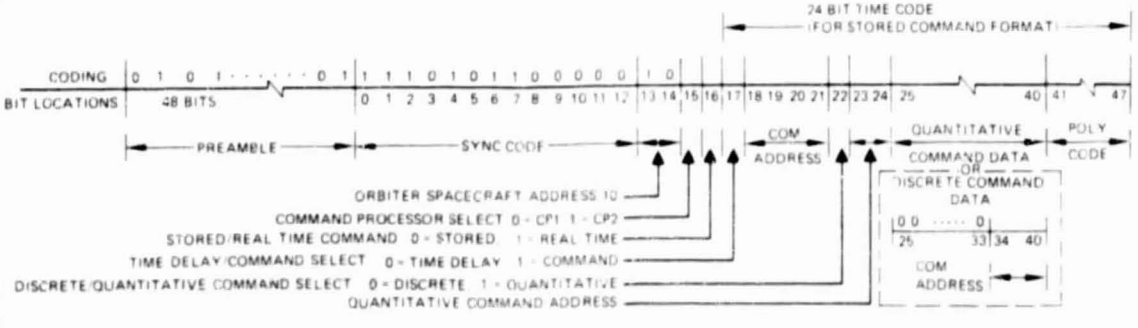
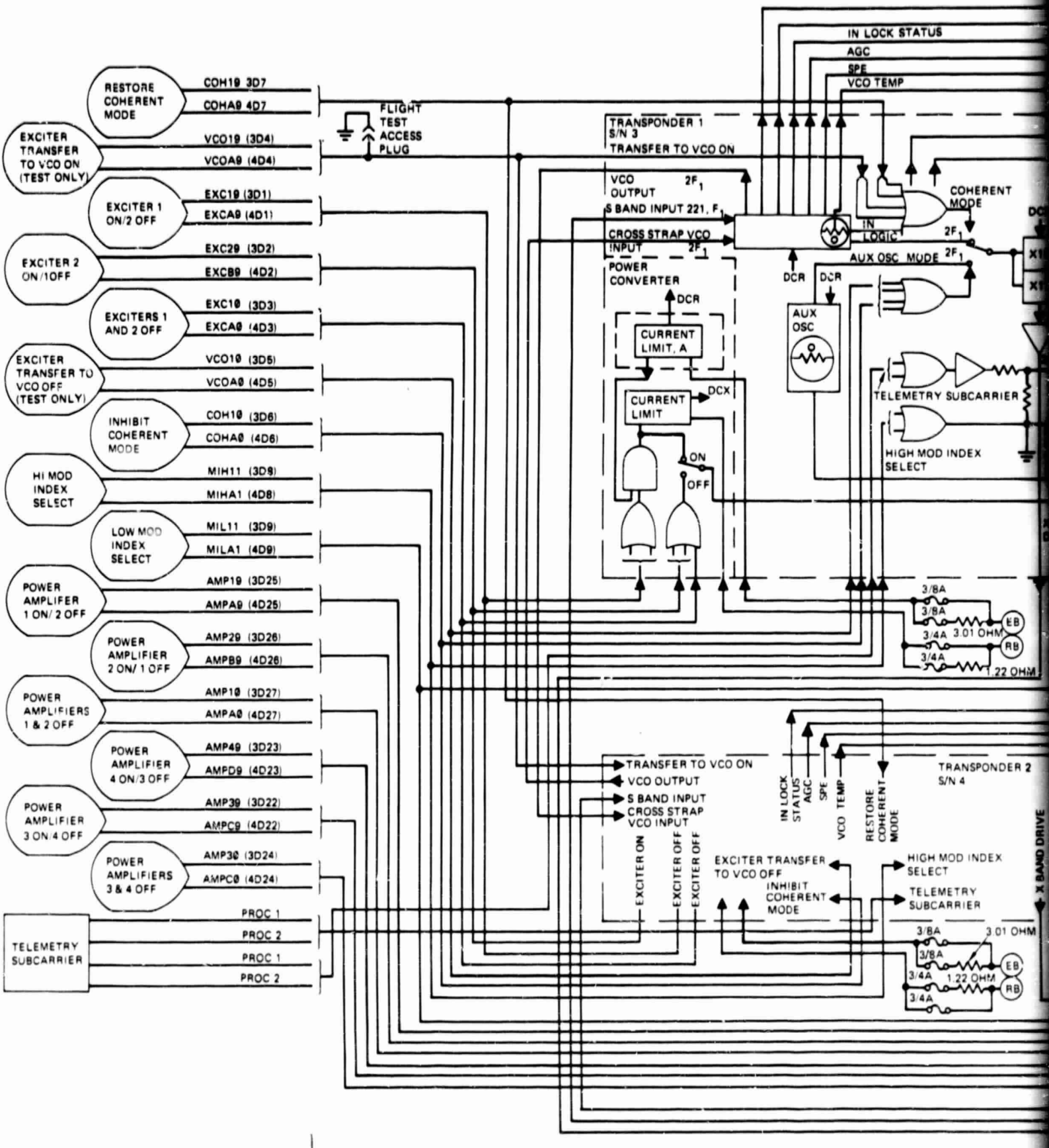


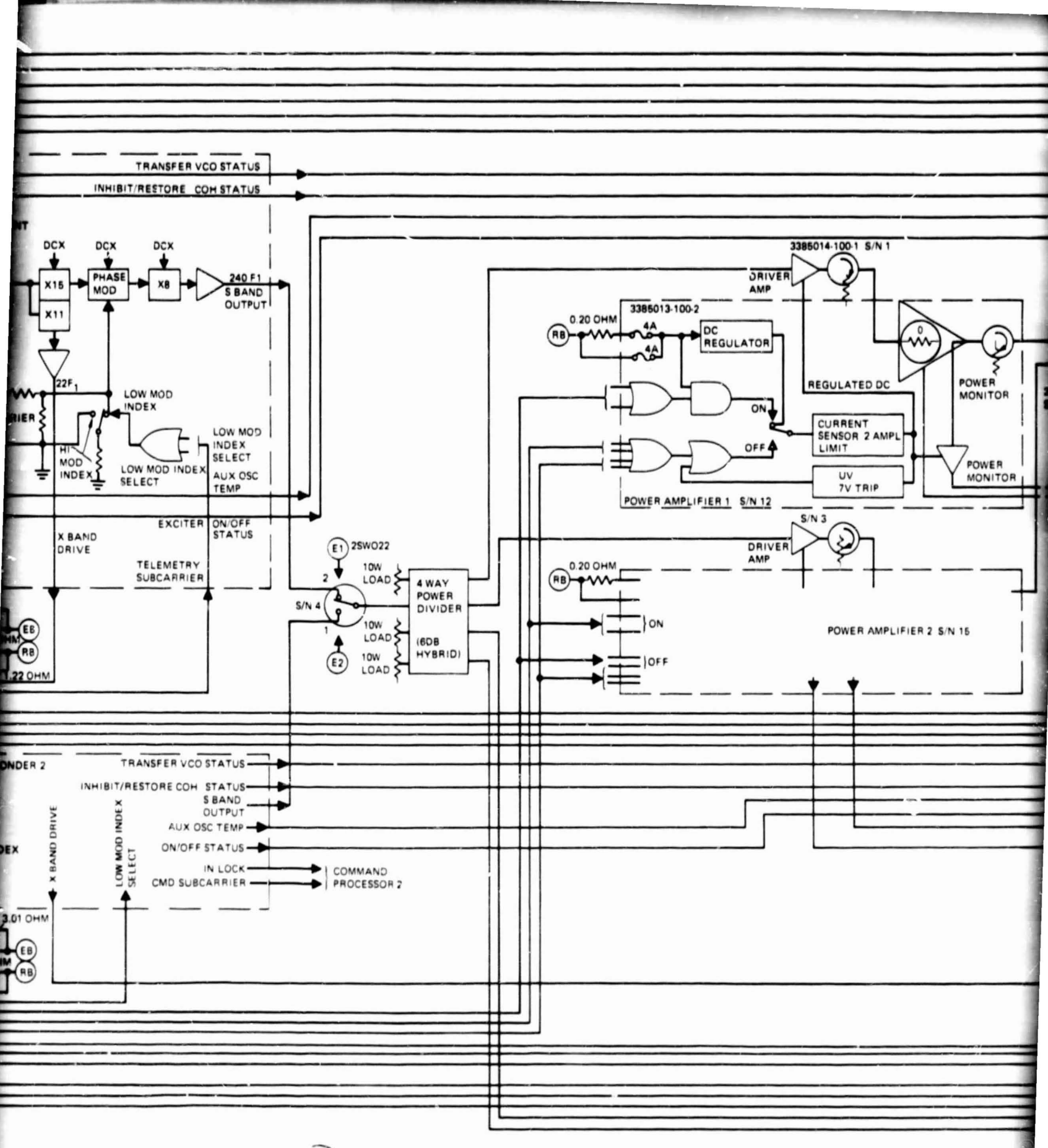
FIGURE 3.6.1.1. ORBITER COMMAND SUBSYSTEM COMMAND AND TELEMETRY LOGIC FLOW DIAGRAM

SHEET NO. C 17/C 18

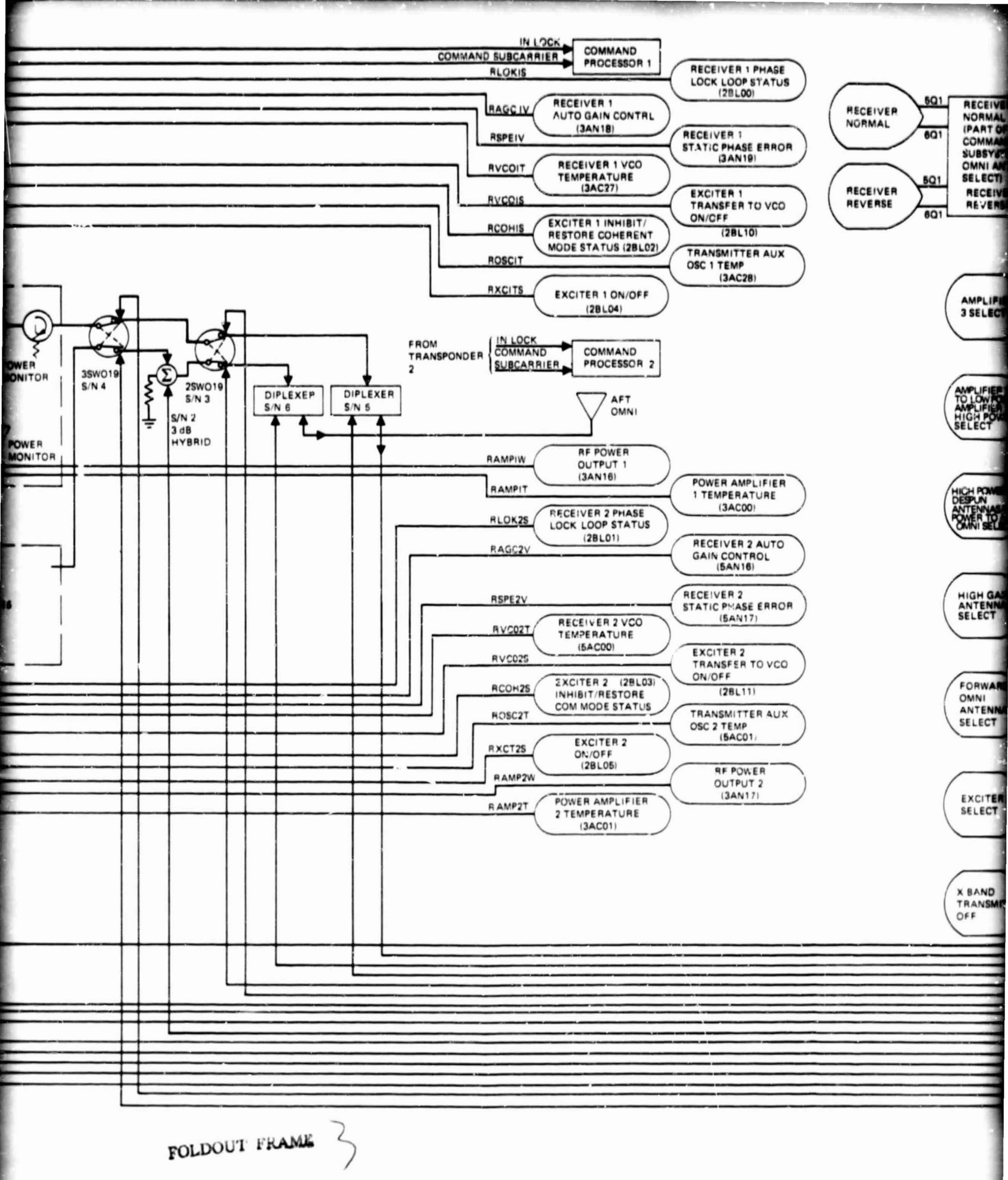
FOLDOUT FRAME 3



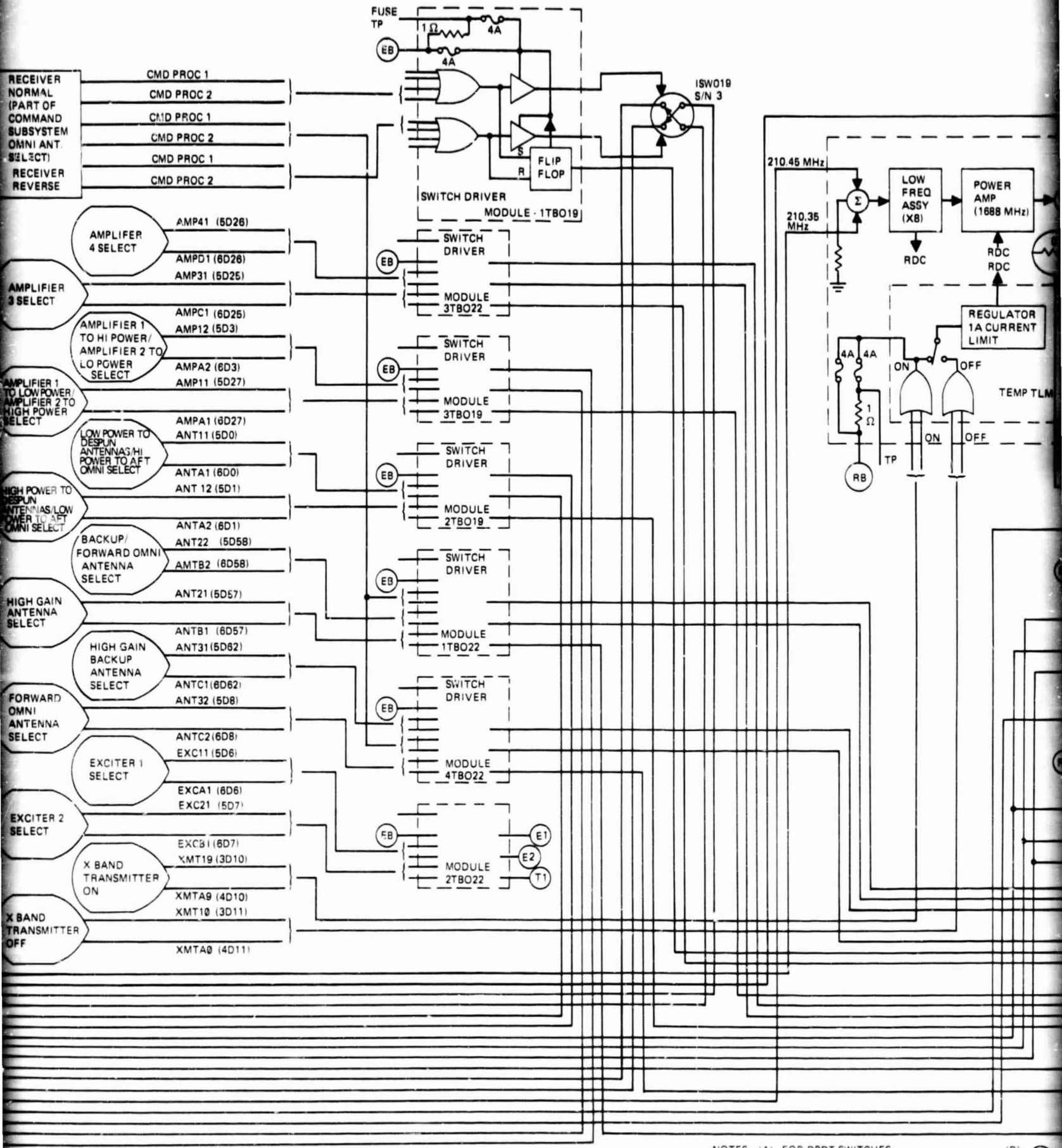
FOLDOUT FRAME



FOLDOUT FRAME 2



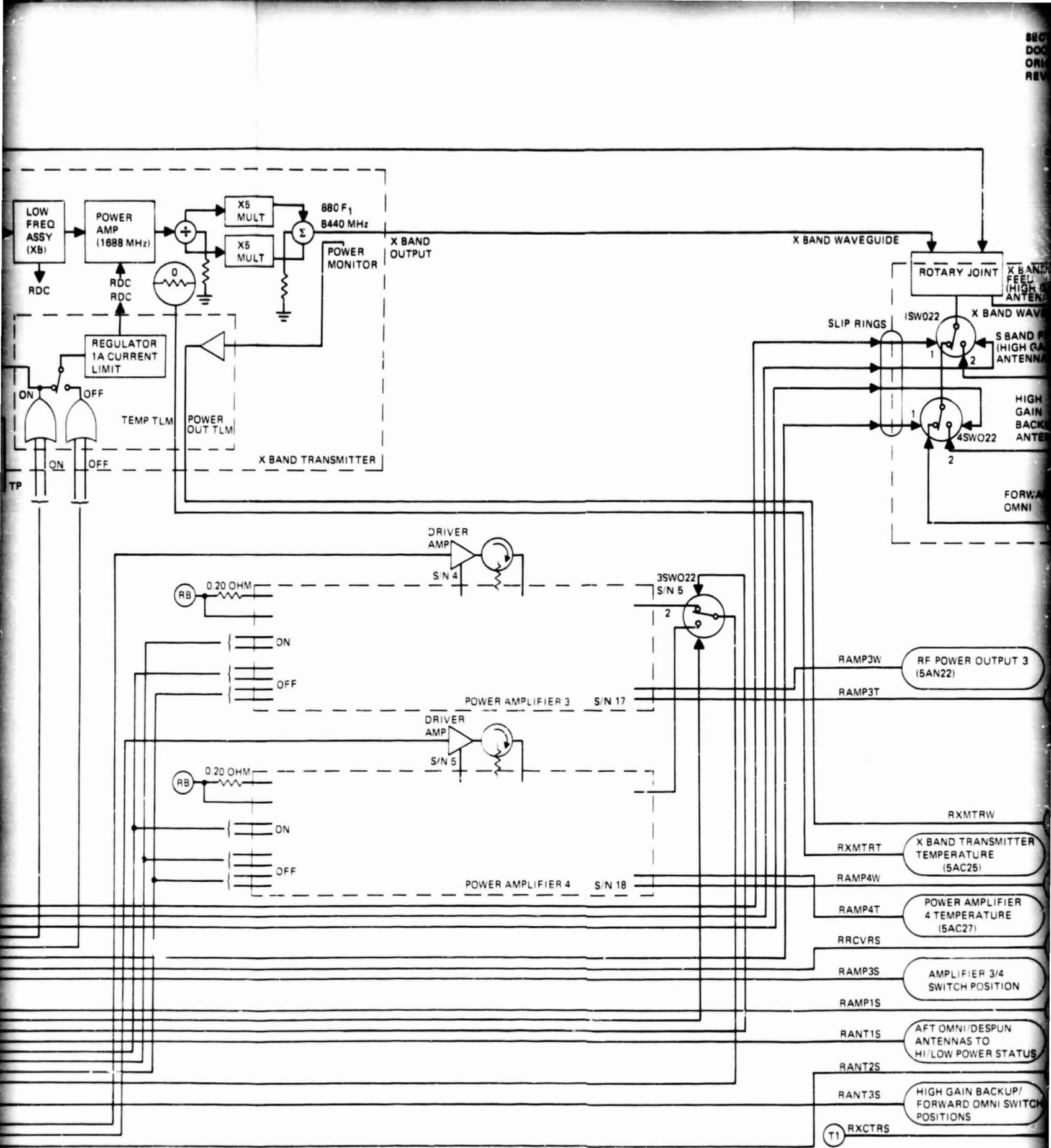
FOLDOUT FRAME 3



NOTES: (A) FOR DPDT SWITCHES THE "2" POSITION IS SHOWN AS SOLID LINES; THE "1" POSITION IS SHOWN AS DASHED LINES.

(B) EB RB

FOLDOUT FRAME 4



FOLDOUT FRAME

FIGURE 3.7.1-4. ORBITER COMMUNICATION SUBSYSTE AND TELEMETRY LOGIC FLOW DIAGR

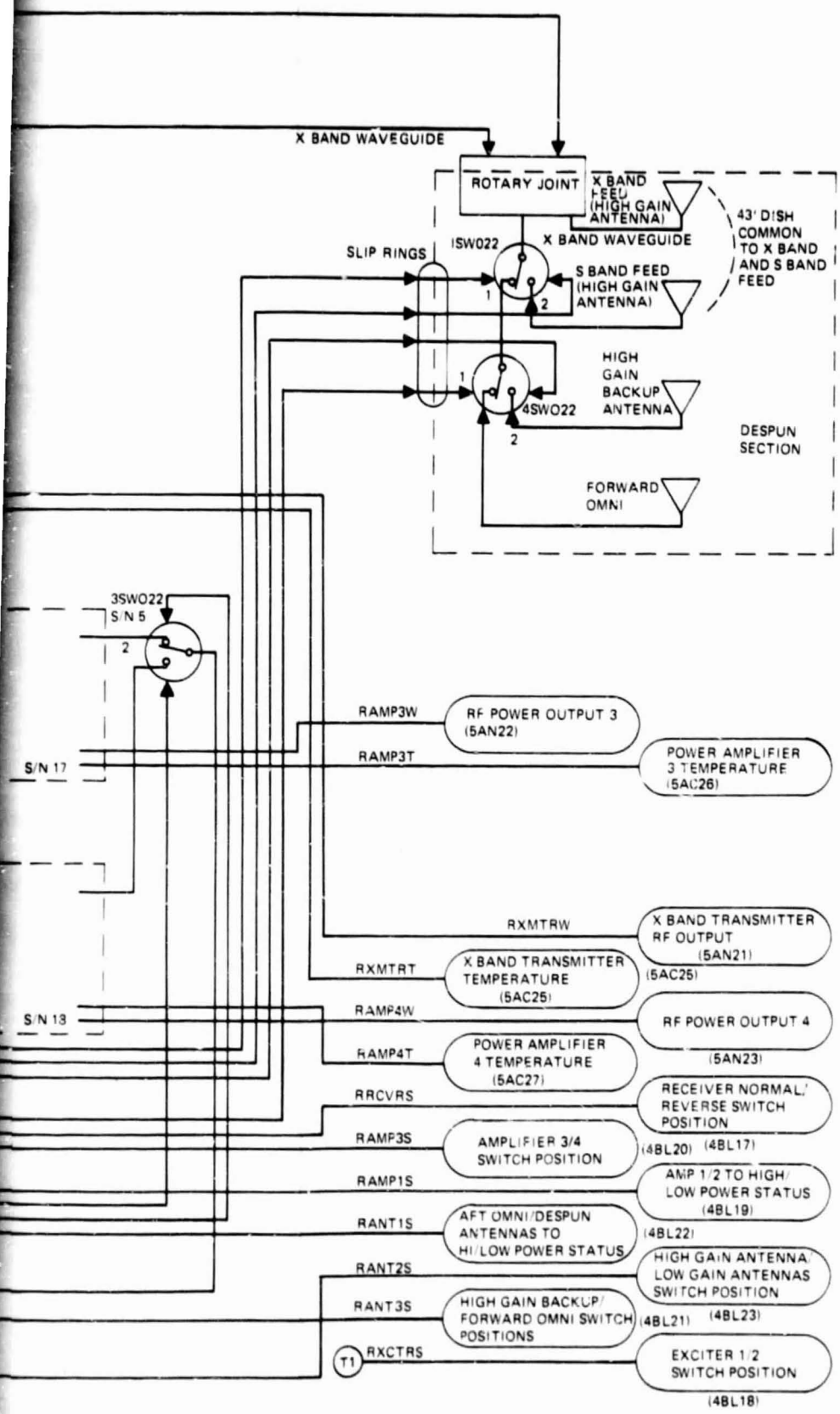
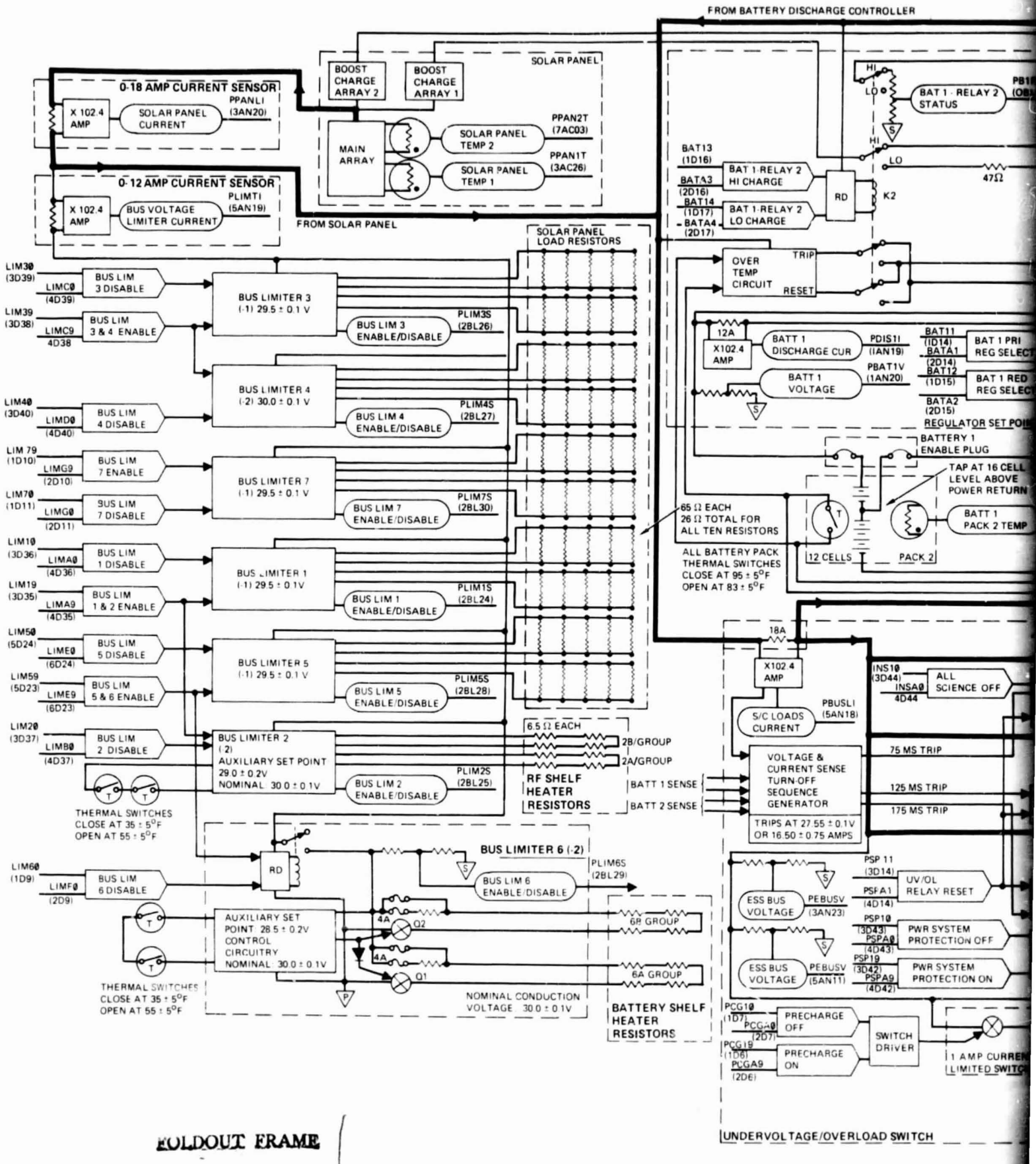
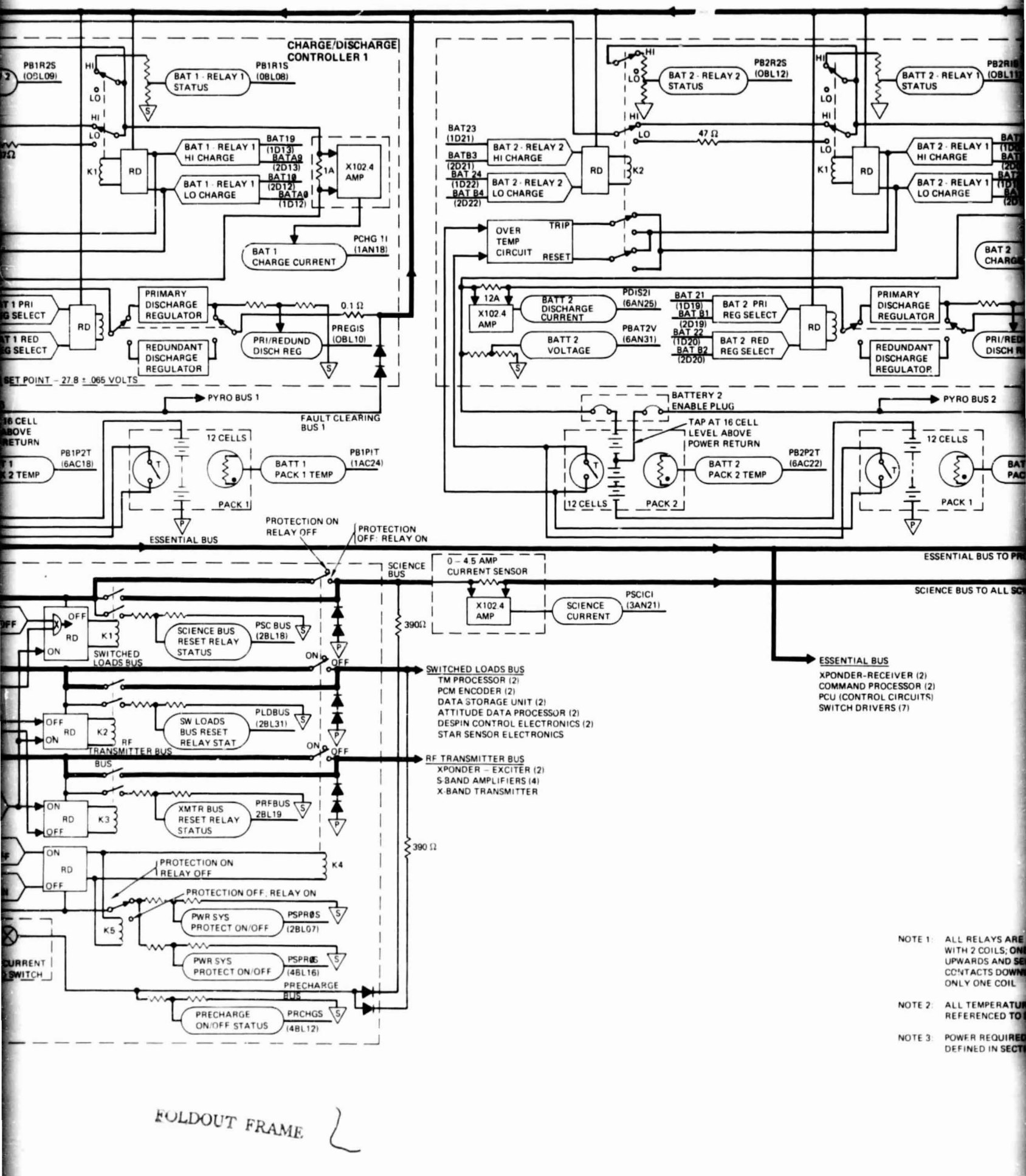


FIGURE 3.7.1-4. ORBITER COMMUNICATION SUBSYSTEM COMMAND AND TELEMETRY LOGIC FLOW DIAGRAM

FOLDOUT FRAME 6



FOLDOUT FRAME



FOLDOUT FRAME

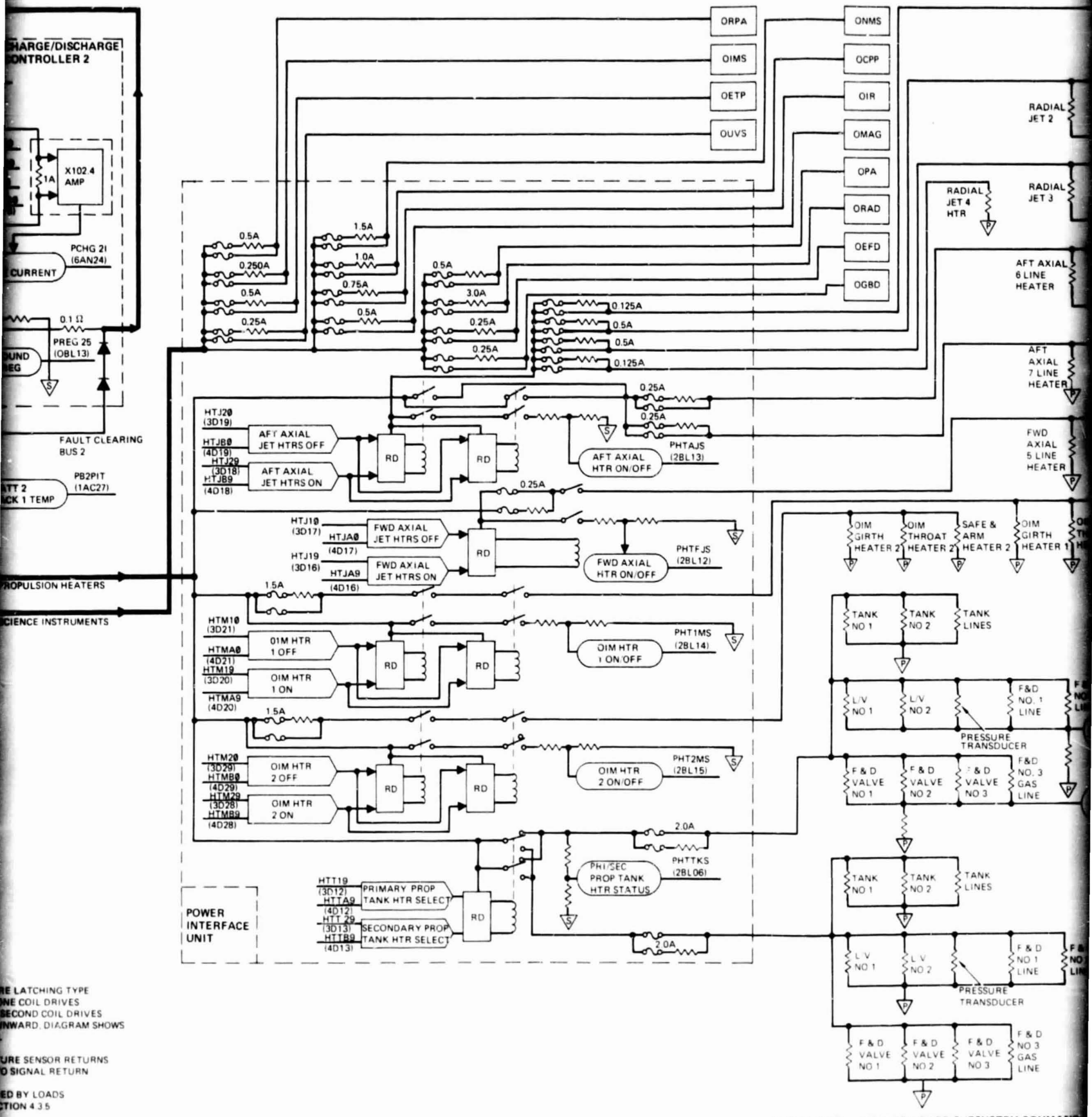
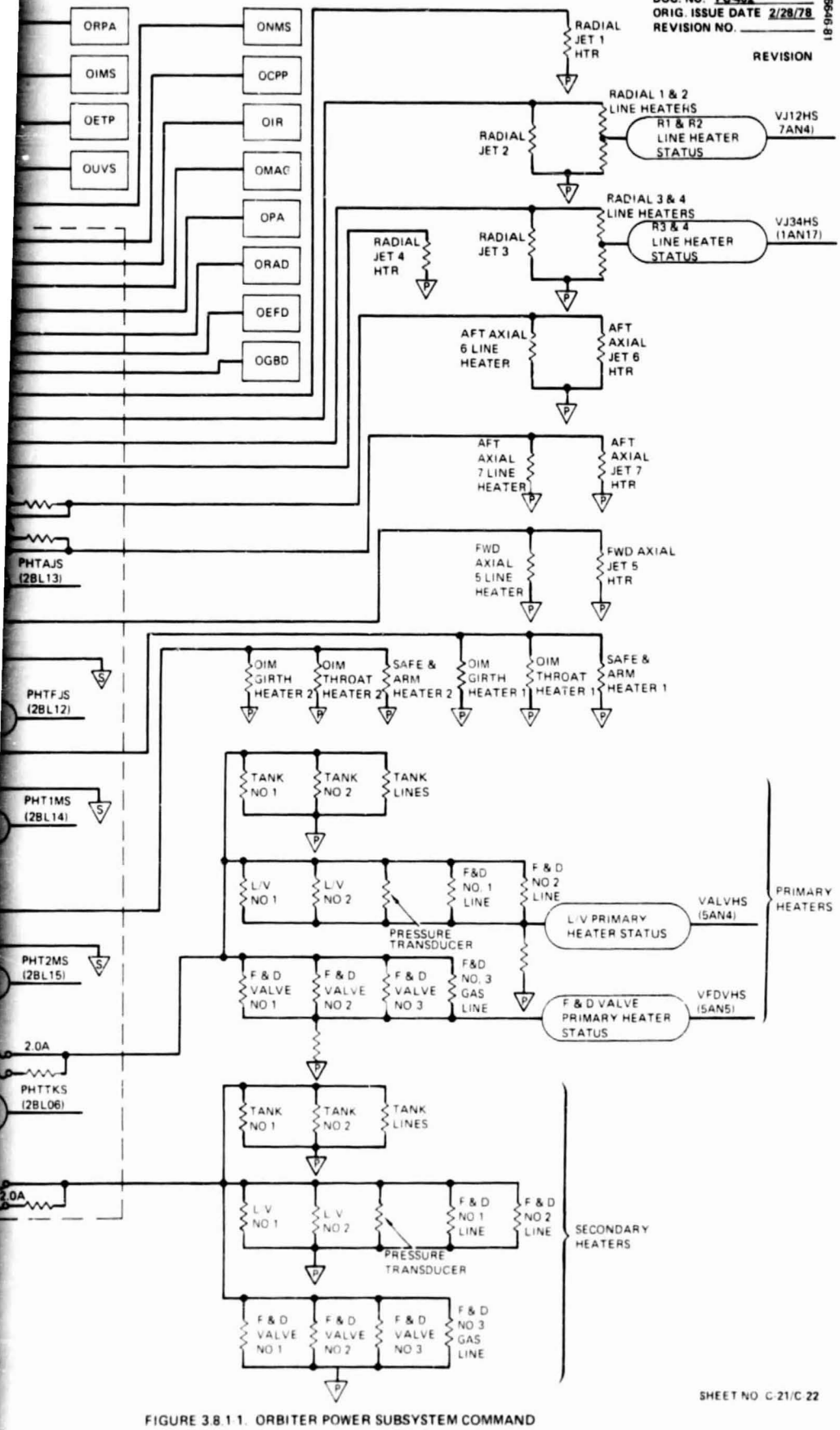


FIGURE 3.8.1-1. ORBITER POWER SUBSYSTEM COMMAND AND TELEMETRY LOGIC FLOW DIAGRAM

FOLDOUT FRAME 3



SHEET NO C-21/C-22

FIGURE 3.8.1.1. ORBITER POWER SUBSYSTEM COMMAND AND TELEMETRY LOGIC FLOW DIAGRAM

4

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